Presentation Objectives:

• UNDERLINE BENEFITS OF HIGHER OPERATING PRESSURES AND TEMPERATURES FOR ENGINE SPECIFIC OUTPUT POWER AND FUEL CONSUMPTION USUALLY OUTWEIGHING GREATER COMPLEXITY AND RELATED COST

• EMPHASIZE CROSS-DISCIPLINARY NATURE OF ADVANCED TURBINE DESIGN

• DEMONSTRATE NECESSITY FOR COOLING GAS TURBINE HOT SECTION COMPONENTS WHEN ENGINE SUPERIOR PERFORMANCE IS REQUIRED

• DISCUSS ADVANCED COOLING TECHNIQUES AND COMPRESSED AIR DELIVERY SYSTEMS FOR MAIN HOT SECTION COMPONENTS

• ILLUSTRATE THERMO-MECHANICAL DESIGN FEATURES MINIMIZING COOLING PENALTIES
Comparing Dwarf and Hercules

Core of PW-200 and -100 Engines for F117 (DC17/F22 at BR 6.9) and F100 (F16/15 at BR 0.78)
Variation of Engine Efficiency with TIT and PR

Thermal Efficiency %

<table>
<thead>
<tr>
<th>Temperature (°C)</th>
<th>Pressure Ratio</th>
</tr>
</thead>
<tbody>
<tr>
<td>1210 (2200F)</td>
<td>870</td>
</tr>
<tr>
<td>1320 (2400F)</td>
<td>1000</td>
</tr>
<tr>
<td>1480 (2700F)</td>
<td>1200</td>
</tr>
<tr>
<td>2040 (3700F)</td>
<td>1600</td>
</tr>
</tbody>
</table>

Small Engines Often with Centrifugal Compressors

Advanced Aero Engines

Specific output kW/(kg/s)

Opt PR = [T3/T1] \( \frac{\gamma}{2 / (\gamma - 1)} \)

\( \eta_t = 1 - \left[1 / PR \right] \frac{\gamma - 1}{\gamma} \)
Engine Advancement Trend

Emission envelope

Commercial Aero Engine

Military Aero Engine

Industrial Gas Turbine

Material Temperature Capability

Conventionally Cast Alloys

DS & Single Crystal

DS Eutectics

1960

1970

1980

1990

2000

Years

Pressure Ratio

Compressor Discharge Temperature

TRIT °C

°C
Cooling Flow Estimates (based on industry experience and mid 80th superalloys) with TBC on liner and airfoils

Tip clearance control from bypass external loop

CFM56-3 Cooling Flow Circuits

% comp. flow

<p>| | |</p>
<table>
<thead>
<tr>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>0.5</td>
</tr>
<tr>
<td>2</td>
<td>0.5</td>
</tr>
<tr>
<td>3</td>
<td>3.8</td>
</tr>
<tr>
<td>4</td>
<td>3.2</td>
</tr>
<tr>
<td>5</td>
<td>0.5</td>
</tr>
<tr>
<td>6</td>
<td>0.7</td>
</tr>
<tr>
<td>7</td>
<td>3.5</td>
</tr>
<tr>
<td>8</td>
<td>0.5</td>
</tr>
<tr>
<td>9</td>
<td>0.5</td>
</tr>
<tr>
<td>10</td>
<td>2.0</td>
</tr>
<tr>
<td>11</td>
<td>0.3</td>
</tr>
<tr>
<td>12</td>
<td>0.3</td>
</tr>
<tr>
<td>13</td>
<td>0.3</td>
</tr>
<tr>
<td>14</td>
<td>0.3</td>
</tr>
<tr>
<td>15</td>
<td>0.3</td>
</tr>
</tbody>
</table>

Total 17.2%
Typical Combustor Exit Averaged Radial and Circumferential Temperature Variation

Pattern factor

\[ \text{PF} = \frac{(T_{\text{max}} - T_{\text{mean}})}{(T_{\text{mean}} - T_{\text{out comp}})} \]

\[ \text{PF} = 0.2 - 0.4 \]

Non-Dimensional Temperature = \( \frac{(T_{\text{local}} - T_3)}{(T_4 - T_3)} \)

RPF = 0.1 - 0.15

T mean

% Blade Height

0 0.2 0.4 0.6 0.8 1

0 0.5 1.0 1.5 2.0

700 800 900 1000 1100 1200 1300 1400 K
Liner Film Cooling Options

1. Cold Side

Hot Side

Wiggle-strips

2. Splash cooling ring

Hot Side

Stacked ring

3. Impingement-film

Hot Side

Advanced Low Emission Options

4. Effusion

5. Dome

6. Augmented back-side convection

7. TBC

Details of Combustor Liner Cooling
Liner Backside Cooling Using Dimpled Surface
Variation of Average Radial Temperature Profile Through HP Turbine

For 20,000 hrs life

Blade max Tm

Vane max Tm

Combustor exit TIT spread
Example of HP Turbine Cooling Flow Distribution

ACC
Counter-rotating discs
T.fan air

Flow Distribution
• **THERMODYNAMIC LOSSES**: Removed Heat From The Airfoils And Temperature Reduction in the Mainstream Due to Mixing W. Spent Cooling Air. For Internally Cooled Nozzles and Blades $\Delta \eta_{\text{isentr}} = 0.2-0.3\%$; For Advanced Engines with Film Cooled Airfoils $\Delta \eta_{\text{isentr}} = 1.0-2.0\%$

• **EQUIVALENT LOSSES RELATED TO COMPRESSION OF THE COOLING AIR**: $\Delta \eta_{\text{isentr}}$ Vary from 0.5 To 2% Depending on a Stage of Air Bleeding From Compressor and Location of the Spent Air Reentry

• **AIR PUMPING LOSSES INSIDE COOLED TURBINE BLADES**: $\Delta \eta_{\text{isentr}} \sim 0.6\%$, If air is Discharged Into The Tip Gap

• **AERODYNAMIC MIXING LOSSES**: Drag Effect On Mainstream Flow Due To Reintroduction Of The Slower Moving Cooling Air $\Delta \eta_{\text{isentr}} = 0.2-0.9\%$ Depending On Location Of The Air Discharge Along An Airfoil (Film And Trailing Edge)

• **LOSSES FROM THICKER PROFILE AND TRAILING EDGES OF COOLED AIRFOILS**: Profile Losses: Insignificant For Thicker Le And Middle Portion Of The Airfoil But High Up To $\Delta \eta_{\text{isentr}} = 4.0\%$ For Thick Trailing Edges

• **LOSSES FROM ENTRIES OF THE COOLING AIR INTO THE MAINSTREAM FROM DISC CAVITIES AND STATOR GAPS**: $\Delta \eta = 2-3\%$ In Case Of 1% Radial Inflow And $\Delta \eta = 1\%$ When Entering Flow Is Directed Parallel To The Mainstream

**Losses Related To Turbine Cooling**

(ref. S.Kopelev “Cooled Gas Turbine Blades- Thermo-aero design”, Moscow, 1983)
Effect of Cooling on Aero Engine Performance

Effect of Cooling on SFC and Specific Thrust

Preferable Flowpath Forming Features
Gaspath Outer and Inner Structure Sealing
Example Of Advanced Nozzle Design
Nozzle Endwall Secondary Flows
“Suppression” Of Horseshoe Vortex With Upstream Film

Endwall 0.8% PCD upstream film flow

Endwall 2% PCD upstream film flow

Flow 3D Prediction: flow streak of low momentum coolant is pushed toward the suction side

(higher film blowing ratio, steeper angle)
Proper introduction of endwall film cooling can provide full film coverage and significantly reduce stage pressure losses suppressing a “horseshoe” vortex.
Nozzle Film Cooling Design for High TIT

As cast  
Finished part  
Cooled air buffered squealer tip  

Tip fence seal with cooling discharge near the trailing edge of the pressure side  

High temperature shroudless blades
Blade Cooling With Combination of Convective, Impingement and Film
(courtesy of GE)
RB211 St.1 Shrouded Blade Cooling Design Evolution

(courtesy of RR)
Effect of Blade Loading on TRIT$_{Rel}$

(ref. “Turbine Gas Path Design For Aeroengines” S. Kopelev, Moscow, 1984)

$\frac{\text{(TRIT-Trel) at } \Delta H/u^2 > 2}{\text{(TRIT-Trel) at } \Delta H/u^2 = 2}$

$\Delta H/u^2$ TRIT - Trel = $u^2 (\gamma - 1) / 2R\gamma$

where $\gamma$-spec. heat ratio and R gas const

(assumed a near 0 reaction at the root)

Example:

20% increase in load ~13% rise in $\Delta T_{rel}$
Airfoil External Thermal Boundary Conditions

$\text{Nu} = 0.0239 \times \text{Re}^{0.805}$

- Transition
- Turbulent
- Laminar
- Stagnation Point
- Possible Goertler Vortices
- Possible Transition Due to Laminarization
- Unsteady Wake Flow

Possible shock that can result in separation

Diagram showing heat transfer coefficient vs. normalized axial distance (X/C).

Legend:
1 - 2.5% tip gap, Tu = 9.7%
2 - 1.5% tip gap, Tu = 9.7%
3 - 1% tip gap, Tu = 9.7%
4 - 2.5% tip gap, Tu = 6.1%
5 - 1.5% tip gap, Tu = 6.1%
6 - 1% tip gap, Tu = 6.1%
\[ \varepsilon = \frac{(T_g - T_m)}{(T_g - T_{ci})} \]

**Required Airfoil Cooling Effectiveness**
Assumes combustor PF = 0.3 and St.1 PR = 2.5
(Compressor PR optimized for TIT) 20,000hrs life

Combined internal-film

Internal w. TBC

St. 1 NOZZLE

St. 1 BLADE

Approximate Guidance for Selection of Cooling Techniques for St.1 GP Turbine
Approximate Effect of TBC (YTZ) on Metal Temperature of Effectively Cooled Airfoil

Before film holes are considered

<table>
<thead>
<tr>
<th>Temperature reduction °C</th>
</tr>
</thead>
<tbody>
<tr>
<td>50</td>
</tr>
<tr>
<td>40</td>
</tr>
<tr>
<td>30</td>
</tr>
<tr>
<td>20</td>
</tr>
<tr>
<td>10</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>TBC thickness mm</th>
</tr>
</thead>
<tbody>
<tr>
<td>0.05</td>
</tr>
<tr>
<td>0.1</td>
</tr>
<tr>
<td>0.15</td>
</tr>
<tr>
<td>0.2</td>
</tr>
<tr>
<td>0.25</td>
</tr>
<tr>
<td>0.3</td>
</tr>
</tbody>
</table>

Few recommended coatings:

- TBC Yttrium stabilized zirconium (YTZ) - airfoils
- PWA 270 (NiCoCrAlY) – oxidation-erosion resistant coating - vanes
- PWA 264 – TBC for endwalls
- TBC 100 for platforms
- PWA 275 – aluminide coat. for internal vane cavity
Effect of Film Cooling Hole Shaping
In general, film cooling effectiveness from discrete holes is less effective than from slot injection due to jets from the individual holes penetrating into the mainstream and permitting the hot mainstream gas to flow under the secondary fluid close to the surface to be cooled.

Hot gas penetration and mixing are not present with injection slots.

Long slots are rarely used in airfoils because of mechanical design considerations. Shaped film holes provide a practical compromise between cooling effectiveness and structural integrity.

In two-dimensional film cooling, the film cooling effectiveness can generally be correlated as a function of blowing rate, or mass flux ratio.

Impact of Film Hole Shaping on C. Effectiveness
Stage Efficiency Loss Due to TE Blockage
Effect of Vane Trailing Edge Thickness on Efficiency of Turbine Stage (based on R-R studies)
Utilization of LE Spent Cooling Flow for Tip "Flag" Cooling
Loss estimate: \( \frac{P}{P_\infty} = \gamma \frac{m_c}{m_\infty} M_a^2 / 2 \left( 1 + \frac{T_c}{T_\infty} - \frac{2V_c}{V_\infty} \cos \alpha \right) \)

Ref. Hartsel, J. E., 1972,

**Film Related Turbine Efficiency Losses**

(Ref. B. Barry, 1976, von Karman LS 83)
Nozzle Film Cooling Discharge Penalties

(Ref. B.Barry, 1976, von Karman LS 83)
Thermal Efficiency as a $\Psi(TIT)$ for Different Cooling Techniques

Ref. “Limitation on GT Performance Imposed by Large Cooling Flows” J. Harlok, et.al IGTI 2000-635
Typical Airfoil Cooling Penalties

Major Requirements for Nozzle and Blade Cooling System Design

- Nozzle Vanes have to be designed for a peak hot-spot temperature anywhere in the LE except near the tip and hub (10-15% from the endwalls)
- Air flow for the shower head cooling of the LE practically does not affect stage performance but reduces effective gas temperature that has to be compensated by higher TIT
- Relatively high endwall film cooling flow introduced upstream of the LE is beneficial for both cooling and turbine performance
- When airfoil film cooling is required, the long compound angle holes provide larger heat transfer area and improved cooling effectiveness
- Certain flow pressure margin is required in the internal cavity upstream of cooling air discharge to the mainstream
- Spent cooling air discharge through the trailing edge or on the pressure side near the TE results in very low performance penalties
- Spent cooling air discharged into the blade tip region usually results in improved stage efficiency
- Design features in the blade interior providing conducting path between suction and pressure surfaces, especially near the TE assist in more uniform temperature distribution along blade profile
- Special design effort is required to prevent cooling air heating by friction in the disc cavity; preswirler has to be always considered as a part of the blade cooling supply system
**Benefits of Air Preswirling Into Disc Cavity**

\[
\Delta T_0 \text{ accel} = \frac{(2UV\cos\alpha - U^2)}{2R\gamma/\gamma - 1} \quad \text{and} \quad \Delta T_0 \text{ frict} = \frac{U^2}{2R\gamma/\gamma - 1}
\]
Effect of Tip Clearance on Turbine Stage Efficiency.
Modulated cooling air flow can control stator growth and blade tip clearance.
Combination of Passive and Active Tip Clearance Control (LM6000 familiarization)
* DISCHARGE THE SPENT COOLING FLOW AS EARLY AS POSSIBLE ALONG THE GAS PATH
* BLADE SPENT COOLING AIR DISCHARGE INTO THE BLADE TIP GAP MAY MINIMIZE PENALTIY AND EVEN IMPROVE STAGE PERFORMANCE
* USE SHOWER HEAD COOLING FOR THE LEADING EDGE OF THE FIRST STAGES OF AIRFOILS ONLY IF NECESSARY
* DESIGN THE COOLING SYSTEM ATTEMPTING TO DISCHARGE THE AIR AT A TEMPERATURE APPROACHING ALLOWABLE LOCAL METAL SURFACE TEMPERATURE
* MINIMIZE MIXING LOSSES BY CLOSELY MATCHING VELOCITY VECTORS BETWEEN MAINSTREAM AND DISCHARGED COOLING FLOWS. THIS REQUIRES MINIMIZING PRESSURE LOSSES IN THE INTERNAL COOLING PASSAGES
* AVOID COOLING AIR DISCHARGE ON SUCTION SIDE OF THE AIRFOIL, ESPECIALLY DOWNSTREAM OF THE THROAT
* REDUCE INTERNAL COOLING FLOWS UTILIZING THERMAL BARRIER COATING (TBC)
* USE PRE-SWIRLING MECHANISM FOR BLADE COOLING SUPPLY SYSTEM LOWERING THE RELATIVE TEMPERATURE OF THE COOLANT AND REDUCING DISC FRICTION LOSSES

SUMMARY: Main Design Rules for Minimizing Cooling Penalties