Experimental Development of Compressor and Turbine Performance Maps for Turbomachinery

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by,

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Turbomachinery performance maps are used to reveal the on and off design performance of compressors and turbines. These maps plot the component’s pressure ratio versus the corrected flow rate for multiple engine rotor speeds. In this experiment, a Boeing T50-BO-8A converted turbojet engine was operated at a wide variety of rotor speeds with orifice plates of different sizes to vary the flow rate. The corrected flow rate through the turbojet ranged from 1.32 to 2.55 lb/s. The pressure ratios for the compressor and turbine ranged from 2.0 to 4.3 and from 1.7 to 4.3, respectively. The resulting performance maps showed the correct overall trend of increasing flow rate and pressure ratio with higher rotor speeds. However, the individual speed lines were very inconsistent and lacked discernable trends. It was concluded that bench testing the compressor and turbine components is the ideal way to develop the performance maps.

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Nomenclature

\( A \) = area, \( \text{in}^2 \)
\( C_p \) = specific heat at constant pressure
\( C_v \) = specific heat at constant volume
\( F \) = thrust, \( \text{lbf} \)
\( H \) = enthalpy, \( \text{BTU/lb} \)
\( J \) = conversion factor, 778 (\( \text{ft-lb/BTU} \))
\( M \) = Mach number
\( N \) = Engine speed, \%RPM
\( P \) = pressure, \( \text{lb/in}^2 \)
\( R \) = universal gas constant
\( T \) = temperature, \( ^\circ \text{R} \)
\( V \) = velocity, \( \text{ft/sec} \)
\( f \) = fuel/air ratio
\( g \) = acceleration of gravity, \( \text{ft/sec}^2 \)
\( m \) = mass flow rate, \( \text{slugs/sec} \)
\( n \) = engine speed, \( \text{rev/min} \)
\( \psi \) = flow rate, \( \text{lb/sec} \)
\( \delta \) = ambient pressure ratio
\( \eta \) = isentropic efficiency
\( \gamma \) = ratio of specific heats
\( \pi \) = pressure ratio
\( \rho \) = density, \( \text{slugs/ft}^3 \)
\( \theta \) = absolute ambient temperature ratio

Subscripts

\( a \) = ambient
\( \text{air} \) = air only (non-combusted)
\( \text{comp} \) = compressor
\( \text{corr} \) = corrected
\( \text{exh} \) = exhaust, air + fuel (combusted)
\( \text{exit} \) = engine exit
\( \text{fuel} \) = fuel
\( \text{i/inlet} \) = inlet
\( \text{ideal} \) = ideal/isentropic
\( \text{turb} \) = turbine
\( 0 \) = stagnation
\( 2 \) = inlet exit/compressor inlet
\( 3 \) = compressor exit/combustor inlet
\( 4 \) = combustor exit/turbine inlet
\( 5 \) = turbine exit
\( 6 \) = nozzle inlet
\( 7 \) = nozzle exit
I. Introduction

PERFORMANCE maps for turbomachinery, also referred to as operating maps, are an essential part of compressor and turbine design. These maps consist of simple, two-dimensional plots of pressure ratio versus corrected flow rate. They allow the designer and operator to easily determine how the machinery will perform at nearly any on or off design condition. Performance maps also reveal the operating limits of the machinery to ensure safe operation of the device. Finally, these maps can show the efficiency of the turbine and compressor and reveal their most favorable operating points.

A. Compressors

In turbomachinery, compressors are used to increase the pressure of the air entering the engine. There are two main types of compressors; axial and centrifugal. Axial compressors are most commonly used in aircraft engines such as turbojets, turbofans, and turboshafts. They use a multitude of stages each equipped with numerous airfoil shaped blades, or vanes, to compress the flow parallel to the axis of rotation. Figure 1 shows an illustration of a multi-stage axial compressor.

![Illustration of Multi-Stage Axial Compressor](image)

Axial compressors utilize both rotors and stators. Rotors, as the name suggests, rotate, while stators remain stationary. The rotors transfer mechanical work to the fluid compressing the air while simultaneously decelerating the flow. Stators however are used to straighten the flow back out so that it is parallel to the axis of rotation. Each set of rotors has its own performance map. A “stage” in the compressor features one set of rotor blades and one set of stator blades and typically increases the pressure by 15% to 60%. In order to achieve the pressures required by the turbojet, the compressor may contain one to more than 15 stages. Each stage of the compressor is designed differently to decelerate and compress the flow as efficiently as possible. For rotor stages that require different rotational velocities, multiple “spools” are often used. A spool consists of a central shaft with attached compressor and turbine rotors. By using multiple spools, each compressor stage is allowed to rotate at its most optimal speed to maximize efficiency. A typical turbojet may contain one, two, or three spools. A CFM56 turbofan contains two spools; low and high pressure. A Rolls-Royce Trent 1000 turbofan however utilizes three spools; low, intermediate, and high pressure.

Centrifugal compressors compress the flow perpendicular to the axis of rotation. Figure 2 shows a typical single stage centrifugal compressor wheel.
Unlike axial compressors, centrifugal compressors are usually only single stage and thus require only one spool. Thus, centrifugal compressors are generally easier and less expensive to manufacture. They also tend to be more reliable and compact as they contain far less moving parts. Unfortunately because centrifugal compressors are only single stage they are unable to achieve high compression ratios much above 10:1. They are most often used in small auxiliary power units (APU), air pumps, automotive turbo and superchargers, and turboshaft engines.

B. Compressor Maps

Figure 3 is an illustration of a typical compressor map from a NASA technical report on axial compressor design. The bottom axis of the plot shows the compressor corrected airflow, defined in this case as a percentage of the design corrected airflow. Oftentimes the corrected airflow will be in lbm/min or kg/s. The left axis shows the compressor pressure ratio, which is the exit pressure divided by the inlet pressure. In this case, the pressure ratio is also defined as a percentage of the design pressure ratio.

---

Figure 2. Centrifugal Compressor Wheel.

Figure 3. Typical Compressor Map.
The lines running up and to the left are the speed lines which represent the rotational velocity of the compressor. Again, the speed is shown as a percentage of design rotational speed, ranging from 70% to 110% (over speed). Typically, the lowest and highest speed lines represent the minimum and maximum operational speeds of the compressor, respectively. The rotational velocity can also be shown in RPM (revolutions per minute).

The elliptical dashed lines represent the adiabatic efficiency “islands” of the compressor. In this case, the efficiency ranges from 70% to 88%. The dashed center line running through the middle of these islands in Figure 3 is the operating, or working line of the compressor; the line that the compressor performance follows given rotational speed and ambient conditions. Point ‘A’ shows the design point of Mach 0 at sea level at 100% throttle. Points ‘B’ and ‘C’ represent two off-design points of Mach 0.9 at altitude and Mach 2.8 at altitude, respectively. It can be seen that this compressor stage operates most efficiently at the on design condition.

One final feature of a compressor map is the stall, or surge line. In Figure 3 this is shown by the solid line at the far left of the data. The surge line is determined when the compressor experiences enough back pressure to stall the airfoil shaped rotor vanes, similar to an aircraft stalling its wing at high angles of attack. If the compressor stalls it may cause a sudden backflow of air through the unit. This could lead to a flameout condition, excess component stress, and even sudden failure of the unit. It is best to avoid operating the compressor to the left of the surge line. Some compressor maps also show surge margins, which are lines to show how close the compressor is to surge. These lines can be defined many ways, but typically are a percentage of the surge air mass flow rate at a given rotational speed.

For multistage compressors, each rotor stage will have its own operating map. This is because each stage is sized differently to cope with the change in pressure, temperature, and velocity of the air as it progresses through the compressor.

C. Turbines

In order to rotate the compressor, turbomachinery utilizes turbines to extract energy from the airflow. Turbines are constructed in a similar fashion to compressors with one more stages consisting of rotors and stators with airfoil shaped blades. However, turbines operate essentially the exact opposite from compressors. When the air exits the combustor, it is at a high pressure and temperature, and thus very high in energy. The turbine vanes exploit this flow and extract energy, using it to spin a central shaft that is connected to the compressor.

Turbine stages may also be located on one or more spools to optimize their speed and efficiency. Generally, turbines are more efficient than compressors, and thus less turbine stages are required to provide the mechanical energy for the compressor. Turbines are almost always axial in design as it is the most efficient design.

D. Turbine Maps

Figure 4 illustrates a typical turbine performance map using the same axes as the compressor map; pressure ratio vs. corrected airflow. In this case, the corrected airflow is in kg/s.
Plotting the turbine performance using the same axes as the compressor map usually results in the lines getting bunched up and thus is difficult to read. Some turbine maps, instead of pressure ratio, use the change in enthalpy divided by temperature as this is a direct representation of the energy extracted. To separate the lines, the corrected airflow may be multiplied by the corrected speed. Figure 5 represents the same turbine map as Figure 4 but with the bottom axis modified to separate the lines.
In Figure 5 the black vertical lines with slight curves at the bottom are speed lines. Each is labeled with the rotational speed as fraction of the maximum permissible speed ranging from 0.8 to 1.1 (over speed). The red dashed lines represent the isentropic efficiency hills of the turbine ranging from 75% to 85%. Finally, the yellow dots show the working line of this turbine as rotational velocity is increased. Unlike compressor maps, turbine maps do not show a surge line. This is due to the inherent design of turbines being devices that flow from high to low pressure. Due to this favorable pressure gradient, there is no back pressure on the turbine to cause stalling of the blades. Similar to the compressor, a turbine performance map will exist for each rotor stage.

II. Apparatus

A. Boeing T50-BO-8A

The experiment for this project was conducted in the Aerospace Engineering Propulsion Lab, building 41 room 144. The engine used was a Boeing T-50-BO-8A turboshaft modified to a turbojet configuration. This engine was originally produced in the early 1960’s for use on a Gyrodyne QH-50 DASH (Drone Anti-Submarine Helicopter) as shown in Figure 6.

![Figure 6. Gyrodyne QH-50 DASH](image)

On August 12, 1960, the DASH was the first unmanned helicopter to take flight. Equipped with torpedoes, the DASH could seek out and destroy enemy submarines giving the Navy a distinct advantage. Many DASH models were equipped with the Boeing T-50-BO-8A before switching to a more powerful -12A model. The -8A model was capable of 300 Hp, giving the DASH a cruise speed of 50 knots and a range of 71 nm. A cutaway version of a Boeing T-50-BO-8A turboshaft is shown in Figure 7.
The Boeing T-50-BO-8A turboshaft engine utilizes a single-spool design with a single stage centrifugal compressor. The pressurized air then enters one of two combustor cans where fuel is added and then ignited. The energetic exhaust then passes through a single stage axial turbine that powers the compressor. Finally, the exhaust passes through a second axial turbine which provides power to an output shaft through a gearbox. This second stage is commonly referred to as a power turbine.

Through multiple senior projects, the turbojet was modified with a fiberglass bell-shaped inlet and equipped with all of the necessary instrumentation. This instrumentation includes pressure transducers, thermocouples, a load cell, and a fuel flow meter. The location of these sensors is shown in Figure 8; an illustration of the apparatus for the Aero 401 jet engine experiment. All of these sensors are hooked to a desktop computer inside the control room of the propulsion lab to record vital data.
A hydraulically actuated variable area nozzle (VAN) was installed on the engine in place of the power turbine, gearbox, and output shaft. This converted the turboshift to a turbojet and allowed the exit area of the nozzle to be varied during operation. The experiment setup for this project is essentially identical to the original Aero 401 jet engine experiment, however the VAN was replaced with a orifice plate nozzle.

**B. Orifice Plate Nozzle**

In order to generate the performance maps for the compressor and turbine, it was necessary to vary the airflow rate and the engine rotational speed. Varying the rotational speed is simple using the throttle lever inside the control room. To vary the airflow rate, it was necessary to restrict the flow at the nozzle end of the turbojet. The restriction must be at the nozzle end to provide backpressure to the compressor in order to find its surge line. Unfortunately, the VAN only had an exit area ranging from 39.5 in$^2$ to 17.8 in$^2$. From the results from the Aero 401 jet engine lab, this change in area was not enough to drastically vary the airflow rate through the turbojet.

Several methods were looked into for solving this problem. First, blockage plates were considered to further reduce the exit area of the VAN. However, these plates would require extensive machining, would heavily distort the flow field, and would require an awkward clamping system to hold the plates in place. The second option was to modify the VAN to close to a smaller area. Unfortunately this would also require extensive machining. Also, the hydraulic actuators were too long to close the nozzle far enough.

The final option was to design a new nozzle that could vary the exit area with little machining and complexity. These requirements led to a simple conical nozzle design with orifice plates of various diameters. This nozzle design is shown in Figure 9.
C. Material selection

The nozzle material was chosen based on machinability, weldability, and melting temperature. The VAN that was removed from the turbojet was fabricated from stainless steel and was extremely heavy as a result. However, it was able to withstand the high exhaust gas temperatures from the turbojet. For the conical nozzle, many aluminum alloys were researched as shown in Table 1.

<table>
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<tr>
<th>Aluminum Alloy</th>
<th>Solidus Temperature (°F)</th>
<th>Liquidus Temperature (°F)</th>
<th>Machinability</th>
<th>Weldability</th>
<th>Price per sq. ft. for 1/8” thick</th>
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<tr>
<td>2024-T3</td>
<td>935</td>
<td>1180</td>
<td>Good</td>
<td>Low</td>
<td>$ 27.88</td>
</tr>
<tr>
<td>5052-H32</td>
<td>1125</td>
<td>1200</td>
<td>Good</td>
<td>Good</td>
<td>$ 11.85</td>
</tr>
<tr>
<td>6061-T6</td>
<td>1080</td>
<td>1205</td>
<td>Good</td>
<td>High</td>
<td>$ 13.86</td>
</tr>
<tr>
<td>7075-T6</td>
<td>890</td>
<td>1175</td>
<td>Low</td>
<td>Low</td>
<td>$ 26.36</td>
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The temperatures and workabilities were found on MatWeb while the prices for a square foot by 1/8 in. thick plate were found on OnlineMetals. The solidus temperature is when the alloy begins to melt, or begins to show properties of a liquid. The liquidus temperature occurs when the alloy becomes homogeneous and is mostly a liquid. For this reason, the solidus temperature was considered to be the maximum temperature to prevent the nozzle from warping or otherwise breaking. From the data collected during the Aero 401 jet engine lab, the maximum nozzle inlet temperature achieved was about 810°F. Alloy 5052-H32 was chosen for the conical nozzle and orifice plates due to its highest solidus temperature and its lowest price. The turbine housing flange from the VAN, made of stainless steel, was used to attach the nozzle to the turbine housing of the turbojet.

D. Nozzle Design

The conical nozzle portion was made from a 0.09 in. thick sheet mainly due to the 0.10 in. limit on the sheet metal roller and cutter in the Cal Poly hanger, building 4. The nozzle was 34 in. long to prevent the recirculation zones at the orifice plate from disturbing the flow through the turbine. The nozzle inlet was 9.125 in. in diameter while the nozzle exit was 7.0 in. in diameter, resulting in a 1.7° converging half-angle. Using inlet boundary conditions derived from the Aero 401 jet engine lab, the nozzle was modeled in Gambit and Fluent. The following figures show contour plots of velocity magnitude inside the nozzle during steady operation. Two different cases were then considered; steady idle at about 60% RPM and steady maximum thrust at 100% RPM. A 3.0 in. diameter orifice plates was used in these examples in order to model the most restrictive setup.
Figure 10. Nozzle Velocity Magnitude (in m/s) for 60% Throttle, $T_{05} = 650^\circ F$, $V_5 = 300$ ft/s.

Figure 11. Magnified View of Orifice Plate from Figure 10.
Both cases were conducted using a 2nd order k-epsilon viscous model with turbulent flow. The grids for each case were also adapted to refine the mesh in necessary areas. The recirculation zone can clearly be seen in the corners where the orifice plate meets the nozzle. Here the velocity of the air is nearly zero. It can also be seen that these zones are far enough away from the turbine to mitigate flow distortion or backflow into the turbine.

An interesting result from these cases is the area of very high speed flow at the leading corner of the orifice plate. For the 60% and 100% throttle cases the velocity of this flow is about 400 m/s (1312 ft/s or Mach 0.80 at T = 650°F) and 710 m/s (2329 ft/s or Mach 1.33 at T = 810°F). The supersonic flow for the 100% throttle case was of some concern, however this area was so small it was unlikely to adversely impact the results. Conversely, this supersonic flow may have acoustic implications. These Fluent cases finally show that the flow at the exit plane or the orifice plate is subsonic even at 100% throttle.
E. Nozzle Flange Design

The flanges for ends of the nozzle were machined from 3/16 in. thick aluminum plate due to the limited availability of thicker material. However, this thickness was felt to be sufficient as the flanges would not be under a significant amount of stress. Figure 14 shows the nozzle flanges that were machined.

![Figure 14. Nozzle Flanges.](image)

The octagonal flange on the right was designed to mate with the stainless steel turbine housing flange from the VAN. It is octagonal in shape to mitigate clearance issues with the turbojet. The circular flange was designed for the exit end of the nozzle and features four bolt holes to attach the various orifice plates. Figure 15 and Figure 16 show the turbojet with the finished nozzle attached.

![Figure 15. Boeing T-50-BO-8A with Nozzle Attached.](image)
F. Orifice Plate Design

The orifice plates were machined from ¼ in. thick aluminum plate. This thickness was chosen so a 0.10 in. chamfer could be machined on each side of the orifice for the smoothest flow possible. Before machining began however, it was realized that having chamfered edges on both sides would nearly double the machining time. Thus, the chamfer was removed from one edge and lengthened to 0.15” on the other. The chamfered edge would face the turbine during operating. Also, since these plates would also be attached to the nozzle with four bolts, a thicker material would mitigate warping and leakage. Eight differently sized orifice plates are shown in Figure 17.

The inside diameters of the orifice plates range from 6.5 in. to 3.0 in. These values were chosen based on the area of the VAN which ranged from 39.5 in² fully open to 17.8 in² fully closed. The VAN’s fully open and closed positions corresponded to a circular diameter of 7.1 in. and 4.8 in., respectively. As mentioned previously, it was necessary to restrict the airflow rate further, so some smaller orifice diameters were chosen. An orifice diameter of 7.0 in. would be achieved by not using any orifice plate.

Before machining the plates however, it was decided not to machine the 3.0 in. orifice plate. This was due to the belief that the turbojet would overheat or not even run with this much of a restriction. Furthermore, the ¼ in. plate
that would have been used to machine this orifice plate could be machined to a different size after initial testing to increase the fidelity of the data in interesting areas.

The orifice plates were machined in the Cal Poly Mechanical Engineering department’s newest machine shop, “Mustang 60”. Figure 18 shows the seven machined orifice plates while Figure 19 shows the chamfer machined into the inside edge of the orifice plates.

Figure 18. Finished Orifice Plates in Descending Inside Diameter from Left to Right.

Figure 19. Close-up of Chamfer Machined on Inside Edge.

G. Nozzle Data Acquisition

To produce the turbine performance map, it was necessary to place instrumentation in the nozzle to measure the turbine exit temperature and pressure. The VAN was already equipped with thermocouples to measure the turbine exit temperature and a pitot probe to measure nozzle exit pressure. To utilize the two thermocouples, two 1/8 National Pipe Thread (NPT) threaded aluminum bungs were welded to the side of the nozzle. These were placed about 2 in. from the exit plane of the turbine. Unfortunately, the threads of the aluminum bungs were destroyed after only one use. This was due mainly to the design of NPT threads, which are tapered to purposely mesh the threads together for a tight seal. The much harder stainless steel threads of the thermocouple appeared to rip out the soft aluminum threads of the bung. Figure 20a shows the thermocouple mounted in the nozzle.

The pitot probe was constructed to fit the much shorter VAN. For this reason, its mounting position on the test stand was modified so the probe could enter through the orifice plate at the rear of the nozzle. Unfortunately, this meant the pitot probe was about 24 in. behind the exit plane of the turbine as shown in Figure 20b. However, with the isentropic assumptions (explained later), it was safe to assume a negligible change in stagnation pressure between the turbine exit and the pitot probe. Figure 20b shows the nozzle with the pitot probe installed.
III. Procedure

The procedures of this experiment followed closely those of the jet engine lab in Aero 401. These procedures include setting up LabView to collect data, pre-test checks, safety checks, operational procedures, and post-test checks. The following sections provide detailed instructions for conducting the experiment.

A. LabView and Remote Camera

1) Turn on LabView computer(gray) located on the table in the control room. Login: aero, Password: aero
2) Turn on camera computer(white) located under the table in the control room. Login: Aero, Password: none
3) On the LabView computer open the turbojet virtual instruments (VI) file for LabView.
4) In the block diagram window, change the filename and directory to correspond with the test being conducted.
5) Run the VI and make sure the data is reading correctly.
6) Record approximately two minutes of data to use as tare values for the calculations.
7) On the camera computer open the web cam program located on the desktop and make sure the camera is working correctly. Do not capture video.
8) On the turbojet test stand, point camera to view area of interest. For some of the tests the camera was pointed at the nozzle exit pitot tube. In other tests it was pointed directly at the nozzle.

B. Pre-Test Checks

1) Record ambient temperature and pressure.
2) Open both lab roll-up doors to the test cell.
3) Roll the blue acoustic absorbing blocks out of the test cell into the area behind the turbojet. Make sure they are not placed immediately behind the turbojet nozzle.
4) Add fuel to the gas tank outside of the lab. Use Jet-A fuel only. The tank holds 13 gallons and should contain about 1 gallon per 2 minutes of expected engine run time to ensure the turbojet does not run out of fuel.
5) Leave the gas cap loose to allow air to enter the tank during testing.
6) Open the two fuel ball valves, one located under the fuel tank and one on the turbojet test stand.
7) Plug the battery cart charger into the wall and the large red plug into the test stand. The red plug should have a ‘+’ and a ‘-’ facing up when plugged in.
8) Install the desired orifice plate onto nozzle with the four bolts.
9) Move the nozzle exit pitot tube into position and tighten the two bolts. It should be placed as far into the nozzle as possible without touching the orifice plate.
10) Check the turbojet oil and add if necessary.

C. Safety Checks

1) Check to make sure no plastic tubing is touching the surfaces of the turbojet
2) Check for and remove loose or flammable debris in the test cell area.
3) Make sure the fire extinguisher is accessible and charged.
4) Place a bucket under the oil drain tube.
5) Make sure all personnel are out of the test cell.
6) Check the EGT chart for the maximum allowable $T_{05}$ given the ambient temperature.

D. Turbojet Pre-Operation
1) Switch all three display panels to the “ON” position.
2) Check that all temperature displays are showing degrees Fahrenheit.
3) Make sure all sensors are reading properly and are close to the values of LabView.
4) Switch on the “Engine Master” and “Oil Fan” switches.
5) Make sure the voltmeter reads at least 24V. If not, do not attempt to start turbojet.
6) Toggle the “Fuel Pump” switch to the “OFF” position.
7) Toggle the “Ignition” switch “ON” and make sure the ignition system is audible.
8) Switch the “Fuel Boost Pump” switch “ON” and make sure there is at least 35 psi of fuel pressure, and then switch it “OFF”.
9) Toggle the “Starter” on for about 5 seconds and make sure there is oil pressure.

E. Turbojet Operation
1) Begin recording data in LabView. Set RPM to zero.
2) Operate throttle lever to the “MIN” position.
3) Toggle the “Fuel Pump” switch to the “OFF” position.
4) Toggle the “Fuel Boost Pump” switch to the “ON” position.
5) Toggle and hold the “Ignition” switch “ON”.
6) After 10 seconds, toggle and hold the “Starter” switch “ON”.
7) Once the engine RPM reaches 15%, toggle the “Fuel Pump” switch “ON”.
8) If turbojet is cold, $T_{04}$ and $T_{05}$ will exceed peak values during spool-up. Shut down the engine only if these temperatures are exceeded for more than 5 seconds.
9) Once the engine RPM reaches 60%, toggle the “Fuel Boost Pump”, “Starter”, and “Ignition” switches “OFF”.
10) Allow the turbojet to operate at idle speed for 1 minute for warm up.
11) Operate the throttle lever to desired engine RPM.
12) Monitor temperature transients in LabView. Once they become level, enter the corresponding engine RPM in LabView and press “Enter”.
13) After about 10 seconds, set the RPM in LabView back to zero. The zero RPM data will separate the transient from the steady-state data in the data file.
14) Operate the throttle level to the next RPM and repeat. Do not exceed a $T_{04}$ of 1400°F or $T_{05}$ of 1125°F (the solidus temperature of the nozzle and orifice plates).

F. Turbojet Shutdown
1) Once all desired data has been recorded, stop recording data in LabView.
2) Operate the throttle lever to the “MIN” position.
3) Wait approximately 10 seconds and then toggle the “Fuel Pump” switch to the “OFF” position.
4) Once the engine RPM is below 10%, cycle the “Starter” switch “ON” until $T_{04}$ stays below 450°F. Do not exceed 15% RPM when cycling.
5) Toggle the “Engine Master” switch to the “OFF” position.

These procedures were repeated for each orifice plate. The down-time between tests required to change the orifice plate was minimized to reduce the amount of time needed for the turbojet to warm up. This resulted in less fuel being consumed for each test. Two individuals with heat resistant gloves could reduce the down-time to just a couple minutes which included changing the orifice plate, adding fuel, and resetting LabView.
IV. Data Analysis

Once the data has been collected in LabView it was placed into an extensive Microsoft Excel program for analysis. The turbojet parameters that were measured and recorded by LabView are shown in Table 2.

Table 2. Turbojet Instrumentation.

<table>
<thead>
<tr>
<th>Parameter Name</th>
<th>Nomenclature</th>
<th>Sensor Location</th>
</tr>
</thead>
<tbody>
<tr>
<td>Inlet Static Temperature</td>
<td>$T_i$</td>
<td>Inlet</td>
</tr>
<tr>
<td>Inlet Static Pressure</td>
<td>$P_i$</td>
<td>Inlet</td>
</tr>
<tr>
<td>Inlet Stagnation Pressure</td>
<td>$P_{0i}$</td>
<td>Inlet</td>
</tr>
<tr>
<td>Compressor Exit Stagnation Pressure</td>
<td>$P_{03}$</td>
<td>Compressor Housing</td>
</tr>
<tr>
<td>Compressor Exit Stagnation Temperature</td>
<td>$T_{03}$</td>
<td>Compressor Housing</td>
</tr>
<tr>
<td>Combustor Exit Stagnation Temperature</td>
<td>$T_{04}$</td>
<td>Turbine Housing</td>
</tr>
<tr>
<td>Turbine Exit Stagnation Temperature</td>
<td>$T_{05}$</td>
<td>Nozzle Entrance</td>
</tr>
<tr>
<td>Turbine Exit Stagnation Pressure</td>
<td>$P_{05}$</td>
<td>Nozzle Exit</td>
</tr>
<tr>
<td>Thrust</td>
<td>$T$</td>
<td>Thrust Stand</td>
</tr>
<tr>
<td>Engine RPM</td>
<td>$n$</td>
<td>Engine Housing</td>
</tr>
</tbody>
</table>

For this project, all of these parameters with the exception of thrust were necessary for creating the compressor and turbine maps. As will be explained later however, the measured thrust was used to calculate the turbine exit pressure. The fuel flow was recorded incorrectly by LabView, so instead it was estimated using thermodynamic principles and fuel properties. To create the performance maps, all that is needed is the pressure ratios of the compressor and turbine and the corrected flow rate. The following assumptions and steps were used to calculate these parameters.

A. Assumptions

The assumptions used in this project are summarized here:
1) Isentropic flow
2) One dimensional flow
3) Steady, irrotational flow
4) Calorically perfect gas
5) Constant pressure combustion
6) Zero pressure loss in the inlet and nozzle

Many these assumptions were made in order to make the calculations simpler and to reduce the amount of required data. The first major assumption of isentropic flow designates the flow to be adiabatic meaning no transfer or heat, and irreversible meaning there is no friction and thus no change in entropy. This assumption yields the following relations.

$$\frac{P_i}{P} = \left[1 + \frac{\gamma-1}{2} M^2\right]^\frac{\gamma}{\gamma-1}$$

$$\frac{T_i}{T} = 1 + \frac{\gamma-1}{2} M^2$$

$$\frac{P_0}{P} = \left[\frac{T_0}{T}\right]^\frac{\gamma}{\gamma-1}$$

where $P$ and $P_0$ are static and stagnation pressure, respectively, $T$ and $T_0$ are static and stagnation temperature, respectively, $M$ is the Mach number, and $\gamma$ is the ratio of specific heats. The isentropic relations offer a comparison between static and stagnation conditions based only on the Mach number and the ratio of specific heats. The calorically perfect gas assumption results in the following relationship.

California Polytechnic State University. Aerospace Engineering
April 16, 2010
\[ \gamma = \frac{C_p}{C_v} \]  

(4)

where \( C_p \) is the specific heat at constant pressure and \( C_v \) is the specific heat at constant volume. The constant pressure combustion, zero pressure loss in the inlet and nozzle, and isentropic flow yield the following relations.

\[ P_{i3} = P_{i4} \]  

(5)

\[ P_{i4} = P_{i2} \]  

(6)

\[ P_{i5} = P_{i7} \]  

(7)

\[ T_{i2} = T_{i1} \]  

(8)

where subscripts \( i, 2, 3, 4, 5, \) and \( 7 \) denote the turbojet’s inlet, compressor inlet, compressor exit, combustor exit, turbine exit, and nozzle exit stations, respectively.

**B. Corrected Flow Rate**

The corrected flow rate for the turbojet is the amount of air that passes through the unit per instance of time. In this project, it was defined as pounds per second (lbm/s), but is also commonly defined as slugs per second (slugs/s), pounds per hour (lb/hr), or kilograms per second (kg/s). To determine the corrected flow rate, first the correction factors were defined for temperature and pressure as shown in Equations 9 and 10, respectively.

\[ \theta = \frac{T_a}{518 \, ^\circ R} \]  

(9)

\[ \delta = \frac{P_a}{14.689 \, psi} \]  

(10)

where \( T_a \) and \( P_a \) are the ambient temperature and pressure determined before testing. These factors were necessary for referencing other parameters to standard temperature and pressure (STP) conditions. Next, the velocity in the inlet of the turbojet was calculated Bernoulli’s equation as shown in Equation 11, and the rearranged version in Equation 12.

\[ P_i = P + \frac{1}{2} \rho V_i^2 \]  

(11)

\[ V_i = \sqrt{\frac{2}{\rho} (P_i - P)} \]  

(12)

where \( P_{i0} \) and \( P_i \) are the inlet stagnation and static pressure, respectively, and \( \rho \) is the density of the air entering the inlet. The inlet velocity was then entered into the mass flow rate equation represented in Equation 13 to determine the mass flow rate of air entering the turbojet.

\[ \dot{m}_{\text{air}} = \rho AV_i \]  

(13)

where \( A \) is the cross-sectional area of the inlet. The mass flow rate was then easily converted to a flow rate using Equation 14.

\[ \dot{w}_{\text{air}} = g \dot{m}_{\text{air}} \]  

(14)

where \( g \) is the acceleration of gravity, assumed to be 32.174 ft/s² in this project. To achieve the corrected airflow rate, the temperature and pressure corrections can be applied in Equation 15.
This represents the amount of air entering the turbojet. In the combustor however, fuel is added to the air and thus the flow rate exiting the turbojet is slightly larger. Even though the ratio of fuel to air is very small (~0.01), it was still considered to increase accuracy. The determination of fuel flow rate is described later.

C. Pressure Ratios
The pressure ratios for the both the compressor and the turbine were determined through simple calculations as shown in Equations 16 and 17.

\[ \pi_{\text{comp}} = \frac{P_0}{P_0} \]  \hspace{1cm} (16)

\[ \pi_{\text{turb}} = \frac{P_{\text{turb}}}{P_{\text{turb}}} \]  \hspace{1cm} (17)

Both of these pressure ratios rely on the assumptions of zero pressure loss in the inlet, combustor, and nozzle. Without these assumptions, much more instrumentation would be required on the turbojet.

D. Efficiencies
The efficiencies for the compressor and turbine compare the measured enthalpy change to the ideal enthalpy change. Equations 18 and 19 represent the compressor and turbine efficiencies, respectively.

\[ \eta_{\text{comp}} = \frac{\Delta H_{\text{comp}}}{\Delta H_{\text{comp ideal}}} \]  \hspace{1cm} (18)

\[ \eta_{\text{turb}} = \frac{\Delta H_{\text{turb}}}{\Delta H_{\text{turb ideal}}} \]  \hspace{1cm} (19)

where \( H \) is enthalpy, defined in Equation 20.

\[ H = C_p T \]  \hspace{1cm} (20)

E. Fuel Flow Rate
The fuel flow rate was important for determining the amount of mass passing through the turbine. Since the fuel flow meter was not recording properly into LabView, an alternative method for determining the fuel flow rate had to be pursued. Enthalpy balance across the combustor and fuel properties were used in this case. The enthalpy balance equation written in terms of temperature is shown here in Equation 21.

\[ f = \frac{\frac{T_{04}}{T_{03}} - 1}{\frac{Q_f}{C_p} \frac{T_{03}}{T_{03}}} \]  \hspace{1cm} (21)

where \( f \) is the fuel/air ratio and \( Q_f \) is the fuel heating value of the fuel, assumed to be 18,400 Btu/lbm for Jet-A. \( C_p \), being a function of temperature and air properties, changes between the inlet and exit of the combustor. For simplification, the average combustor temperature and properties of standard air were used. Next the fuel flow rate was calculated using Equation 22 and corrected in Equation 23.

\[ \dot{w}_{\text{fuel}} = \dot{w}_{\text{air}} f \]  \hspace{1cm} (22)
\[ \dot{w}_{\text{fuel,corr}} = \frac{\dot{w}_{\text{fuel}}}{\delta \theta} \]  \hspace{1cm} (23)

Finally, the total flow rate exiting the turbojet through the turbine is simply the summation of the air and fuel flow rates as shown in Equation 24.

\[ \dot{w}_{\text{exh,corr}} = \dot{w}_{\text{fuel,corr}} + \dot{w}_{\text{air,corr}} \]  \hspace{1cm} (24)

**F. Work Balance**

A fundamental property of turbomachinery is that the work done by the compressor is equal to the work done by the turbine for a steady state operating condition. This relation can be useful for determining errors in the data or calculations. The compressor and turbine work relations are shown here in Equations 25-27.

\[ \text{Work}_{\text{comp}} = \Delta H_{\text{comp}} \dot{w}_{\text{air,corr}} \]  \hspace{1cm} (25)

\[ \text{Work}_{\text{turb}} = \Delta H_{\text{turb}} \dot{w}_{\text{exh,corr}} \]  \hspace{1cm} (26)

\[ \text{Work}_{\text{comp}} = \text{Work}_{\text{turb}} \]  \hspace{1cm} (27)

For simplification purposes, many of these equations were incorporated into an Excel Visual Basic macro named *thprop*, written by Mark Waters (*thprop* is copied into the Appendix B for reference). This macro uses simple thermodynamic relations to calculate parameters such as enthalpy, temperature, pressure, fuel/air ratio, specific heats, and more given a wide combination of inputs.

**V. Data Validity**

After the data was collected and averaged, it was important to analyze each measured parameter to determine the validity of the data. Small errors in the collected data can severely affect the results of the experiment. LabView recorded the turbojet data into an Excel file and included the parameters from Table 2 as well as time in seconds and engine speed in RPM. A sample raw data set is shown here in Table 3.

<table>
<thead>
<tr>
<th>Time (s)</th>
<th>Thrust (lbf)</th>
<th>T1 (°F)</th>
<th>T03 (°F)</th>
<th>T04 (°F)</th>
<th>T05 (°F)</th>
<th>Pi (psig)</th>
<th>P0i (psig)</th>
<th>P03 (psig)</th>
<th>P07 (psig)</th>
<th>Speed (%)</th>
<th>Speed (RPM)</th>
</tr>
</thead>
<tbody>
<tr>
<td>96.789</td>
<td>22.667</td>
<td>74.15</td>
<td>270.77</td>
<td>932.895</td>
<td>687.195</td>
<td>0.38</td>
<td>0.121</td>
<td>22.165</td>
<td>0.55</td>
<td>75</td>
<td>28500</td>
</tr>
<tr>
<td>96.992</td>
<td>18.018</td>
<td>74.324</td>
<td>270.77</td>
<td>931.55</td>
<td>687.624</td>
<td>0.349</td>
<td>0.082</td>
<td>22.009</td>
<td>0.523</td>
<td>75</td>
<td>28500</td>
</tr>
<tr>
<td>97.195</td>
<td>17.992</td>
<td>74.107</td>
<td>270.974</td>
<td>931.277</td>
<td>688.587</td>
<td>0.414</td>
<td>0.105</td>
<td>22.229</td>
<td>0.606</td>
<td>75</td>
<td>28500</td>
</tr>
<tr>
<td>97.398</td>
<td>18.333</td>
<td>74.468</td>
<td>270.925</td>
<td>931.004</td>
<td>689.004</td>
<td>0.301</td>
<td>0.068</td>
<td>22.198</td>
<td>0.569</td>
<td>75</td>
<td>28500</td>
</tr>
<tr>
<td>97.586</td>
<td>19.852</td>
<td>74.418</td>
<td>270.892</td>
<td>929.753</td>
<td>689.004</td>
<td>0.365</td>
<td>0.103</td>
<td>22.079</td>
<td>0.545</td>
<td>75</td>
<td>28500</td>
</tr>
</tbody>
</table>

For each orifice plate diameter and engine speed, the average of the data was found and placed into a calculation spreadsheet. The average of the recorded pressure tare values was then subtracted from this data to correct the values. Table 4 shows an example of the averaged data.
## Table 4. Sample Averaged Data: 6.0 in. Orifice Plate.

<table>
<thead>
<tr>
<th>Plate Size in</th>
<th>Throttle % RPM</th>
<th>Thrust lbf</th>
<th>$T_i$ °F</th>
<th>$T_{03}$ °F</th>
<th>$T_{04}$ °F</th>
<th>$T_{05}$ °F</th>
<th>$P_i$ psig</th>
<th>$P_{0i}$ psig</th>
<th>$P_{03}$ psig</th>
<th>$P_{07}$ psig</th>
</tr>
</thead>
<tbody>
<tr>
<td>6.0</td>
<td>65% Below minimum idle</td>
<td>70%</td>
<td>29.604</td>
<td>63.582</td>
<td>240.556</td>
<td>940.603</td>
<td>718.829</td>
<td>-0.188</td>
<td>-0.021</td>
<td>19.548</td>
</tr>
<tr>
<td></td>
<td>75%</td>
<td>35.061</td>
<td>62.713</td>
<td>260.526</td>
<td>940.924</td>
<td>721.269</td>
<td>-0.212</td>
<td>-0.015</td>
<td>22.467</td>
<td>1.102</td>
</tr>
<tr>
<td></td>
<td>80%</td>
<td>43.295</td>
<td>63.461</td>
<td>290.627</td>
<td>951.843</td>
<td>728.256</td>
<td>-0.214</td>
<td>-0.024</td>
<td>26.940</td>
<td>1.331</td>
</tr>
<tr>
<td></td>
<td>85%</td>
<td>52.488</td>
<td>61.392</td>
<td>317.993</td>
<td>976.516</td>
<td>717.772</td>
<td>-0.247</td>
<td>-0.035</td>
<td>31.642</td>
<td>1.576</td>
</tr>
<tr>
<td></td>
<td>90%</td>
<td>63.205</td>
<td>60.690</td>
<td>350.890</td>
<td>1042.693</td>
<td>740.910</td>
<td>-0.291</td>
<td>-0.045</td>
<td>37.116</td>
<td>1.700</td>
</tr>
<tr>
<td></td>
<td>95%</td>
<td>75.917</td>
<td>59.987</td>
<td>382.372</td>
<td>1121.300</td>
<td>774.058</td>
<td>-0.345</td>
<td>-0.059</td>
<td>43.312</td>
<td>1.593</td>
</tr>
<tr>
<td></td>
<td>100%</td>
<td>86.155</td>
<td>61.622</td>
<td>411.472</td>
<td>1200.825</td>
<td>824.653</td>
<td>-0.382</td>
<td>-0.064</td>
<td>47.870</td>
<td>1.725</td>
</tr>
</tbody>
</table>

### A. Turbojet Operation Envelope

With the 6.0 in. orifice plate from the example in Table 4, the turbojet was not able to operate at 65% RPM because the minimum idle was higher than 65%. Low engine speeds were restricted by the turbojet’s minimum idle, while high engine speeds were limited by $T_{04}$ or $T_{05}$ temperature limits. These limits are shown in Figure 21 were derived from the experimental data. All of the test points shown in this figure were investigated in this project.

![Figure 21. Theoretical Turbojet Operation Envelope.](image)

The smaller orifice diameters reduced the minimum idle by restricting the amount of air that was allowed to pass through the turbojet. However, the smaller orifice diameters also caused to the turbojet to operate at a higher temperature. The limited temperatures occurred at the turbine inlet and the turbine exit, and both temperature limits were reached at approximately the same time. The idle and temperature limit lines cross at an orifice plate diameter of about 3.5 in. where theoretically the turbojet would idle at or above the temperature limits. These limitations severely reduced the operating range of the turbojet and the range of compressor and turbine maps.

### B. Data Validity

As mentioned previously, it was necessary to determine the validity of the collected data. To do this, each measured parameter was plotted against engine speed for the various orifice plates. This would reveal any abnormalities in the data as one would expect each parameter to follow a certain trend. Figure 22 through Figure 25 show the data trends of thrust and engine temperatures.
Figure 22. Thrust vs. Engine Speed.

Figure 23. Compressor Exit Stagnation Temperature vs. Engine Speed.

Figure 24. Combustor Exit Stagnation Temperature vs. Engine Speed.
The trend of thrust vs. engine speed is very consistent and behaves as expected. Smaller orifice plates and larger engine speeds both increase the nozzle exit velocity, which directly related to gross thrust. The compressor exit temperature, $T_{03}$, also behaves very well with a clear linear upward trend with higher engine speeds. However, the combustor and turbine exit temperatures, $T_{04}$ and $T_{05}$, seem to increase very little at low engine speeds, but then increase almost exponentially afterwards. This may be due to the fueling trends of the engine. These two temperatures also experience some unexpected dips and oscillations.

The trends of these temperatures were considered in more depth by looking at the raw data. Figure 26 and Figure 27 show the variation of temperature during the experiment.
Each upward step in the temperatures represents an increase in engine speed. However, $T_{03}$ increases very gradually while $T_{04}$ and $T_{05}$ increase sharply with each new engine speed. If the temperature probe is placed directly in the flow path, it is expected that the temperature would show a sharp change similar to $T_{04}$ and $T_{05}$. The temperature probe for $T_{03}$ was placed on the very outside of the compressor casing. The ideal location is at the compressor discharge or the entrance to the combustor. For this reason it is expected that the $T_{03}$ probe is underestimating the actual compressor exit temperature. This will be shown in the next section when corrections to the data are made.

Figure 28 through Figure 31 show the trends of measured engine pressures.
Figure 29. Inlet Stagnation Pressure vs. Engine Speed.

Figure 30. Compressor Exit Stagnation Pressure vs. Engine Speed.

Figure 31. Turbine Exit Stagnation Pressure vs. Engine Speed.
The trends of the inlet static, inlet stagnation, and compressor exit pressures, $P_i$, $P_{0i}$, and $P_{03}$, seem reasonable. As engine speed increases, static pressure through the inlet will decrease as the inlet air velocity rises. The inlet stagnation pressure should remain relatively constant but does decrease slightly due to friction losses. The turbine exit pressure, $P_{05}$, does not behave as expected. While the pressure does increase with smaller orifice plates, it oscillates with engine speed for the larger orifice plates. For the 7.0 in. orifice plate, the gauge pressure drops below zero at 90% RPM. This is clearly an incorrect measurement. The reason for this is still unknown, but it likely due to unstable flow or random recirculation zones forming inside the nozzle.

In the next section, the issues with compressor exit temperature, $T_{03}$, and turbine exit pressure, $P_{05}$, will be resolved. $T_{03}$ will be calculated using a work balance method between the compressor and turbine and $P_{05}$ will be calculated using the measured thrust.

VI. Results and Discussion

With the data checked, the necessary calculations could then be performed. The calculations begin at the inlet of the turbojet, step through each component, and finish at the nozzle. Sample data for each component will be shown as well as relevant figures and data trends. Once each component of the turbojet is analyzed, the resulting compressor and turbine performance maps will be shown.

A. Inlet Data

Sample calculated data for the inlet is shown in Table 5. The calculations include everything from the density of the incoming air to the corrected flow rate and Mach number.

Table 5. Sample Calculated Data for Turbojet Inlet: 7.0 in. Orifice Plate.

<table>
<thead>
<tr>
<th>Test Case</th>
<th>Plate Size</th>
<th>Throttle % RPM</th>
<th>$\rho_i$</th>
<th>$V_i$</th>
<th>$m$</th>
<th>$\dot{m}_{corr}$</th>
<th>$\dot{m}_{corr}$</th>
<th>$M_i$</th>
<th>$T_{0i}$</th>
<th>Pressure Recovery</th>
</tr>
</thead>
<tbody>
<tr>
<td>7.0</td>
<td>65%</td>
<td>Below minimum idle</td>
<td>0.00227</td>
<td>0.0429</td>
<td>0.0439</td>
<td>1.41</td>
<td>0.101</td>
<td>535.46</td>
<td>99.84%</td>
<td></td>
</tr>
<tr>
<td>7.0</td>
<td>70%</td>
<td>Below minimum idle</td>
<td>0.00227</td>
<td>0.0520</td>
<td>0.0532</td>
<td>1.71</td>
<td>0.123</td>
<td>535.96</td>
<td>99.89%</td>
<td></td>
</tr>
<tr>
<td>7.0</td>
<td>75%</td>
<td>139.08</td>
<td>0.0581</td>
<td>0.0594</td>
<td>1.91</td>
<td>0.137</td>
<td>535.47</td>
<td>99.79%</td>
<td></td>
<td></td>
</tr>
<tr>
<td>7.0</td>
<td>80%</td>
<td>155.48</td>
<td>0.0585</td>
<td>0.0636</td>
<td>2.05</td>
<td>0.148</td>
<td>535.12</td>
<td>99.71%</td>
<td></td>
<td></td>
</tr>
<tr>
<td>7.0</td>
<td>85%</td>
<td>166.85</td>
<td>0.0622</td>
<td>0.0636</td>
<td>2.05</td>
<td>0.148</td>
<td>537.17</td>
<td>99.64%</td>
<td></td>
<td></td>
</tr>
<tr>
<td>7.0</td>
<td>90%</td>
<td>167.32</td>
<td>0.0621</td>
<td>0.0636</td>
<td>2.05</td>
<td>0.148</td>
<td>537.12</td>
<td>99.63%</td>
<td></td>
<td></td>
</tr>
<tr>
<td>7.0</td>
<td>95%</td>
<td>183.07</td>
<td>0.0678</td>
<td>0.0694</td>
<td>2.23</td>
<td>0.162</td>
<td>537.12</td>
<td>99.63%</td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

Calculating the inlet density and velocity was necessary for determining the flow rate through the inlet, which was used in many future calculations. For the entire experiment, the corrected flow rate ranged from 1.33 to 2.52 lb/s and is plotted in Figure 32.
The corrected flow rate behaves as expected, increasing with engine speed. However, the flow rate also increases as the orifice plates reduce in size. This trend is the opposite of what was predicted earlier. The reason for this is still not fully understood, but is likely due to the increased nozzle pressure ratio and increased nozzle exit velocity of the smaller orifice plates. Nevertheless, the goal of the orifice plates was to vary the flow rate through the engine, and from this figure they appear to have worked. Figure 33, from the Boeing T50-BO-8A model specification, shows the engine (in its turboshaft configuration) air flow as a function of ambient temperature and power rating at sea level static conditions.

![Figure 32. Corrected Flow Rate vs. Engine Speed.](image)

![Figure 33. Boeing T50-BO-8A Air Flow vs. Ambient Temperature at SLS Conditions.](image)
Given an ambient temperature of 74°F at normal rated power, the turbojet should consume about 4.1 lb/s of air according to the dashed red lines in Figure 33. However, the greatest air flow calculated for the turbojet was 2.52 lb/s, a 39% error. Although the turboshaft from Figure 33 is equipped with a power turbine and gearbox, it is not believed that this would cause such a significant increase in air flow. On the contrary, one would expect the air flow to decrease with the addition of a power turbine. This result aroused some suspicion about the operation of the turbojet.

The inlet Mach number was required to find $T_{0i}$, the inlet stagnation temperature, and ranged from 0.09 to 0.18. The inlet pressure recovery was also calculated and is plotted in Figure 34.

Plotting the pressure recovery of the inlet is a reliable check to see if the data is valid. From this figure, the pressure recovery ranges from 100% to about 99.6% and decreases linearly with higher engine speeds. This is expected because as engine speed is increased, the velocity through the inlet increases as well, which results in higher pressure losses due to friction. Overall, the inlet is a fairly simple component of the turbojet and was anticipated to perform as expected.

B. Calculation of Compressor Exit Temperature

Before the calculated data and relevant plots for the compressor can be presented, it was necessary to recalculate the compressor exit temperature as mentioned before. This was performed with a work balance method using the measured compressor and turbine data. Since many of the calculated parameters between the compressor, combustor, and turbine are intertwined, an iterative process was required to calculate the new compressor exit temperature. This process is illustrated in Figure 35 where calculations are performed from left to right.
Figure 35. Illustration of Iterative Calculation of Compressor Exit Temperature.

From this figure it can be seen that the compressor exit temperature directly affects the compressor work by altering compressor exit enthalpy, $H_3$. However, it also affects the fuel/air ratio, $f$. The fuel/air ratio alters the specific heat at constant pressure, $C_p$, of the combustor exit and turbine exit, which alters $H_4$ and $H_5$. The fuel/air ratio also changes the flow rate through the turbine.

Before calculating the compressor exit temperature, the error in the work balance ranged from -2% to 57%. After a single iteration the error was reduced to about -0.15%. With the new compressor exit temperature calculated, the compressor and combustor components can now be presented.

C. Compressor Data

Unlike the inlet, the compressor of the turbojet is a much more complicated component consisting of a rotating compressor wheel and complex passageways. Some sample calculations for the turbine using the calculated compressor exit temperature are shown here in Table 6.

<table>
<thead>
<tr>
<th>Test Case</th>
<th>Plate Size</th>
<th>Compressor Data</th>
</tr>
</thead>
<tbody>
<tr>
<td>Throttle</td>
<td>Plate %</td>
<td>$\pi_{comp}$</td>
</tr>
<tr>
<td>in</td>
<td>RPM</td>
<td>$^\circ F$</td>
</tr>
<tr>
<td>65%</td>
<td>Below min.</td>
<td>2.48</td>
</tr>
<tr>
<td>70%</td>
<td>Below min.</td>
<td>2.74</td>
</tr>
<tr>
<td>75%</td>
<td>Below min.</td>
<td>3.08</td>
</tr>
<tr>
<td>80%</td>
<td>Below min.</td>
<td>3.45</td>
</tr>
<tr>
<td>85%</td>
<td>Below min.</td>
<td>3.82</td>
</tr>
<tr>
<td>90%</td>
<td>Below min.</td>
<td>4.21</td>
</tr>
<tr>
<td>95%</td>
<td>Below min.</td>
<td>4.61</td>
</tr>
<tr>
<td>100%</td>
<td>Below min.</td>
<td>5.01</td>
</tr>
</tbody>
</table>

The compressor calculations included many essentails such as the pressure ratio, inlet and exit enthalpies, and work done by the compressor, and the compressor isentropic efficiency. The pressure ratio ranged from 2.0 to 4.3 and is plotted in Figure 36.
The trend of the compressor pressure ratio was just as expected. As the turbojet’s speed increases, the pressure ratio of the compressor increases linearly. The size of the orifice plates also appeared to have no effect on the pressure ratio. Next, the compressor efficiency is shown in Figure 37 and ranged from 37% to 80%.

Although less consistent than some of the other parameters, the compressor efficiency also behaved as expected. The efficiency increased with engine speed, hit a maximum, and then began to decrease. The maximum compressor efficiency occurred at around 85% RPM, but does vary with orifice plate size. However, there is no discernible correlation between compressor efficiency and orifice plate size.

D. Combustor Data
Analyzing the combustor was required in order to determine the fuel flow. The fuel flow was calculated by balancing the combustor inlet and exit enthalpies using the fuel flow and fuel heating value, \( Q_r \). Some sample calculations for the combustor are shown here in Table 7.
Table 7. Sample Calculated Data for Turbojet Combustor: 7.0 in. Orifice Plate.

<table>
<thead>
<tr>
<th>Test Case</th>
<th>Combustor Data</th>
</tr>
</thead>
<tbody>
<tr>
<td>Plate Size</td>
<td>Test Case</td>
</tr>
<tr>
<td>in</td>
<td>% RPM</td>
</tr>
<tr>
<td>7.0</td>
<td>65%</td>
</tr>
<tr>
<td></td>
<td>70%</td>
</tr>
<tr>
<td></td>
<td>75%</td>
</tr>
<tr>
<td></td>
<td>80%</td>
</tr>
<tr>
<td></td>
<td>85%</td>
</tr>
<tr>
<td></td>
<td>90%</td>
</tr>
<tr>
<td></td>
<td>95%</td>
</tr>
</tbody>
</table>

The specific heat at constant pressure was determined using the average of the inlet and exit temperatures and was required for determining the fuel/air ratio. The fuel flow rate and fuel/air ratio in the combustor are shown in Figure 38 and Figure 39, respectively.

![Figure 38. Fuel Flow Rate vs. Engine Speed.](image1)

![Figure 39. Fuel/Air Ratio vs. Engine Speed.](image2)
The trend of fuel flow behaved as expected since it directly relates to the air flow through the combustor, which also increased with engine speed as shown in Figure 32. The fuel flow ranged from 0.012 lb/s to 0.034 lb/s. Unexpectedly, the fuel/air ratio had a clear dependency on the orifice plate size when one would expect the fuel/air ratio to remain fairly constant for most engine speeds and configurations. The smaller orifice plates also exhibit a slightly increasing fuel/air ratio with engine speed while it remains fairly constant for the larger orifice plates. The calculated fuel flow data was compared to data from Figure 40, also Boeing T50-BO-8A model specification.

![Figure 40. Boeing T50-BO-8A Fuel Flow vs. Ambient Temperature at Normal Power.](image)

Figure 40 shows the Boeing turboshift fuel flow as a function of ambient temperature and altitude at normal rated power. Given an ambient temperature of 74°F at normal rated power, the turbojet should consume about 260 lb/hr of fuel according to the dashed red lines. For the experiment, the highest calculated fuel flow rate was 0.034 lb/s, or just 122.4 lb/hr, a 47% error. This was similar to the error in the air flow, which aroused further suspicion about the validity of the collected data. However, the fuel flow and air flow errors are similar which is expected.

E. Calculation of Turbine Exit Pressure

The turbine exit pressure, for the reasons stated earlier, needed to be calculated in order to determine the correct turbine pressure ratio. The turbine exit pressure was calculated using the measured thrust. First, the nozzle exit velocity was determined using Equation 28.

\[
F = \frac{\dot{W}_{\text{exh,corr}}}{g} V_7
\]

where \( F \) is thrust, \( \dot{W}_{\text{exh,corr}} \) is the nozzle corrected flow rate, and \( g \) is the acceleration of gravity. Using the velocity, the enthalpy drop in the nozzle was found using Equation 29.
\[
\Delta H = \frac{V^2}{2gJ}
\]

where \( J \) is the conversion factor of 778 ft-lb/BTU. Knowing the enthalpy drop in the nozzle, the nozzle pressure ratio and nozzle exit pressure were calculated. Using the assumption from Equation 7, the turbine exit pressure was assumed to be equal to the nozzle exit pressure.

**F. Turbine Data**

With the new turbine exit pressure, the calculations for the turbine could be performed. The calculations for the turbine were similar to those for the compressor. Sample calculations for the turbine are shown in Table 8.

**Table 8. Sample Calculated Data for Turbojet Turbine: 7.0 in. Orifice Plate.**

<table>
<thead>
<tr>
<th>Test Case</th>
<th>Throttle</th>
<th>( \pi_{\text{turb}} )</th>
<th>( T_{\text{AS}} )</th>
<th>( H_5 )</th>
<th>( \Delta H_{\text{turb}} )</th>
<th>( \Delta H_{\text{turb, ideal}} )</th>
<th>Work(_{\text{turb}} )</th>
<th>( \eta_{\text{turb}} )</th>
</tr>
</thead>
<tbody>
<tr>
<td>7.0 in</td>
<td>65%</td>
<td>688.06</td>
<td>281.00</td>
<td>62.91</td>
<td>93.96</td>
<td>89.52</td>
<td>66.95%</td>
<td></td>
</tr>
<tr>
<td></td>
<td>70%</td>
<td>688.76</td>
<td>281.18</td>
<td>61.83</td>
<td>105.96</td>
<td>106.79</td>
<td>58.35%</td>
<td></td>
</tr>
<tr>
<td></td>
<td>75%</td>
<td>691.16</td>
<td>281.81</td>
<td>65.68</td>
<td>122.10</td>
<td>126.51</td>
<td>53.79%</td>
<td></td>
</tr>
<tr>
<td></td>
<td>80%</td>
<td>684.34</td>
<td>280.03</td>
<td>83.00</td>
<td>145.58</td>
<td>171.22</td>
<td>57.01%</td>
<td></td>
</tr>
<tr>
<td></td>
<td>85%</td>
<td>711.63</td>
<td>287.14</td>
<td>96.19</td>
<td>172.59</td>
<td>198.53</td>
<td>55.73%</td>
<td></td>
</tr>
<tr>
<td></td>
<td>90%</td>
<td>781.98</td>
<td>305.63</td>
<td>102.11</td>
<td>195.46</td>
<td>230.32</td>
<td>52.24%</td>
<td></td>
</tr>
<tr>
<td></td>
<td>95%</td>
<td>781.98</td>
<td>305.63</td>
<td>102.11</td>
<td>195.46</td>
<td>230.32</td>
<td>52.24%</td>
<td></td>
</tr>
</tbody>
</table>

The pressure ratio ranged from 1.7 to 4.3 and is shown in Figure 41. The pressure ratio behaved as expected, increasing with engine speed. Furthermore, the smaller orifice plates appear to have reduced the turbine pressure ratio. This was also expected as the smaller orifice plates lead to higher turbine exit pressures. The turbine isentropic efficiency ranged from 51% to 79% as shown in Figure 42.

![Figure 41. Turbine Pressure Ratio vs. Engine Speed.](image-url)
The turbine efficiency appears to decrease with higher engine speeds. The efficiency also appears to have a slight dependency on the orifice plate size, where smaller orifice plates produce higher turbine efficiencies. This result seems appropriate as the turbine was likely designed to operate most efficiently with some back pressure, back pressure that could be provided by the addition of a power turbine.

G. Nozzle Data

Calculations for the nozzle were not required for this project, but were performed regardless as further checks of the data. For the nozzle, the pressure ratio, enthalpy drop, and exit Mach number were calculated as shown in Table 9. Plots of the nozzle pressure ratio and exit Mach number are shown in Figure 43 and Figure 44, respectively.

Table 9. Sample Calculated Data for Turbojet Nozzle: 7.0 in. Orifice Plate.

<table>
<thead>
<tr>
<th>Test Case</th>
<th>Nozzle Data</th>
</tr>
</thead>
<tbody>
<tr>
<td>Plate Size</td>
<td>Throttle % RPM</td>
</tr>
<tr>
<td>in</td>
<td></td>
</tr>
<tr>
<td>7.0</td>
<td>65%</td>
</tr>
<tr>
<td></td>
<td>70%</td>
</tr>
<tr>
<td></td>
<td>75%</td>
</tr>
<tr>
<td></td>
<td>80%</td>
</tr>
<tr>
<td></td>
<td>85%</td>
</tr>
<tr>
<td></td>
<td>90%</td>
</tr>
<tr>
<td></td>
<td>95%</td>
</tr>
<tr>
<td></td>
<td>100%</td>
</tr>
</tbody>
</table>
Both the nozzle pressure ratio and exit Mach number behave appropriately. Smaller orifice plates and higher engine speeds lead to greater pressures inside the nozzle, and thus higher pressure ratios. The nozzle pressure ratio ranged from 1.05 to 1.58. The Mach number was calculated to ensure that the exiting flow was subsonic for every test case, verifying the CFD simulation. The Mach number ranged from 0.27 to 0.80, concluding that the exiting flow was subsonic for every case.
H. Compressor Map

Once all of the data was analyzed and checked for errors and anomalies, the compressor map was generated. This map is shown in Figure 45.

![Figure 45. Compressor Performance Map.](image)

Unfortunately, this map bears little resemblance to a typical compressor performance map. The constant speed lines should ideally move to the right away from the stall line and then fall down as in Figure 3. These speed lines however appear to only describe the compressor near the stall line. The speed lines move slightly to the right towards higher flow rates while maintaining a nearly constant pressure ratio. The smaller orifice plates tended to increase the flow rate, stepping the speed lines to the right. However, the final orifice plate, 4.5 in., drastically cut down the flow rate, folding the speed lines back to the left. Furthermore, the speed lines are not smooth or consistent.

A clear and correct trend of this performance map is the up and to the right progression of the speed lines as the speed is increased. This is an intuitive result since higher engine speeds result in higher compressor pressure ratios and flow rates.
I. Turbine Performance Map

The turbine performance map was created next and is shown in Figure 46.

![Turbine Performance Map](image)

Once again, the turbine performance map minimally resembles the one shown in Figure 5. The speed lines should move to the right at low pressure ratios and then go nearly vertical as the turbine becomes choked. The low speed lines, 65%-80%, do exhibit a slight rightward trend, while the high speed lines, 85%-100%, exhibit a slight vertical trend. This may be an indication that the turbine didn’t become choked until the higher engine speeds. Similar to the compressor map, the 4.5 in. orifice plate drastically reduced the flow rate. The speed lines are also very inconsistent.

VII. Conclusion

Despite the odd nature of the resulting compressor and turbine performance maps, the experiment was an overall success. The data was collected with minimal issues and resulted in reasonable and predictable trends once the calculations were completed. The orifice plates also served their purpose of varying the flow rate through the turbojet despite them behaving the opposite of what was expected.

The compressor and turbine performance maps do have a correct overall trend of increased pressure ratio and airflow with higher engine speeds. However, the individual speed lines bear little resemblance to the ideal performance maps of Figure 3 and Figure 4. The reason for this error may lay with several sources. First is that some of the data that was collected was clearly incorrect. For example, the turbine exit pressure for some of the cases dropped below zero gauge pressure. Also in some cases, the combustor exit temperature dropped as engine speed was increased. Finally, some of the recorded temperatures and pressures across the turbine resulted in incorrect isentropic efficiencies as high at 109%. While it is unlikely that these issues were caused by calculation or data acquisition errors, they may be due to the incorrect placement of vital sensors. It was shown earlier that the compressor exit temperature probe was incorrectly placed. Future investigation is needed to analyze the placement of the combustor exit and turbine exit temperature probes.

The performance of the turbojet from run to run was also questionable. From this experiment and those performed in the Aero 401 laboratory, it is known that this turbojet tends to operate inconsistently despite constant procedures and testing conditions. They’re may be several reasons for this, but it is likely due to simply the antiquity of this turbojet, being produced more than four decades ago. More specifically, the fueling system is a leading culprit for the inconsistencies. Even the slightest change in fuel flow is enough to drastically affect the results. A
rebuild and cleaning of the fueling system including the injectors and fuel pump may alleviate some of these issues, but it is likely that these components are simply too worn and too old to work with much stability.

If this experiment was to be performed again with a larger budget, the majority of time would be spent designing a variable area nozzle that does not utilize orifice plates. It is believed that the orifice plates caused some unsteady flow inside the nozzle that resulted in poor turbine exit pressure measurements. Additionally, the orifice plate nozzle required the turbojet to be shut down in order to change the orifice plate. This may have led to inconsistent tests between orifice plates as atmospheric properties do change. Even the properties (temperature, heat soak, geometry) of the turbojet could change between tests as it heats and cools, significantly affecting the results. Furthermore, some effort would be placed into rebuilding/renovating the turbojet, specifically the fueling system as mentioned previously.

The ideal variable area nozzle would be able to open from a diameter larger than 7.0 in. and close down to a diameter of 4.0 in. or less. The nozzle should also be made from steel or stainless steel to be able to withstand the high turbine exit temperatures caused by high engine speeds and low nozzle exit areas. The nozzle exit area would also need to be adjustable from inside the control room. With this, the engine speed could be held constant while the exit area (and flow rate) is varied. In terms of the performance maps, this would cause the engine to move up and down a speed line instead of stepping from one speed line to the next as was done in this project. This undoubtedly would lead to smoother and more reliable speed lines. A continuously variable area nozzle would also allow for the collection of many more data points per speed line, instead of just two to six data points as the orifice plates allowed. The nozzle would also have the turbine exit pressure probe located directly behind the turbine instead of halfway down the nozzle. Essentially the ideal nozzle for this project is one similar to the original variable area nozzle but with a wider range of exit areas.

Regardless, this project begged the question whether or not it is even possible to create a performance map for a compressor or turbine while they are installed into a running piece of turbomachinery. Compressors and turbines are typically “bench tested”, meaning they are installed into a very precise rig that is designed only to test that specific compressor or turbine. They are then rotated using an electric motor that is digitally controlled to maintain an exact RPM while the inlet and outlet conditions are measured using very precise instrumentation. Essentially, the compressor and turbine maps are created in the most ideal of environments. It goes without saying that the environment inside a running turbojet is far from ideal. With unsteady air entering the turbojet, various combustion instabilities, violent flow fields created by the compressor and turbine, and fluctuating rotor speeds and fuel flow rates, it may be almost impossible to generate reasonable performance maps. Furthermore, the compressor and turbine are mechanically linked by the main rotor shaft, meaning that any instability in one is transferred directly to the other. Finally, the Boeing T50-BO-8A, being an engine that is nearly half a century old, is simply not likely to operate with much reliability or consistency to begin with.

Despite the issues and shortcomings of this project, an incredible amount of knowledge was gained through the analysis of the collected data. With the assistance of Mark Waters, hundreds of hours were spent scrutinizing every piece of data and every calculation. Many different approaches utilizing various thermodynamic relations were used to analyze the data. This included everything from the isentropic relations to simple enthalpy balance equations and even weight flow functions. In the end the best suited approach was chosen based on the validity of the collected data and the required calculations.

Overall, this project was an incredible and unexpected learning experience. My knowledge of thermodynamics and engine cycle analysis has reached a whole new level. In terms of education and increased understanding, this project was a smashing success!

Thanks To:
- Mark Waters for assistance with the data analysis
- Eric Pusle and the Mustang 60 machine shop for machining the orifice plates
- Matt Pittenger and the San Luis Jet Center for providing Jet-A fuel
- Joey Sanchez for assistance during testing

---

3 CC3 - Centrifugal Compressor. NASA Glenn Research Center, Cleveland, OH.


Appendix A

**Test Conditions & Constants:**

<table>
<thead>
<tr>
<th>Atmospheric Conditions</th>
<th>Inlet Geometry</th>
<th>Orifice Plate Geometry</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>May 16, 2009, 1pm</strong></td>
<td><strong>D</strong></td>
<td><strong>A</strong></td>
</tr>
<tr>
<td>T&lt;sub&gt;a&lt;/sub&gt; °F</td>
<td>74.000</td>
<td>in</td>
</tr>
<tr>
<td>R</td>
<td>534.000</td>
<td>in²</td>
</tr>
<tr>
<td>P&lt;sub&gt;a&lt;/sub&gt; in-Hg</td>
<td>29.730</td>
<td></td>
</tr>
<tr>
<td>θ --</td>
<td>1.029</td>
<td></td>
</tr>
<tr>
<td>R ft-lbf/slug-°R</td>
<td>1776.000</td>
<td></td>
</tr>
<tr>
<td>y --</td>
<td>1.400</td>
<td></td>
</tr>
<tr>
<td><strong>Cold</strong></td>
<td><strong>D</strong></td>
<td><strong>A</strong></td>
</tr>
<tr>
<td>R ft-lbf/slug-°R</td>
<td>1544.000</td>
<td></td>
</tr>
<tr>
<td>y --</td>
<td>1.333</td>
<td></td>
</tr>
<tr>
<td><strong>Hot</strong></td>
<td><strong>D</strong></td>
<td><strong>A</strong></td>
</tr>
<tr>
<td>R ft-lbf/slug-°R</td>
<td>1544.000</td>
<td></td>
</tr>
<tr>
<td>y --</td>
<td>1.333</td>
<td></td>
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</table>

<table>
<thead>
<tr>
<th>Fuel Data</th>
<th>Type</th>
<th>Jet-A</th>
</tr>
</thead>
<tbody>
<tr>
<td>Qb Btu/lbm-°R</td>
<td>18400</td>
<td></td>
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</tbody>
</table>

<table>
<thead>
<tr>
<th>Instrument Tares</th>
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<th>P&lt;sub&gt;in&lt;/sub&gt;</th>
<th>P&lt;sub&gt;0&lt;/sub&gt;</th>
<th>P&lt;sub&gt;3&lt;/sub&gt;</th>
</tr>
</thead>
<tbody>
<tr>
<td>psig</td>
<td>psig</td>
<td>psig</td>
<td>psig</td>
<td></td>
</tr>
<tr>
<td>0.471</td>
<td>0.161</td>
<td>0.593</td>
<td>0.159</td>
<td></td>
</tr>
</tbody>
</table>
### Appendix B

#### Data Analysis:

<table>
<thead>
<tr>
<th>Test Case</th>
<th>Measured Data</th>
</tr>
</thead>
<tbody>
<tr>
<td>Plate Size</td>
<td>Plate Area</td>
</tr>
<tr>
<td>38.5</td>
<td>65% Below minimum idle</td>
</tr>
<tr>
<td>38.5</td>
<td>70% Below minimum idle</td>
</tr>
<tr>
<td>38.5</td>
<td>75% Below minimum idle</td>
</tr>
<tr>
<td>38.5</td>
<td>80% Below minimum idle</td>
</tr>
<tr>
<td>38.5</td>
<td>85% Below minimum idle</td>
</tr>
<tr>
<td>38.5</td>
<td>100% Below minimum idle</td>
</tr>
<tr>
<td>6.5</td>
<td>Below minimum idle</td>
</tr>
<tr>
<td>6.5</td>
<td>70% Below minimum idle</td>
</tr>
<tr>
<td>6.5</td>
<td>80% Below minimum idle</td>
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<tr>
<td>6.5</td>
<td>90% Below minimum idle</td>
</tr>
<tr>
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<td>100% Below minimum idle</td>
</tr>
<tr>
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<td>Below minimum idle</td>
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<td>6.0</td>
<td>65% Below minimum idle</td>
</tr>
<tr>
<td>6.0</td>
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<tr>
<td>6.0</td>
<td>80% Below minimum idle</td>
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<tr>
<td>6.0</td>
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<td>6.0</td>
<td>100% Below minimum idle</td>
</tr>
<tr>
<td>5.5</td>
<td>Below minimum idle</td>
</tr>
<tr>
<td>5.5</td>
<td>70% Below minimum idle</td>
</tr>
<tr>
<td>5.5</td>
<td>80% Below minimum idle</td>
</tr>
<tr>
<td>5.5</td>
<td>90% Below minimum idle</td>
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<tr>
<td>5.5</td>
<td>100% Below minimum idle</td>
</tr>
<tr>
<td>5.0</td>
<td>Below minimum idle</td>
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<td>5.0</td>
<td>70% Below minimum idle</td>
</tr>
<tr>
<td>5.0</td>
<td>80% Below minimum idle</td>
</tr>
<tr>
<td>5.0</td>
<td>90% Below minimum idle</td>
</tr>
<tr>
<td>5.0</td>
<td>100% Below minimum idle</td>
</tr>
<tr>
<td>4.5</td>
<td>Below minimum idle</td>
</tr>
<tr>
<td>4.5</td>
<td>70% Below minimum idle</td>
</tr>
<tr>
<td>4.5</td>
<td>80% Below minimum idle</td>
</tr>
<tr>
<td>4.5</td>
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</tr>
<tr>
<td>4.5</td>
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</tr>
<tr>
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</tr>
<tr>
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<td>70% Below minimum idle</td>
</tr>
<tr>
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<td>80% Below minimum idle</td>
</tr>
<tr>
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<td>90% Below minimum idle</td>
</tr>
<tr>
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<tr>
<td>3.5</td>
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</tr>
<tr>
<td>3.5</td>
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```
<table>
<thead>
<tr>
<th>H2</th>
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<th>Work Balance (%)</th>
<th>Work Balance (%)</th>
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<th>%H2</th>
<th>Work Balance (%)</th>
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<tbody>
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**Computed T3 Using Work Balance Method and Updated Work Balance Using New T3**

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<thead>
<tr>
<th>H2</th>
<th>%H2</th>
<th>Work Balance (%)</th>
<th>Work Balance (%)</th>
</tr>
</thead>
<tbody>
<tr>
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</table>

<table>
<thead>
<tr>
<th>H2</th>
<th>%H2</th>
<th>Work Balance (%)</th>
<th>Work Balance (%)</th>
</tr>
</thead>
<tbody>
<tr>
<td>100</td>
<td>100</td>
<td></td>
<td></td>
</tr>
</tbody>
</table>
```
| Pressure Plate Size | Throttle | Inlet Data | Test Case | Plate Size
|---------------------|----------|------------|-----------|-------------
|                     | p_in    | V_in      | m_in     | w_in      | M_in | T_d | PR |
|                     | slugs/s | ft/s      | slugs/s  | slugs/s   | lb/s | R   | Recovery |
|                     | pneumatic | pneumatic | pneumatic | pneumatic | pneumatic | pneumatic | pneumatic |
| 6.0                | 0.00223  | 118.47    | 0.0454   | 0.0495    | 1.48  | 0.106 | 542.44 |
|                    | 0.00225  | 117.25    | 0.0451   | 0.0489    | 1.47  | 0.106 | 542.44 |
| 5.5                | 0.00227  | 119.51    | 0.0457   | 0.0495    | 1.49  | 0.106 | 542.44 |
|                    | 0.00227  | 117.73    | 0.0451   | 0.0489    | 1.47  | 0.106 | 542.44 |
| 5.0                | 0.00228  | 119.63    | 0.0457   | 0.0495    | 1.49  | 0.106 | 542.44 |
|                    | 0.00228  | 117.85    | 0.0451   | 0.0489    | 1.47  | 0.106 | 542.44 |
| 4.5                | 0.00229  | 119.75    | 0.0457   | 0.0495    | 1.49  | 0.106 | 542.44 |
|                    | 0.00229  | 117.97    | 0.0451   | 0.0489    | 1.47  | 0.106 | 542.44 |
| 4.0                | 0.00230  | 119.87    | 0.0457   | 0.0495    | 1.49  | 0.106 | 542.44 |
|                    | 0.00230  | 118.09    | 0.0451   | 0.0489    | 1.47  | 0.106 | 542.44 |
| 3.5                | 0.00231  | 119.99    | 0.0457   | 0.0495    | 1.49  | 0.106 | 542.44 |
|                    | 0.00231  | 118.21    | 0.0451   | 0.0489    | 1.47  | 0.106 | 542.44 |

Below minimum idle

70% Below minimum idle

90% Below minimum idle

100% Below minimum idle

MAX: 0.00232 208.1444 0.06713 0.07836 2.52101 0.18183 543.9112 0.99997

MIN: 0.00221 106.89504 0.04039 0.04120 1.32255 0.09459 522.95839 0.99561
<table>
<thead>
<tr>
<th>Test Case</th>
<th>Turbine Data</th>
<th>Combustor Data</th>
</tr>
</thead>
</table>

### Data Test Case

- **Turbine Data**: Test Case
  - **\( T_{\text{in}} \)**: \( 929.17 \) °F
  - **\( H_{\text{in}} \)**: \( 343.90 \) Btu/lb
  - **\( \Delta H_{\text{in}} \)**: \( 152.34 \) Btu/lb
  - **\( T_{\text{in}} \)**: \( 929.17 \) °F
  - **\( H_{\text{in}} \)**: \( 343.90 \) Btu/lb
  - **\( \Delta H_{\text{in}} \)**: \( 152.34 \) Btu/lb

### Combustor Data

- **MIN**: 
  - **\( T \)**: \( 925.76 \) °F
  - **\( H \)**: \( 343.00 \) Btu/lb
  - **\( \Delta H \)**: \( 152.47 \) Btu/lb
  - **Work**
  - **\( T \)**: \( 925.76 \) °F
  - **\( H \)**: \( 343.00 \) Btu/lb
  - **\( \Delta H \)**: \( 152.47 \) Btu/lb

- **MAX**: 
  - **\( T \)**: \( 1373.98 \) °F
  - **\( H \)**: \( 467.39 \) Btu/lb
  - **\( \Delta H \)**: \( 247.76 \) Btu/lb
  - **Work**
  - **\( T \)**: \( 1373.98 \) °F
  - **\( H \)**: \( 467.39 \) Btu/lb
  - **\( \Delta H \)**: \( 247.76 \) Btu/lb

### Table

<table>
<thead>
<tr>
<th>Plate Size</th>
<th>Throttle</th>
<th>( T_{\text{in}} )</th>
<th>( H_{\text{in}} )</th>
<th>( \Delta H_{\text{in}} )</th>
<th>Work</th>
<th>%</th>
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</thead>
<tbody>
<tr>
<td>4.0</td>
<td>100%</td>
<td>( T ) 650°F</td>
<td>( H ) 150 Btu/lb</td>
<td>( \Delta H ) 100 Btu/lb</td>
<td>Work 150 Btu/lb</td>
<td>% 65%</td>
</tr>
<tr>
<td>4.5</td>
<td>100%</td>
<td>( T ) 700°F</td>
<td>( H ) 155 Btu/lb</td>
<td>( \Delta H ) 105 Btu/lb</td>
<td>Work 155 Btu/lb</td>
<td>% 70%</td>
</tr>
<tr>
<td>5.0</td>
<td>100%</td>
<td>( T ) 750°F</td>
<td>( H ) 160 Btu/lb</td>
<td>( \Delta H ) 110 Btu/lb</td>
<td>Work 160 Btu/lb</td>
<td>% 75%</td>
</tr>
</tbody>
</table>

### Plate Size

- **MIN**: 
  - **\( T \)**: \( 925.76 \) °F
  - **\( H \)**: \( 343.00 \) Btu/lb
  - **\( \Delta H \)**: \( 152.47 \) Btu/lb

- **MAX**: 
  - **\( T \)**: \( 1373.98 \) °F
  - **\( H \)**: \( 467.39 \) Btu/lb
  - **\( \Delta H \)**: \( 247.76 \) Btu/lb
<table>
<thead>
<tr>
<th>$\text{V}_{\text{nozz}}$</th>
<th>$\text{V}_{T}$</th>
<th>$\delta H_{\text{nozz}}$</th>
<th>$\delta H_{\text{T}}$</th>
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<td></td>
<td></td>
</tr>
<tr>
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<td></td>
</tr>
<tr>
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<tr>
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<tr>
<td>1.07</td>
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<td>1337.12</td>
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<th>$\text{V}_{\text{nozz}}$</th>
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<th>$\delta H_{\text{nozz}}$</th>
<th>$\delta H_{\text{T}}$</th>
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Appendix C

Visual Basic macro, *therm*:

```
Const tref = 0
Const R = 53.3365
Const CJ = 778
Const G = 32.2
Const AC = 0.24062
Const BB = 0.00017724
Const CC = 0.00000038506
Const DD = 0.00000012662
Const EE = 1.3012E-15
Const AAA = 0.22091
Const BBB = 0.00051822
Const CCC = 0.00000019462
Const DDD = 0.000000000045089
Const EEE = 4.3275E-15

'C     THERMODYNAMIC PROPERTIES
'c
'c specific heat @ temperature = t and fuel/air = f
Function cp(T, F)
    cp = (AC - BB * T + CC * T * T - DD * T * T * T + EE * T * T * T * T + (AAA + BBB * T - CCC * T * T + DDD * T * T * T - EEE * T * T * T) * F) / (1# + F)
End Function

'c molecular weight @ fuel/air = f
Function W(F)
    W = (1 + F) / (0.034522 + 0.035648 * F)
End Function

'c enthalpy difference between temps t2 & t1 with fuel/air = f
'c the equation below is int cp(t,f)*del(t) between t2 and t1
Function h(T2, T1, F)
    h = (1 / (1 + F)) * (AC * (T2 - T1) - (BB / 2) * (T2 * T2 - T1 * T1) + (CC / 3) * (T2 ^ 3 - T1 ^ 3) - (DD / 4) * (T2 ^ 4 - T1 ^ 4) + (EE / 5) * (T2 ^ 5 - T1 ^ 5) + (AAA * (T2 - T1) + (BBB / 2) * (T2 * T2 - T1 * T1) - (CCC / 3) * (T2 ^ 3 - T1 ^ 3) + (DDD / 4) * (T2 ^ 4 - T1 ^ 4) - (EEE / 5) * (T2 ^ 5 - T1 ^ 5)) * F)
End Function

'c entropy difference between temps t2 & t1 with fuel/air = f
'c the equation below is int cp(t,f)*del(t)/t between t2 and t1
Function S(T2, T1, F)
    S = (1 / (1 + F)) * (AC * Log(T2 / T1) - BB * (T2 - T1) + (CC / 2) * (T2 ^ 2 - T1 * T1) + (CC / 3) * (T2 ^ 3 - T1 ^ 3) - (DD / 3) * (T2 ^ 3 - T1 ^ 3) + (EE / 4) * (T2 ^ 4 - T1 ^ 4) + F * (AAA * (T2 - T1) + (BBB / 2) * (T2 * T2 - T1 * T1) - (CCC / 3) * (T2 ^ 3 - T1 ^ 3) + (DDD / 4) * (T2 ^ 4 - T1 ^ 4) - (EEE / 5) * (T2 ^ 5 - T1 ^ 5))) * F)
End Function
```
Log(T2 / T1) + BBB * (T2 - T1) - (CCC / 2) * (T2 * T2 - T1 * T1) + (DDD / 3) * (T2 ^ 3 - T1 ^ 3) - (EEE / 4) * (T2 ^ 4 - T1 ^ 4))

End Function

'c  ratio of specific heats for a given temperature & fuel/air

Function GAM(T, F)
    GAM = 1 / (1 - (1545 / (W(F) * 778 * cp(T, F))))
End Function

'c  flow function for a given flow Mach no., temperature & fuel/air

Function XEX(T, F)
    XEX = (GAM(T, F) + 1) / (2 * (GAM(T, F) - 1))
End Function

'c flow function

Function WFF(sm, T, F)
    WFF = Sqr(GAM(T, F) * 32.2 / (1545 / W(F))) * sm * (1 / (1 + ((GAM(T, F) - 1) / 2) * sm ^ 2)) ^ XEX(T, F)
End Function
Visual Basic macro, thprop:

Const tref = 0
Const R = 53.3365
Const CJ = 778
Const G = 32.2
Const AC = 0.24062
Const BB = 0.000017724
Const CC = 0.000000038506
Const DD = 0.0000000012662
Const EE = 1.3012E-15
Const AAA = 0.22091
Const BBB = 0.00051822
Const CCC = 0.00000019462
Const DDD = 0.000000000045089
Const EEE = 4.3275E-15

Function thprop_jtype_1(pr, tin, far, eta, outputsequenceNumber)
'COMPRESSION WITH A KNOWN PRESSURE RATIO, TEMP IN, FUEL/AIR, AND ADIABATIC EFFICIENCY

'd An isentropic compression or expansion is being computed for a
'd known pressure ratio.
d' In the gas tables, del(s) = cp*del(t)/t - r*delp/p.
d' When integrated, S = int(cp*del(t)/t) - r*ln(pr).
d' In this routine, the function s(t2,t1,f) is the difference in the
d' first term on the rhs between the temperatures t2 & t1.
d' (with the gas having a fuel/air of f)
d' DS computed below is the difference in the second term between the
d' pressures p2 & p1 (expressed as the pressure ratio, pr).
d' An iteration proceeds until the difference, DDS, is driven to zero.
d' (i.e., S2 - S1)

ds = (R / CJ) * Log(pr)
touti1 = 0
touti = tin * Exp(ds / 0.24062)
kount = 0

10

DDS = ds - S(touti, tin, far)
touti = touti + DDS * touti / cp(touti, 0)
kount = kount + 1
If (Abs(touti - touti1) < 0.5 Or kount > 200) Then GoTo 16
touti1 = touti
GoTo 10

d' The function H(t2,t1,f) is the enthalpy difference between t2 & t1
d' with the gas having a fuel/air of f (t2 > t1).

16

delh = h(touti, tin, far) / eta
tout = tin + delh / 0.24062

tout1 = 0
kount = 0

d' iterate to drive ddh = 0.
ddh = delh - h(tout, tin, far)
tout = tout + ddh / cp(tout, far)
'c tout = tout + ddh / cp(tout, far)

kount = kount + 1
If (Abs(tout - tout1) = 0.5 Or kount > 200) Then GoTo 200

tout1 = tout
GoTo 17

200
If (outputsequenceNumber = 1) Then thprop_jtype_1 = touti
If (outputsequenceNumber = 2) Then thprop_jtype_1 = tout
If (outputsequenceNumber = 3) Then thprop_jtype_1 = delh
End Function

Function thprop_jtype_2(pr, tin, far, eta, outputsequenceNumber)

' EXPANSION WITH A KNOWN PRESSURE RATIO, TEMP IN, FUEL/AIR AND ADIABATIC EFFICIENCY

'c An isentropic compression or expansion is being computed for a
'c known pressure ratio.
'c In the gas tables, del(s) = cp*del(t)/t - r*delp/p.
'c When integrated, s = int(cp*del(t)/t) - r*ln(pr).
'c In this routine, the function s(t2,t1,f) is the difference in the
'c first term on the rhs between the temperatures t2 & t1.
'c (with the gas having a fuel/air of f)
'c DS computed below is the difference in the second term between the
'c pressures p2 & p1 (expressed as the pressure ratio, pr).
'c An iteration proceeds until the difference, DDS, is driven to zero.
'c (i.e., S2 = S1)

ds = (R / CJ) * Log(pr)
ds = -ds
touti1 = 0
touti = tin * Exp(ds / 0.24062)
kount = 0

10

DDS = ds + S(tin, touti, far)
touti = touti + DDS * touti / cp(touti, 0)
kount = kount + 1
If (Abs(touti - touti1) < 0.5 Or kount > 200) Then GoTo 16

touti1 = touti
GoTo 10

'd The function H(t2,t1,f) is the enthalpy difference between t2 & t1
'd with the gas having a fuel/air of f (t2 > t1).

16

delh = h(tin, touti, far) * eta
tout = tin - delh / 0.24062

tout1 = 0
kount = 0

'c iterate to drive ddh = 0.
ddh = delh - h(tin, tout, far)
tout = tout - ddh / cp(tout, far)
'c tout = tout + ddh / cp(tout, far)
kount = kount + 1
If (Abs(tout - tout1) < 0.5 Or kount > 200) Then GoTo 200
tout1 = tout
GoTo 17
200
If (outputsequenceNumber = 1) Then thprop_jtype_2 = touti
If (outputsequenceNumber = 2) Then thprop_jtype_2 = tout
If (outputsequenceNumber = 3) Then thprop_jtype_2 = delh
End Function

Function thprop_jtype_3(delh, tin, far, eta, outputsequenceNumber)
'c EXPANSION WITH A KNOWN CHANGE IN ENTHALPY (delh is +), TEMP IN, FUEL/AIR & ADIBATIC EFFICIENCY

tout = tin - delh / 0.24062
tout1 = 0
kount = 0
'c iterate to drive ddh = 0.
30    ddh = h(tin, tout, far) - delh
    CPT3 = cp(tout, far)
kount = kount + 1
tout = tout + ddh / CPT3
If (Abs(tout - tout1) < 0.5 Or kount > 200) Then GoTo 31
tout1 = tout
GoTo 30
'c ideal enthalpy change
31    delhi = delh / eta
    touti = tin - delhi / 0.24062
touti1 = 0
kount = 0
'c iterate to drive ddh = 0.
32    ddh = h(tin, touti, far) - delhi
    touti = touti + ddh / cp(touti, far)
kount = kount + 1
If (Abs(touti - touti1) < 0.5 Or kount > 200) Then GoTo 33
touti1 = touti
GoTo 32
33
'c note that the constant .50356 = 1/(1545/778)
    pr = Exp(S(tin, touti, far) * W(far) * 0.50356)
GoTo 200
If (outputsequenceNumber = 1) Then thprop_jtype_3 = touti
If (outputsequenceNumber = 2) Then thprop_jtype_3 = tout
If (outputsequenceNumber = 3) Then thprop_jtype_3 = pr
End Function

Function thprop_jtype_4(tin, far, outputsequenceNumber)
'WITH A GIVEN TEMPERATURE & FUEL/AIR
'COMPUTE GAMMA, SPECIFIC HEAT & GAS CONSTANT -- TEMP SHOULD BE STATIC
'COMPUTE TOTAL ENTHALPY WITH TOTAL TEMPERATURE or STATIC ENTHALPY WITH STATIC
TEMPERATURE

cpin = cp(tin, far)
ga = GAM(tin, far)
rgas = 1545 / W(far)
hin = h(tin, tref, far)
If (outputsequenceNumber = 1) Then thprop_jtype_4 = cpin
If (outputsequenceNumber = 2) Then thprop_jtype_4 = ga
If (outputsequenceNumber = 3) Then thprop_jtype_4 = rgas
If (outputsequenceNumber = 4) Then thprop_jtype_4 = hin
End Function

Function thprop_jtype_5(tin, far, sm, outputsequenceNumber)
'COMPUTE FLOW FUNCTION WITH GIVEN TOTAL TEMP, FUEL/AIR, AND MACH NO.
Dim gamma(3) As Double, gamx(3) As Double
Tt = tin
ga = GAM(Tt, far)
For i = 1 To 3
st = Tt / (1 + (ga - 1) / 2 * sm ^ 2)
ga = GAM(st, far)
st = Tt / (1 + (ga - 1) / 2 * sm ^ 2)
gamma(i) = GAM(st, far)
stx = Tt / ((ga + 1) / 2)
gamx(i) = GAM(stx, far)
Next i
rgas = 1545 / W(far)
wffg = WFF(sm, st, far)
wffMach1 = WFF(1, stx, far)
AqAstar = wffMach1 / wffg
If (outputsequenceNumber = 1) Then thprop_jtype_5 = ga
If (outputsequenceNumber = 2) Then thprop_jtype_5 = rgas
If (outputsequenceNumber = 3) Then thprop_jtype_5 = wffg
If (outputsequenceNumber = 4) Then thprop_jtype_5 = AqAstar
Function thprop_jtype_6(t3, T4, LHV, eta, outputsequenceNumber)
'COMPUTE FUEL/AIR RATIO WITH TOTAL TEMP IN, TOTAL TEMP OUT, HEATING VALUE &
'COMBUSTION EFFICIENCY

    Const tolerance = 0.00001
    Const probestep = 0.000001

    far3 = 0
    H3 = h(t3, tref, far3)

    far4_guess = 0
    far4 = tolerance + probestep

    Do
        If (far4 > far4_guess) Then far4_guess = far4 + probestep
        If (far4 < far4_guess) Then far4_guess = far4 - probestep

        H4 = h(T4, tref, far4_guess)
        far4 = (H4 - H3) / (eta * LHV - H4)
        i = i + 1
    Loop Until (Abs(far4 - far4_guess) <= tolerance)

    If (outputsequenceNumber = 1) Then thprop_jtype_6 = far4

End Function

Function thprop_jtype_7(hin, far, outputsequenceNumber)
'COMPUTE TEMPERATURE WITH KNOWN ENTHALPY & FUEL/AIR

    Const tolerance = 0.00001
    'Const probestep = 0.001

    'c initial guess of gas temperature
    Temp = 0.0000002 * hin ^ 3 - 0.0009 * hin ^ 2 + 4.379 * hin - 10.955
    h_calc = thprop_jtype_4(Temp, far, 4)

    'THIS LOOP NEEDS TO BE CHANGED SO IT DOESN'T REQUIRE SO MANY ITERATIONS
    Do
        If (h_calc > hin) Then
            'temp = temp - probestep
            j = -1
        ElseIf (h_calc < hin) Then
            'temp = temp + probestep
            j = 1
        End If

        h_error = Abs(h_calc - hin) / hin
        probestep = 100 * h_error

        'temp = j * temp + probestep * 10000 * h_error
        Temp = Temp + j * probestep
        h_calc = thprop_jtype_4(Temp, far, 4)
        i = i + 1
    Loop Until (Abs(h_calc - hin) <= tolerance)

    If (outputsequenceNumber = 1) Then thprop_jtype_7 = Temp
End Function

Function thprop_jtype_8(delh, tin, far, etapoly, outputsequenceNumber)
' This function is the same as function no. 3 except input is with a polytropic efficiency

    Const tolerance = 0.0001
    kount = 0

    ' Initial guess of eta
    eta = etapoly

10
    pr = thprop_jtype_3(delh, tin, far, eta, 3)
    tout = thprop_jtype_3(delh, tin, far, eta, 2)
    tbar = (tin + tout) / 2
    ga = GAM(tbar, far)

    eta_calc = (1 - (1 / pr) ^ (((ga - 1) / ga) * etapoly)) / (1 - (1 / pr) ^ ((ga - 1) / ga))

    kount = kount + 1
    diff = (eta - eta_calc) / eta
    If (Abs(diff) <= tolerance Or kount > 500) Then GoTo 20
    eta = eta + Abs(diff) / 2
    GoTo 10
20
    touti = thprop_jtype_3(delh, tin, far, eta, 1)
    tout = thprop_jtype_3(delh, tin, far, eta, 2)
    pr = thprop_jtype_3(delh, tin, far, eta, 3)

    If (outputsequenceNumber = 1) Then thprop_jtype_8 = eta
    If (outputsequenceNumber = 2) Then thprop_jtype_8 = touti
    If (outputsequenceNumber = 3) Then thprop_jtype_8 = tout
    If (outputsequenceNumber = 4) Then thprop_jtype_8 = pr

End Function

Function thprop_jtype_9(Tt, far, wffi, msect, outputsequenceNumber)

    ' This routine will find the Mach number that corresponds to a known value of the flow function.
    ' Input Values
    '   tt    total temperature in deg R
    '   far   fuel-air ratio
    '   wffi  flow function = w*sqrt(tt)/A/pt
    '   msect indicator, if msect = 1, subsonic speed; if msect = 2, supersonic speed
    ' Output Value
    '   Mach_by_Wff   Mach number

    ' This is a first guess for gamma.  The value is updated as part of the iteration below.
    ' The iteration starts at Mach = 1. Values are reduced or increased according to the value of
    ' msect until the problem is bounded.  Three points are determined on the flow function curve
    ' and a quadratic equation is determined and used to solve for Mach Number

    Dim M_guess(3) As Double, Wff_calc(3) As Double, tstatic(3) As Double, gamma(3) As Double
' initialization of first 3 points
probe_step = 0.0004

' subsonic section
' 3 values of M_guess
If msect = 1 Then
    M_guess(1) = 1
    M_guess(2) = M_guess(1) - probe_step
    M_guess(3) = M_guess(2) - probe_step
End If

' supersonic section
' 3 values of M_guess
If msect = 2 Then
    M_guess(1) = 1
    M_guess(2) = M_guess(1) + probe_step
    M_guess(3) = M_guess(2) + probe_step
End If

' initialization
' 3 values of WFF_calc are computed
' However, only last value of st and gamma are saved
' gamma = 1.4
For i = 1 To 3
    st = Tt / (1 + (gamma - 1) / 2 * M_guess(i) ^ 2)
    gamma = thprop_jtype_4(st, far, 2)
    st = Tt / (1 + (gamma - 1) / 2 * M_guess(i) ^ 2)
    gamma = thprop_jtype_4(st, far, 2)
    Wff_calc(i) = thprop_jtype_5(st, far, M_guess(i), 3)
    ga = 1.4
    st = Tt / (1 + (ga - 1) / 2 * M_guess(i) ^ 2)
    ga = GAM(st, far)
    st = Tt / (1 + (ga - 1) / 2 * M_guess(i) ^ 2)
    gamma(i) = GAM(st, far)
    tstatic(i) = Tt / (1 + (gamma(i) - 1) / 2 * M_guess(i) ^ 2)
    Wff_calc(i) = WFF(M_guess(i), tstatic(i), far)
Next i

' iteration loop
If Wff_calc(1) < wffi Then
    ' there is no solution, b/c Wffi is > maximum value
    ' wffi set to max value and exit from iteration
    next_iteration = 0
    wffi = Wff_calc(1)
Else
    next_iteration = 1
End If

tolerance = 0.00003
Do
    For i = 1 To 3
        ' In this first loop, the computed Wff_calc is within
        ' tolerance, then prepare to exit the program
        If Abs(Wff_calc(i) - wffi) <= tolerance Then
            ' problem is converged
            Msolved = M_guess(i)

        If outputsequenceNumber = 1 Then
            thprop_jtype_9 = Msolved
    Next i
    If next_iteration = 1 Then
        ' problem solved, exit iteration loop
        Exit Do
    End If
    ' problem not converged, continue iteration loop
End Do
End If

ga = 1.4
st = Tt / (1 + (ga - 1) / 2 * Msolved ^ 2)
ga = GAM(st, far)
st = Tt / (1 + (ga - 1) / 2 * Msolved ^ 2)
ga = GAM(st, far)
st = Tt / (1 + (ga - 1) / 2 * Msolved ^ 2)
If outputsequenceNumber = 2 Then
    thprop_jtype_9 = st
End If

If outputsequenceNumber = 3 Then
    thprop_jtype_9 = ga
End If

next_iteration = 0
Exit For
End If

' At this point there is still is no solution -- continue to step in Mach number to bound the problem. Once it is bounded then compute a quadratic equation and solve for the result. If next_iteration = 1 Then
If (Wff_calc(3) < wffi) And (wffi < Wff_calc(1)) Then
    'The problem is bounded, calculate the Mach number and jump out the loop

    'THE FOLLOWING PROGRAM WILL DETERMINE THE COEFFICIENTS OF THE QUADRATIC EQUATION - AX**2 + BX + C - DERIVED FROM A NEWTON SERIES USING THREE POINTS IN A PLANE

    y1 = M_guess(1)
x1 = Wff_calc(1)
y2 = M_guess(2)
x2 = Wff_calc(2)
y3 = M_guess(3)
x3 = Wff_calc(3)

    D1 = (y1 - y2) / (x1 - x2)
    D2 = (y2 - y3) / (x2 - x3)
    D3 = (D1 - D2) / (x1 - x3)
a = D3
b = D1 - D3 * (x2 + x1)
c = D3 * x1 * x2 - D1 * x1 + y1
Msolved = a * wffi * wffi + b * wffi + c

If outputsequenceNumber = 1 Then
    thprop_jtype_9 = Msolved
End If

If outputsequenceNumber = 2 Then
    thprop_jtype_9 = st
End If

If outputsequenceNumber = 3 Then
    thprop_jtype_9 = ga
End If

next_iteration = 0
Exit For
End If

End If
If outputsequenceNumber = 2 Then
    thprop_jtype_9 = st
End If

If outputsequenceNumber = 3 Then
    thprop_jtype_9 = ga
End If

' WFF1 = thprop_jtype_5(tt, far, thprop_jtype_9, 3)
next_iteration = 0
Else

' Problem is not bounded -- continue to step in Mach number and recompute gamma, st and Wff_calc

For i = 1 To 2
    M_guess(i) = M_guess(i + 1)
    Wff_calc(i) = Wff_calc(i + 1)
    tstatic(i) = tstatic(i + 1)
    gamma(i) = gamma(i + 1)
Next i

If msect = 1 Then
    M_guess(3) = M_guess(3) - probe_step
End If
If msect = 2 Then
    M_guess(3) = M_guess(3) + probe_step
End If

st = Tt / (1 + (gamma(3) - 1) / 2 * M_guess(3) ^ 2)
gamma(3) = GAM(st, far)
st = Tt / (1 + (gamma(3) - 1) / 2 * M_guess(3) ^ 2)
gamma(3) = GAM(st, far)
tstatic(3) = Tt / (1 + (gamma(3) - 1) / 2 * M_guess(3) ^ 2)
Wff_calc(3) = WFF(M_guess(3), tstatic(3), far)
End If
End If
Loop Until (next_iteration = 0)
End Function

Function thprop_jtype_11(pr, tin, far, etapoly, outputsequenceNumber)
' THIS ROUTINE IS THE SAME AS FUNCTION NO. 1, EXCEPT EFFICIENCY IS POLYTROPIC

' Initial guess of eta
eta = etapoly

For i = 1 To 3
    tout = thprop_jtype_1(pr, tin, far, eta, 2)
    tbar = (tin + tout) / 2
    ga = GAM(tbar, far)
    eta_calc = (pr ^ ((ga - 1) / ga) - 1) / (pr ^ ((ga - 1) / ga / etapoly) - 1)
    eta = eta_calc
Next i

touti = thprop_jtype_1(pr, tin, far, eta, 1)
tout = thprop_jtype_1(pr, tin, far, eta, 2)
delh = thprop_jtype_1(pr, tin, far, eta, 3)
If (outputsequenceNumber = 1) Then thprop_jtype_11 = eta
If (outputsequenceNumber = 2) Then thprop_jtype_11 = touti
If (outputsequenceNumber = 3) Then thprop_jtype_11 = tout
If (outputsequenceNumber = 4) Then thprop_jtype_11 = delh
End Function

Function thprop_jtype_12(delh, tin, far, eta, outputsequenceNumber)
'COMPRESSION WITH A KNOWN ENTHALPY CHANGE (delh is +), TEMP IN, FUEL/AIR & 'ADIABATIC EFFICIENCY

tout = tin + delh / 0.24062
tout1 = 0
kount = 0

'c iterate to drive ddh = 0.
30    ddh = h(tout, tin, far) - delh
    CPT3 = cp(tout, far)
kount = kount + 1
tout = tout - ddh / CPT3
    If (Abs(tout - tout1) < 0.5 Or kount > 200) Then GoTo 31
    tout1 = tout
    GoTo 30

'c ideal enthalpy change
31    delhi = delh * eta
    touti = tin + delhi / 0.24062
    touti1 = 0
    kount = 0

'c iterate to drive ddh = 0.
32    ddh = h(touti, tin, far) - delhi
    touti = touti - ddh / cp(touti, far)
kount = kount + 1
    If (Abs(touti - touti1) < 0.5 Or kount > 200) Then GoTo 33
    touti1 = touti
    GoTo 32
33
'c note that the constant .50356 = 1/(1545/778)
    pr = Exp(S(touti, tin, far) * W(far) * 0.50356)
    GoTo 200
200
    If (outputsequenceNumber = 1) Then thprop_jtype_12 = touti
    If (outputsequenceNumber = 2) Then thprop_jtype_12 = tout
    If (outputsequenceNumber = 3) Then thprop_jtype_12 = pr
End Function

Function thprop_jtype_13(delh, tin, far, etapoly, outputsequenceNumber)
'This function is the same as function no. 12 except input is with a polytropic efficiency

Const tolerance = 0.0001
kount = 0

' Initial guess of eta
eta = etapoly

For i = 1 To 5
    pr = thprop_jtype_12(delh, tin, far, eta, 3)
    tout = thprop_jtype_12(delh, tin, far, eta, 2)
    tbar = (tin + tout) / 2
    ga = GAM(tbar, far)
    eta_calc = (pr ^ ((ga - 1) / ga) - 1) / (pr ^ ((ga - 1) / ga / etapoly) - 1)
    eta = eta_calc
Next i

touti = thprop_jtype_12(delh, tin, far, eta, 1)
tout = thprop_jtype_12(delh, tin, far, eta, 2)
pr = thprop_jtype_12(delh, tin, far, eta, 3)

If (outputsequenceNumber = 1) Then thprop_jtype_13 = eta
If (outputsequenceNumber = 2) Then thprop_jtype_13 = touti
If (outputsequenceNumber = 3) Then thprop_jtype_13 = tout
If (outputsequenceNumber = 4) Then thprop_jtype_13 = pr
End Function

Function thprop_jtype_14(Ttin, far, Mach, outputsequenceNumber)
'COMPUTE GAMMA & STATIC TEMPERATURE
'Inputs are Ttin -- total temperature; far -- fuel-air ratio; Mach -- Mach no.
    ga = 1.4
    For i = 1 To 4
        st = Ttin / (1 + (ga - 1) / 2 * Mach ^ 2)
        ga = GAM(st, far)
        Ttqts = Ttin / st
        Ptqps = Ttqts ^ (ga / (ga - 1))
    Next i

    If (outputsequenceNumber = 1) Then thprop_jtype_14 = ga
    If (outputsequenceNumber = 2) Then thprop_jtype_14 = Ttqts
    If (outputsequenceNumber = 3) Then thprop_jtype_14 = Ptqps
End Function

Function thprop_jtype_15(pr, tin, far, etapoly, outputsequenceNumber)
'This function is the same as function no. 2 except input is with a polytropic efficiency
    Const tolerance = 0.0001
    kount = 0

    ' Initial guess of eta
    eta = etapoly

    delh = thprop_jtype_2(pr, tin, far, eta, 3)
    tout = thprop_jtype_2(pr, tin, far, eta, 2)
    tbar = (tin + tout) / 2
    ga = GAM(tbar, far)

    eta_calc = (1 - (1 / pr) ^ (((ga - 1) / ga) * etapoly)) / (1 - (1 / pr) ^ ((ga - 1) / ga))
kount = kount + 1

diff = (eta - eta_calc) / eta
If (Abs(diff) <= tolerance Or kount > 500) Then GoTo 20
eta = eta + Abs(diff) / 2
GoTo 10

20

touti = thprop_jtype_2(pr, tin, far, eta, 1)
tout = thprop_jtype_2(pr, tin, far, eta, 2)
delh = thprop_jtype_2(pr, tin, far, eta, 3)

If (outputsequenceNumber = 1) Then thprop_jtype_15 = eta
If (outputsequenceNumber = 2) Then thprop_jtype_15 = touti
If (outputsequenceNumber = 3) Then thprop_jtype_15 = tout
If (outputsequenceNumber = 4) Then thprop_jtype_15 = delh

End Function