15th Israeli Symposium Jet Engines and Gas Turbines, Technion, Haifa Nov 17, 2016

“Getting Outmost Benefits from Turbine Cooling”

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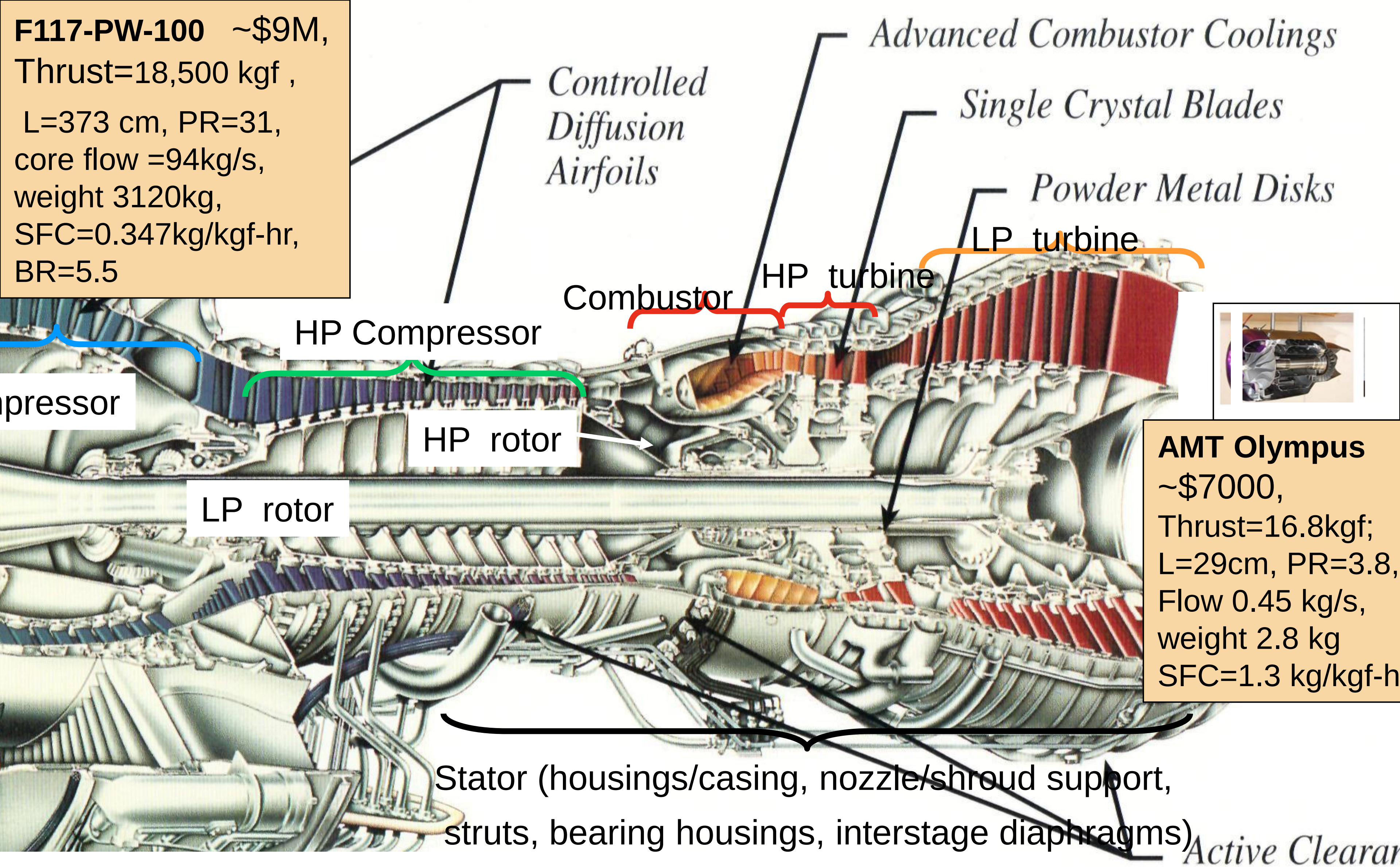
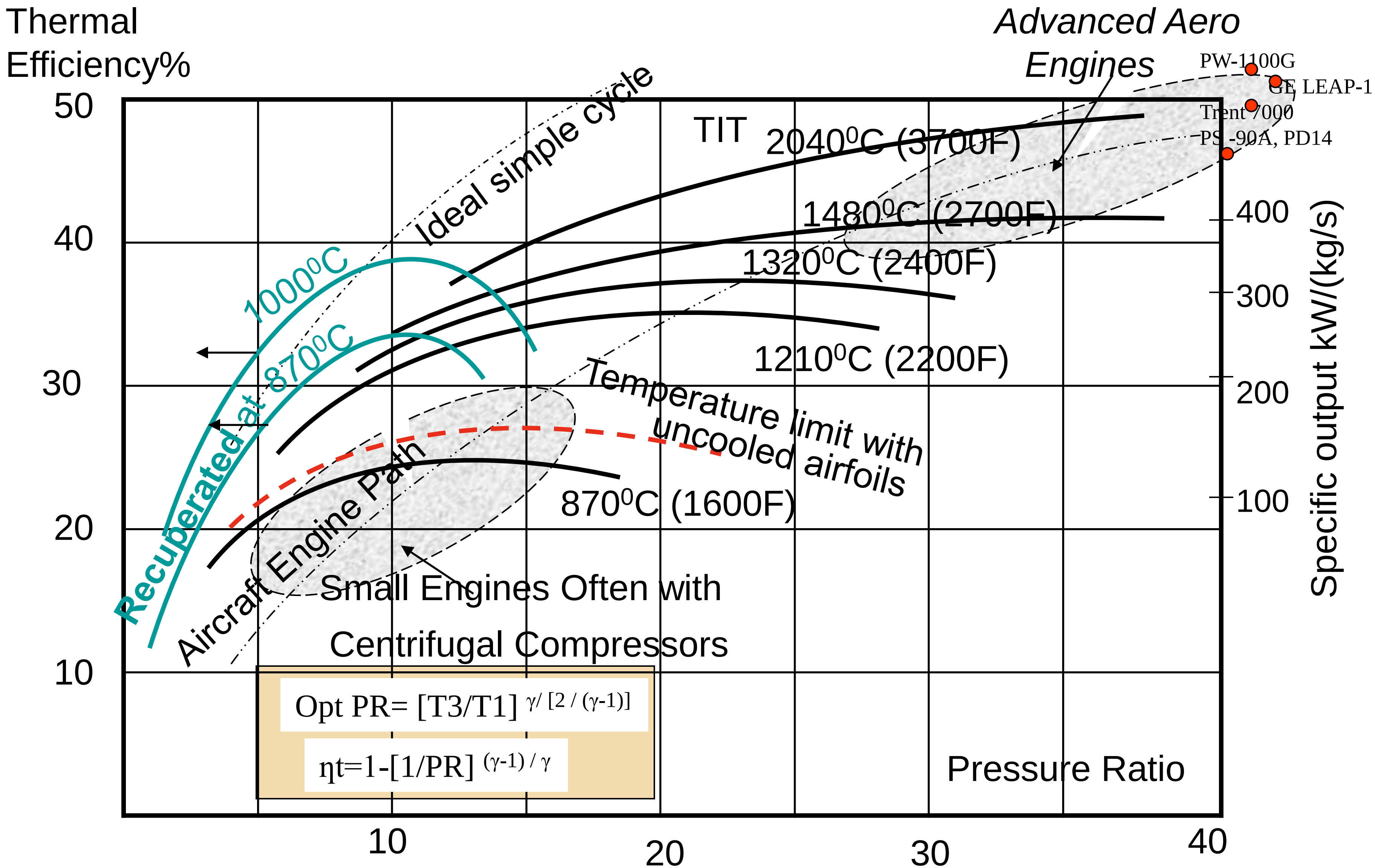
Presentation Objectives:

- **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**

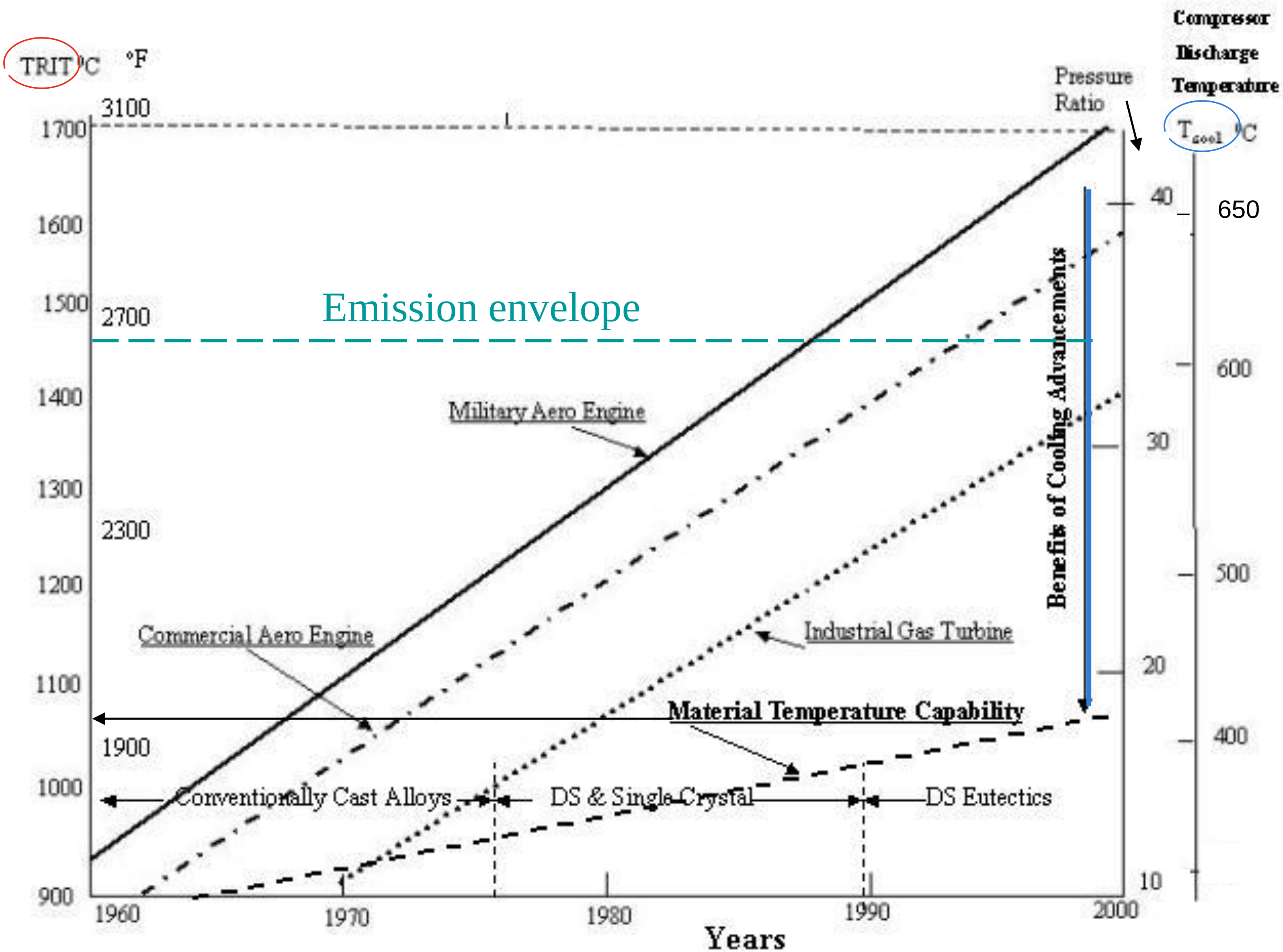


Comparing Dwarf and Hercules

Core of PW-200 and-100 Engines for F117 (DC17/F22 at BR 6.9) and F100 (F16/15 at BR 0.78)



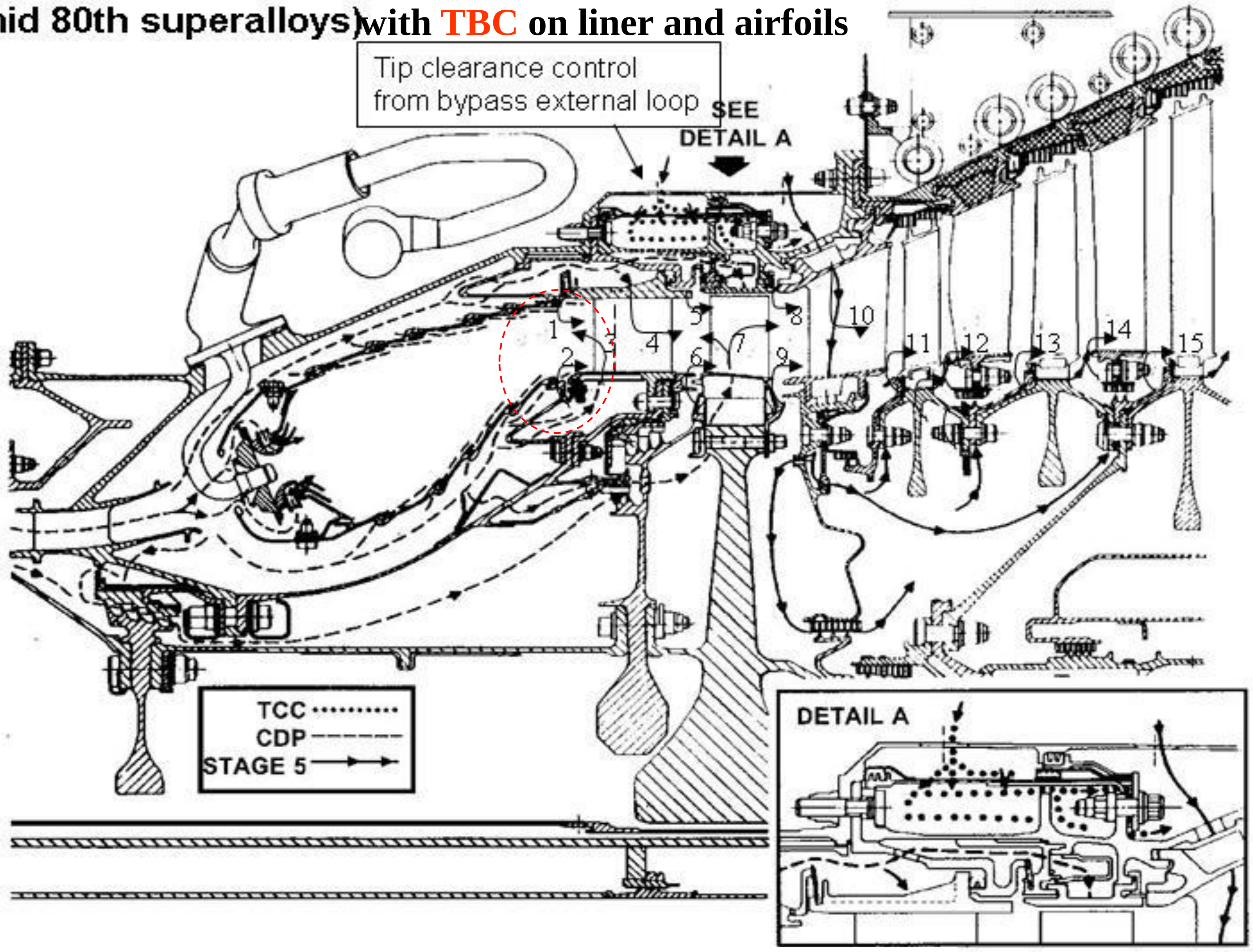
Variation of Engine Efficiency with TIT and PR



Engine Advancement Trend

HOT SECTION COOLING

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



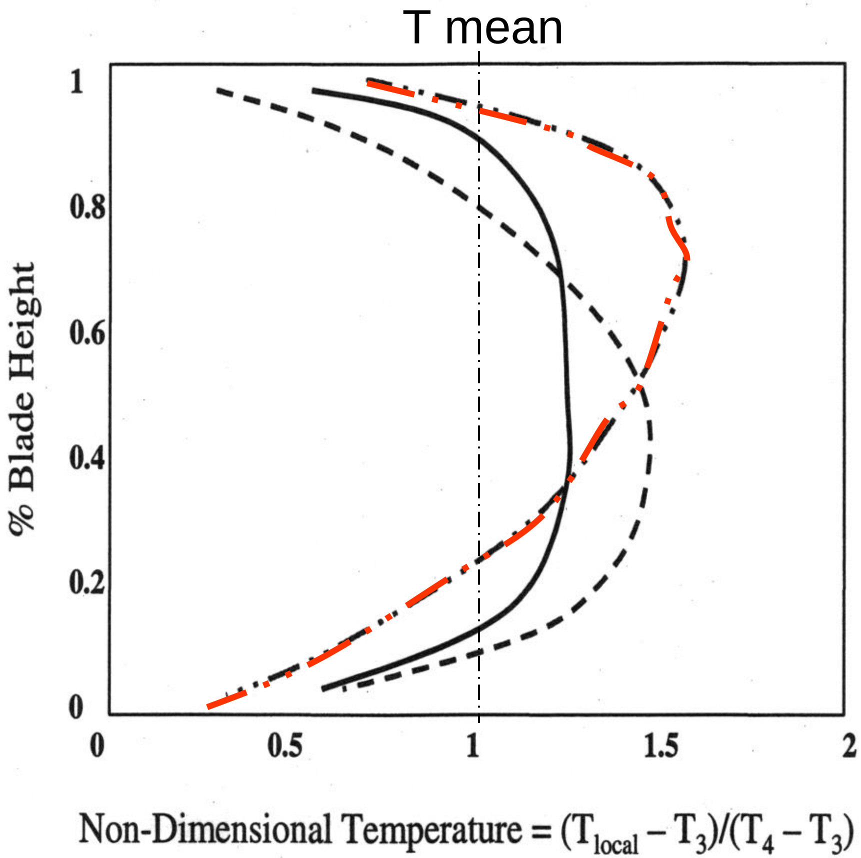
% comp. flow

1	0.5
2	0.5
3	3.8
4	3.2
5	0.5
6	0.7
7	3.5
8	0.5
9	0.5
10	2.0
11	0.3
12	0.3
13	0.3
14	0.3
15	0.3

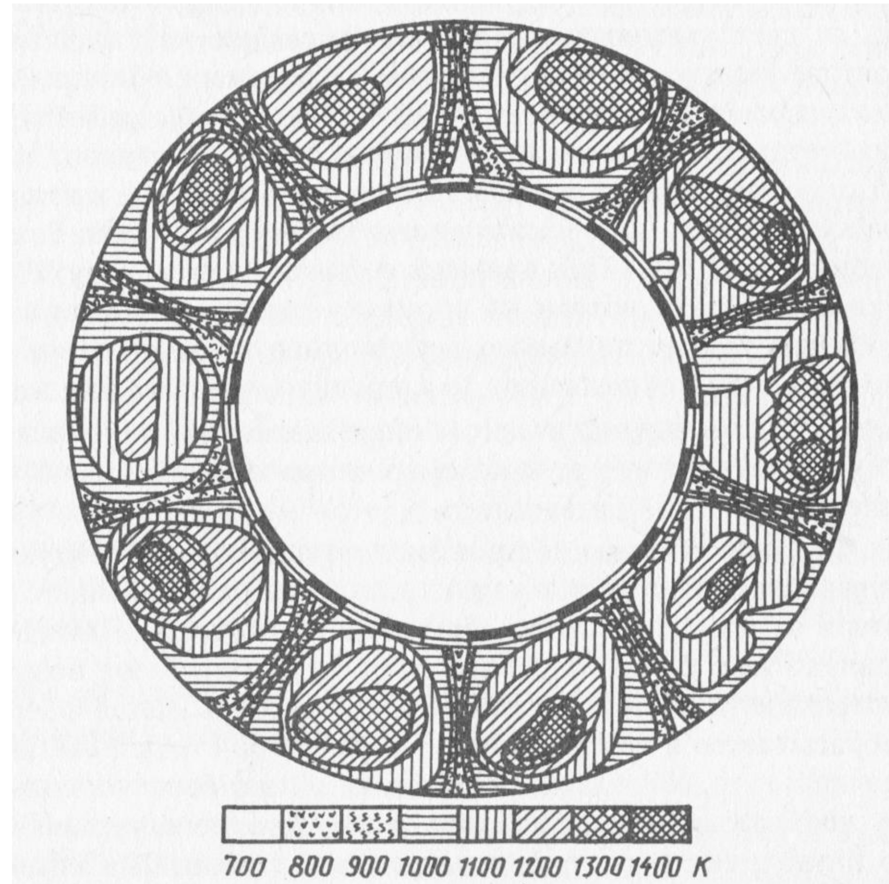
Total 17.2%

CFM56-3 Cooling Flow Circuits

Typical Combustor Exit Averaged Radial and Circumferential Temperature Variation



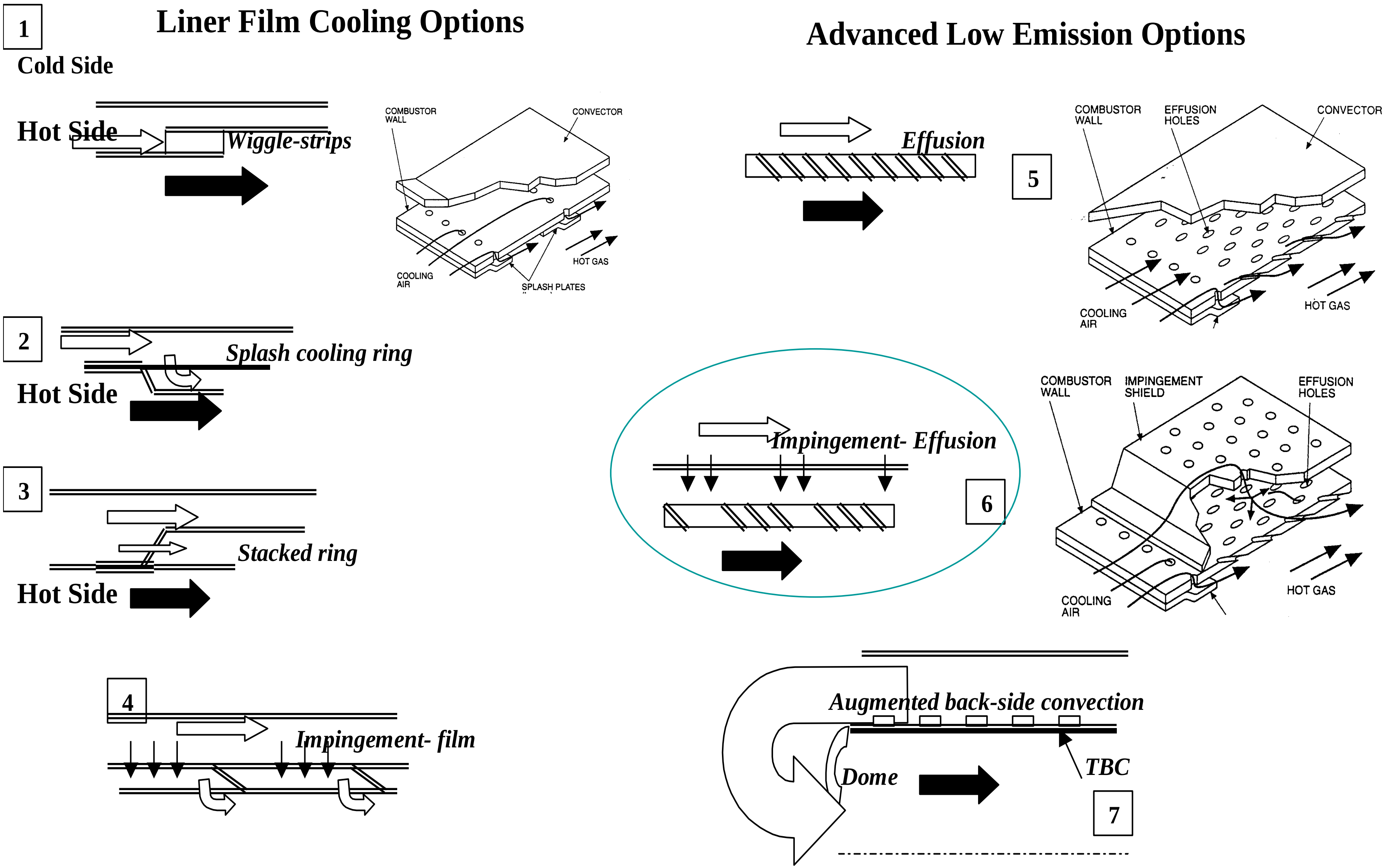
RPF=0.1-0.15



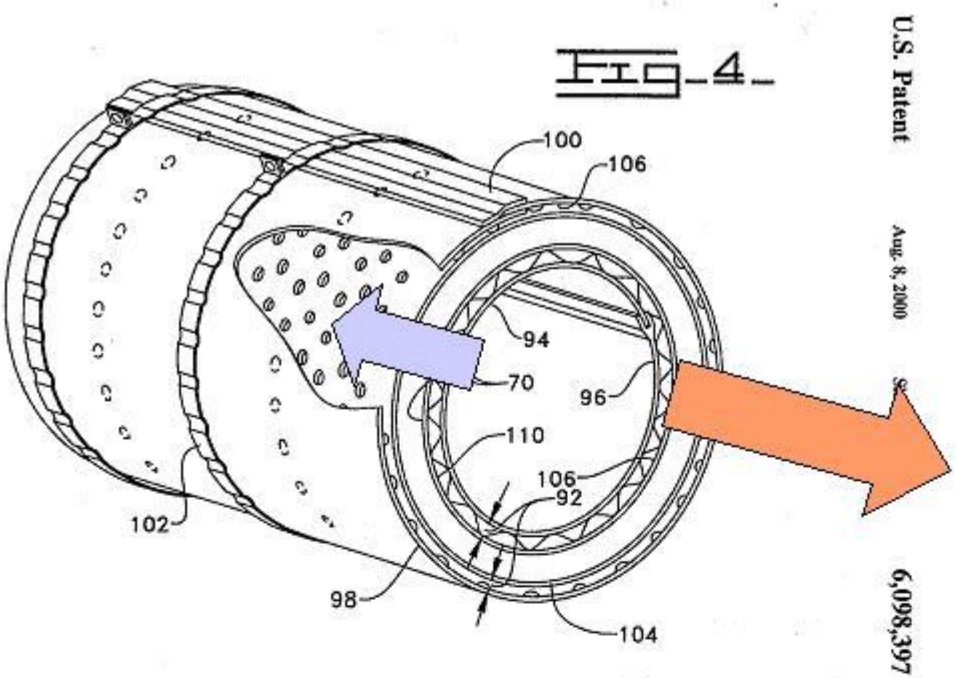
Pattern factor

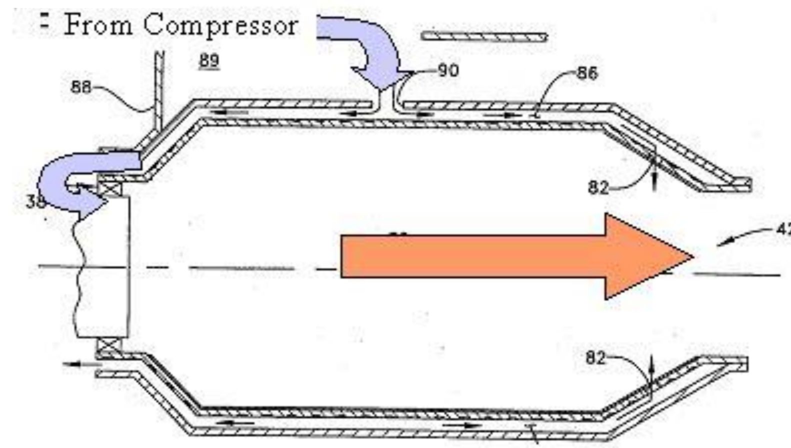
$$\boxed{PF = (T_{max} - T_{mean})/(T_{mean} - T^{out}_{comp})}$$

PF=0.2 - 0.4

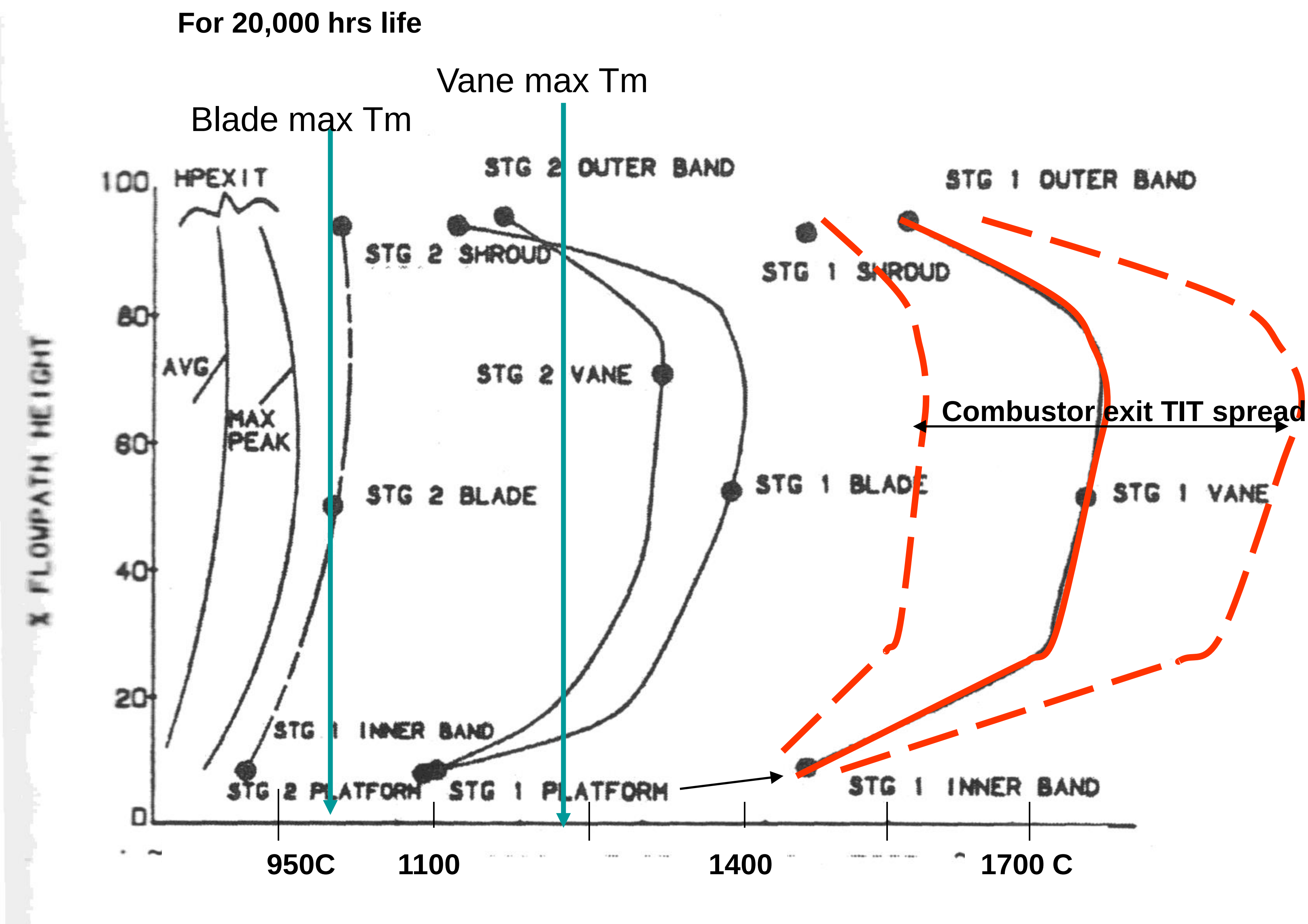


Details of Combustor Liner Cooling



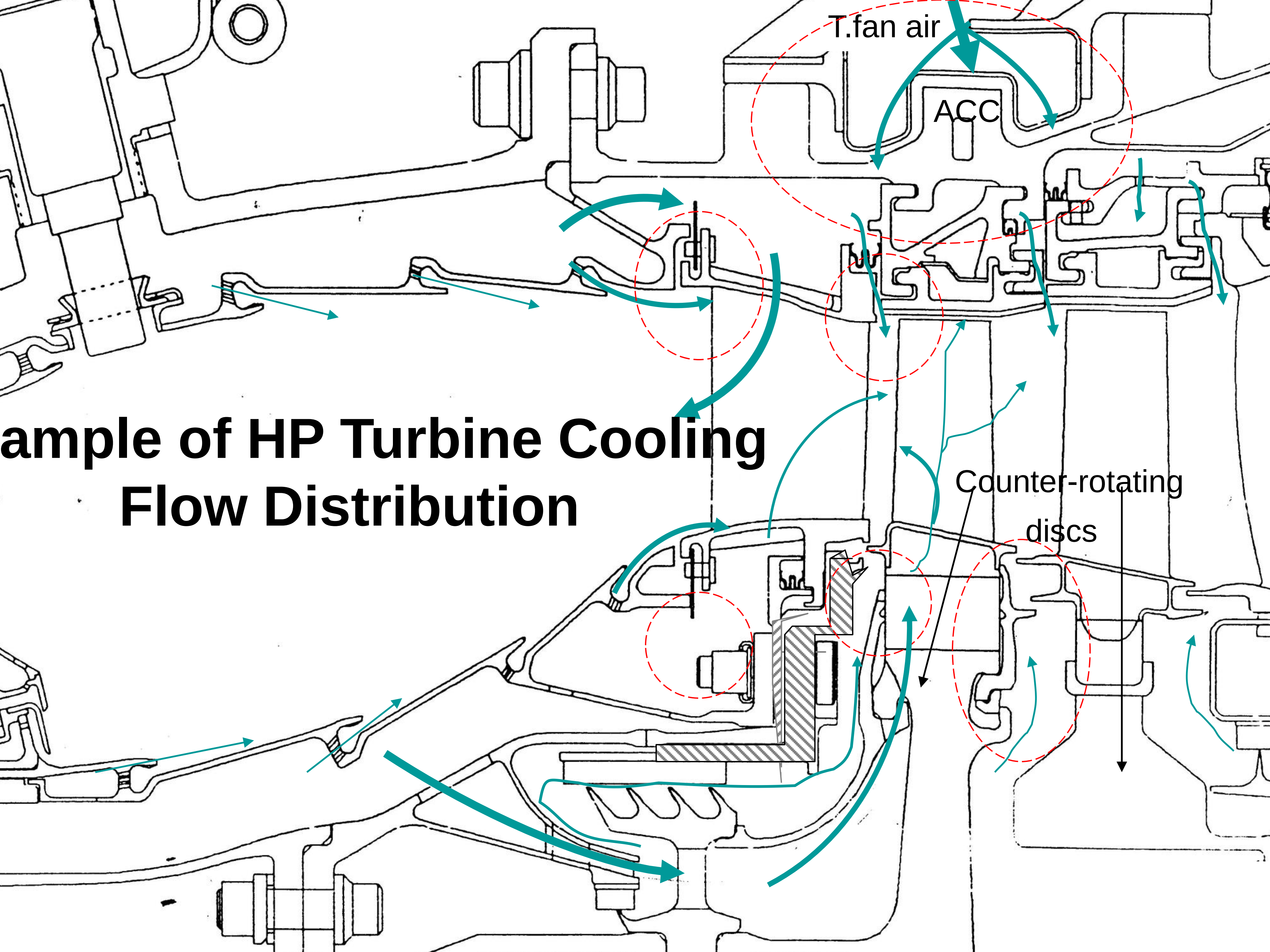


Liner Backside Cooling Using Dimpled Surface



Variation of Average Radial Temperature Profile Through HP Turbine

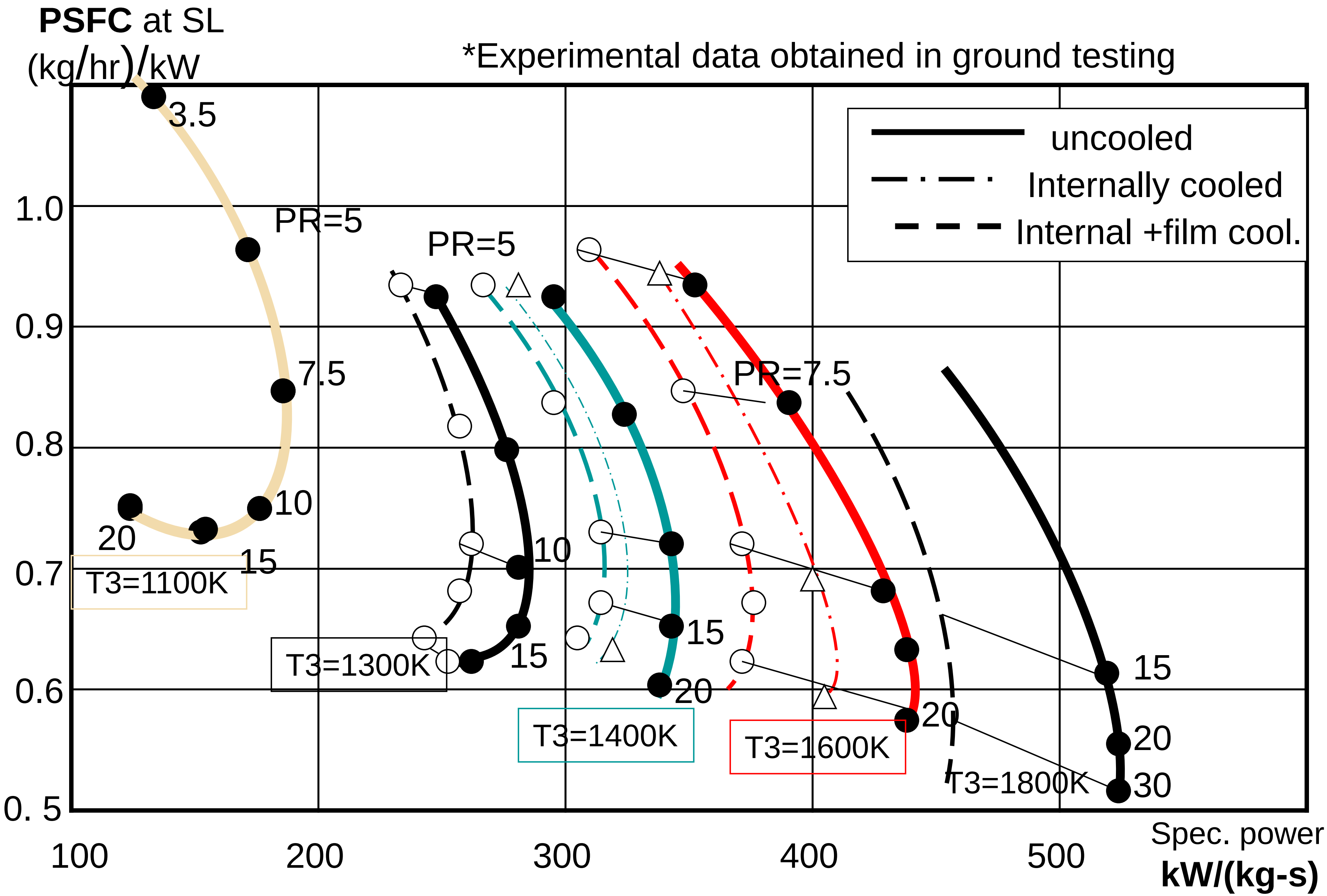
ample of HP Turbine Cooling Flow Distribution



- **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 $\Delta\eta_{isentr}$=0.2-0.3%; For Advanced Engines with Film Cooled Airfoils $\Delta\eta_{isentr}$ =1.0-2.0 %
- **EQUIVALENT LOSSES RELATED TO COMPRESSION OF THE COOLING AIR:** $\Delta\eta_{isentr}$ 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**: $\Delta\eta_{isentr}$ ~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 $\Delta\eta_{Isentr}$=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 $\Delta\eta_{isentr}$=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

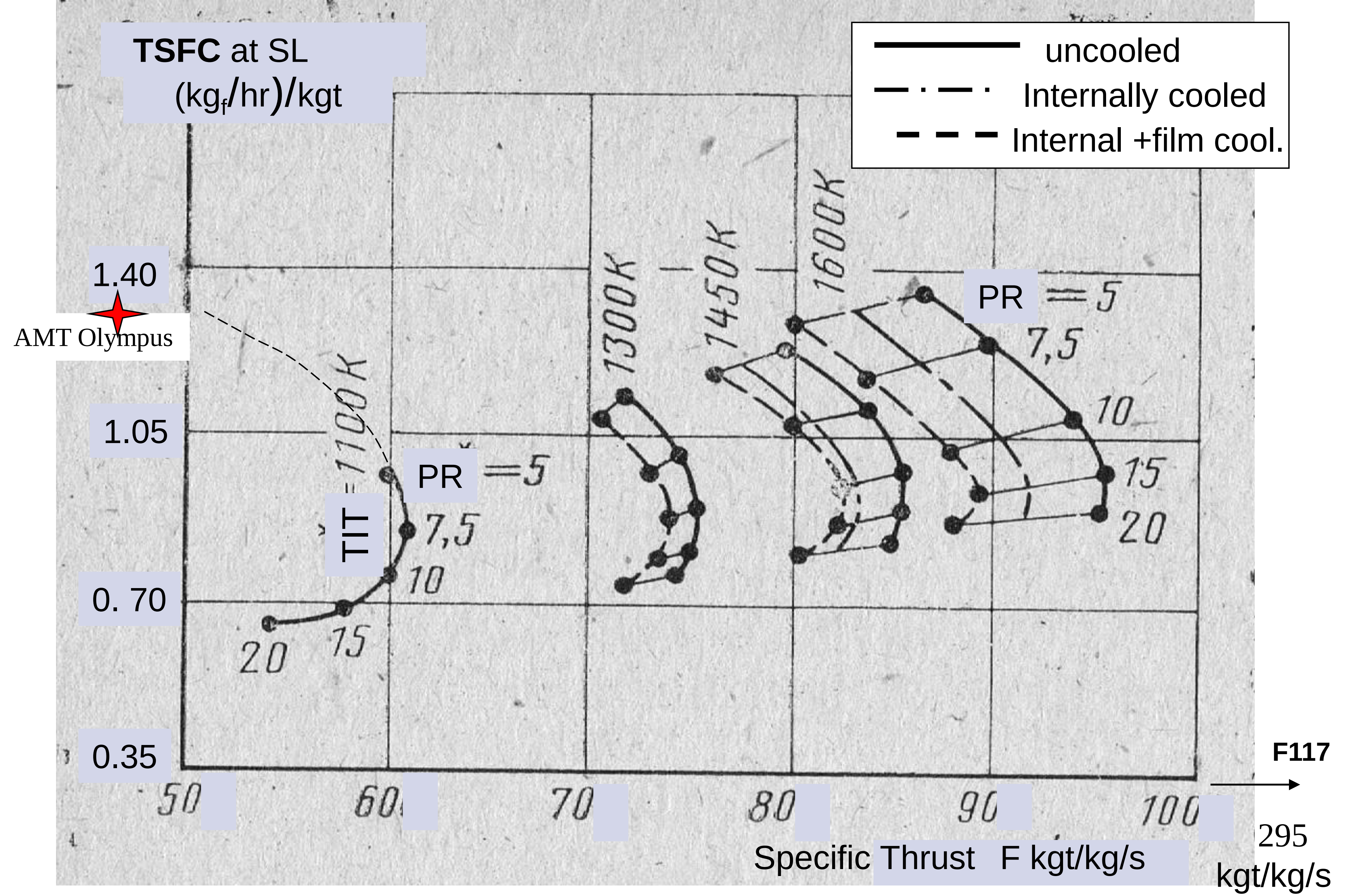
Losses Related To Turbine Cooling

(ref. S.Kopelev “Cooled Gas Turbine Blades- Thermo-aero design”, Moscow, 1983)



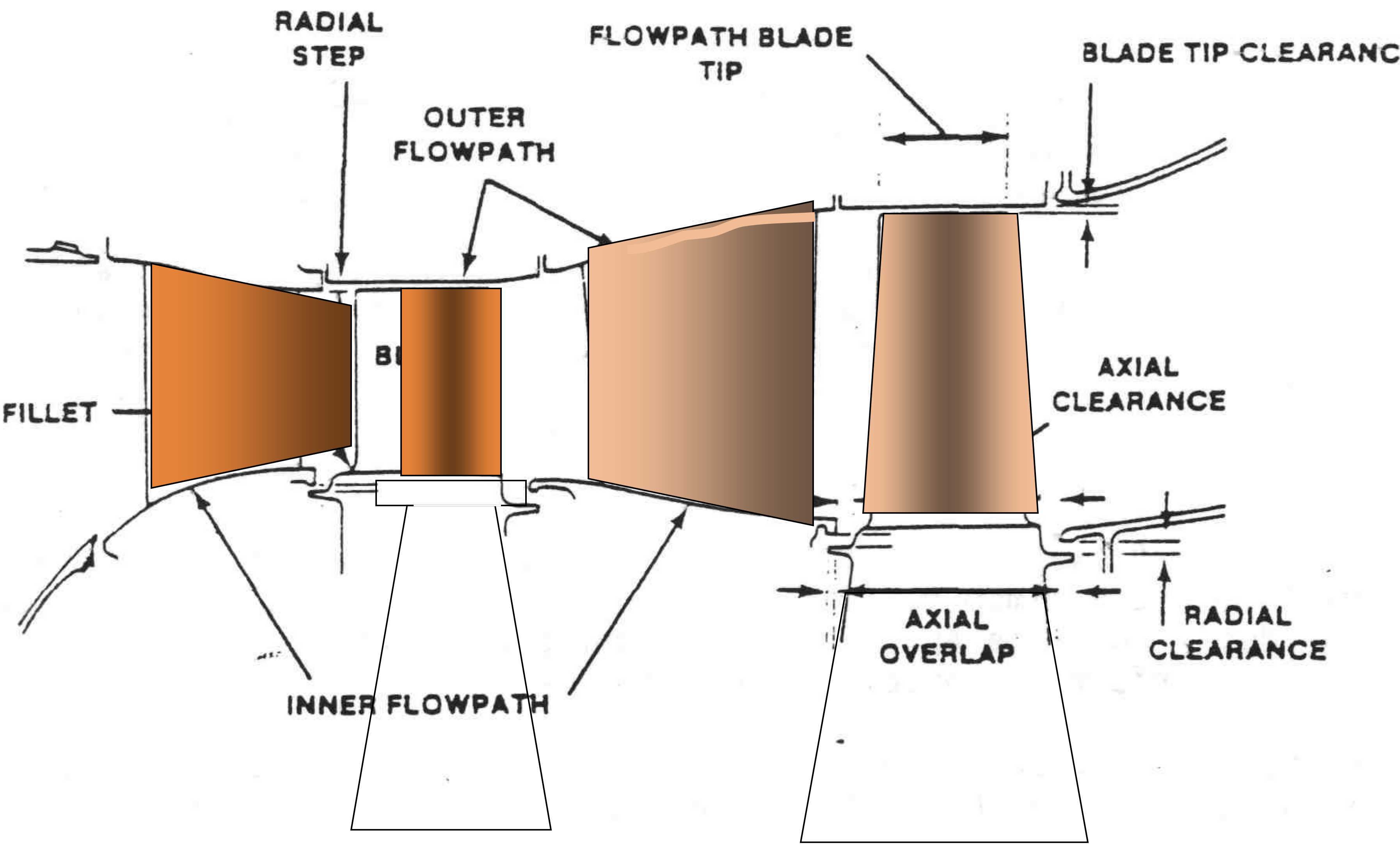
Effect of Cooling on Aero Engine Performance

Ref. “Gas Turbine Engines for Airborne Systems”, Lokai et.al, Moscow, Russia 1991.

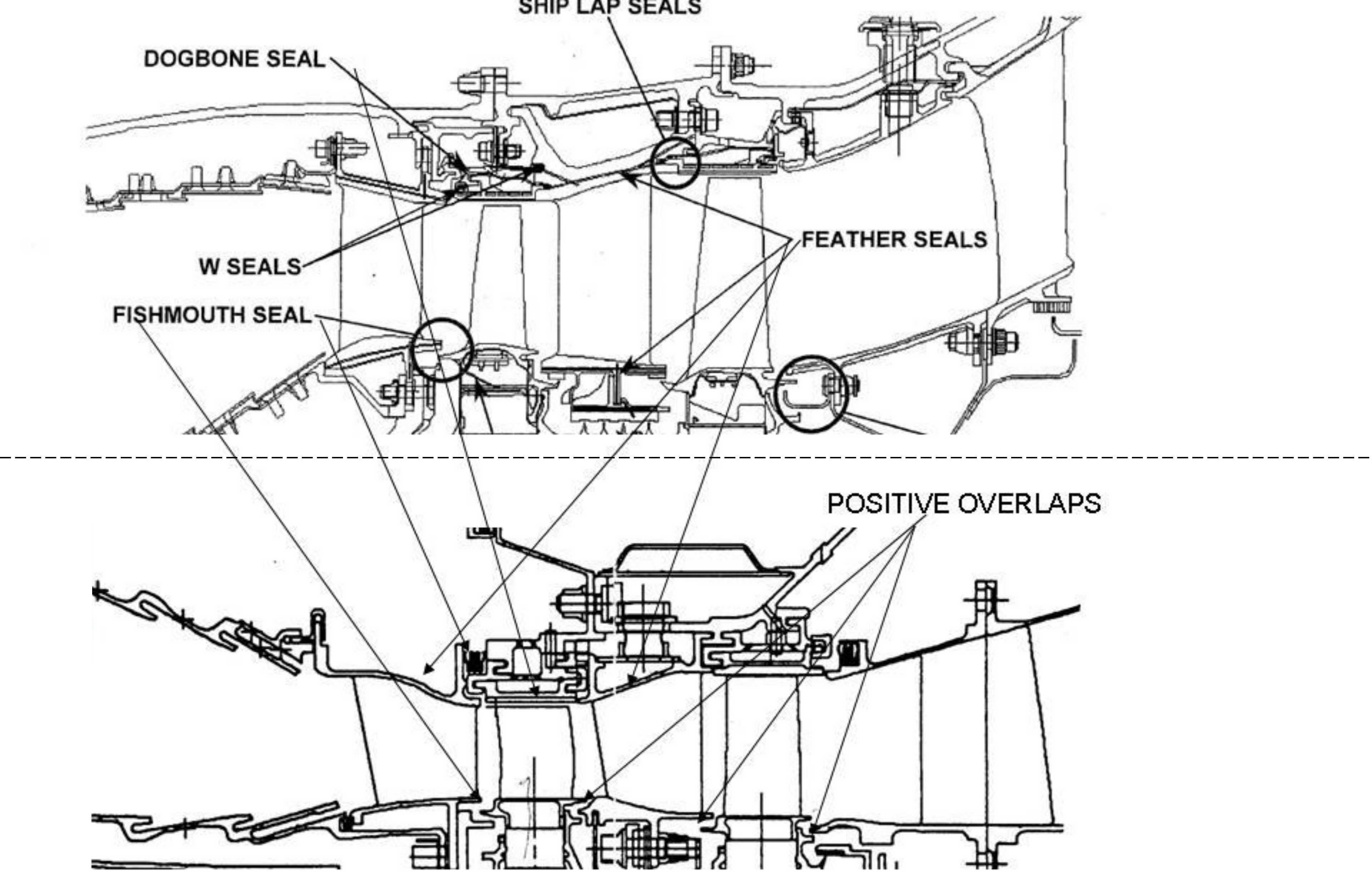


Effect of Cooling on SFC and Specific Thrust

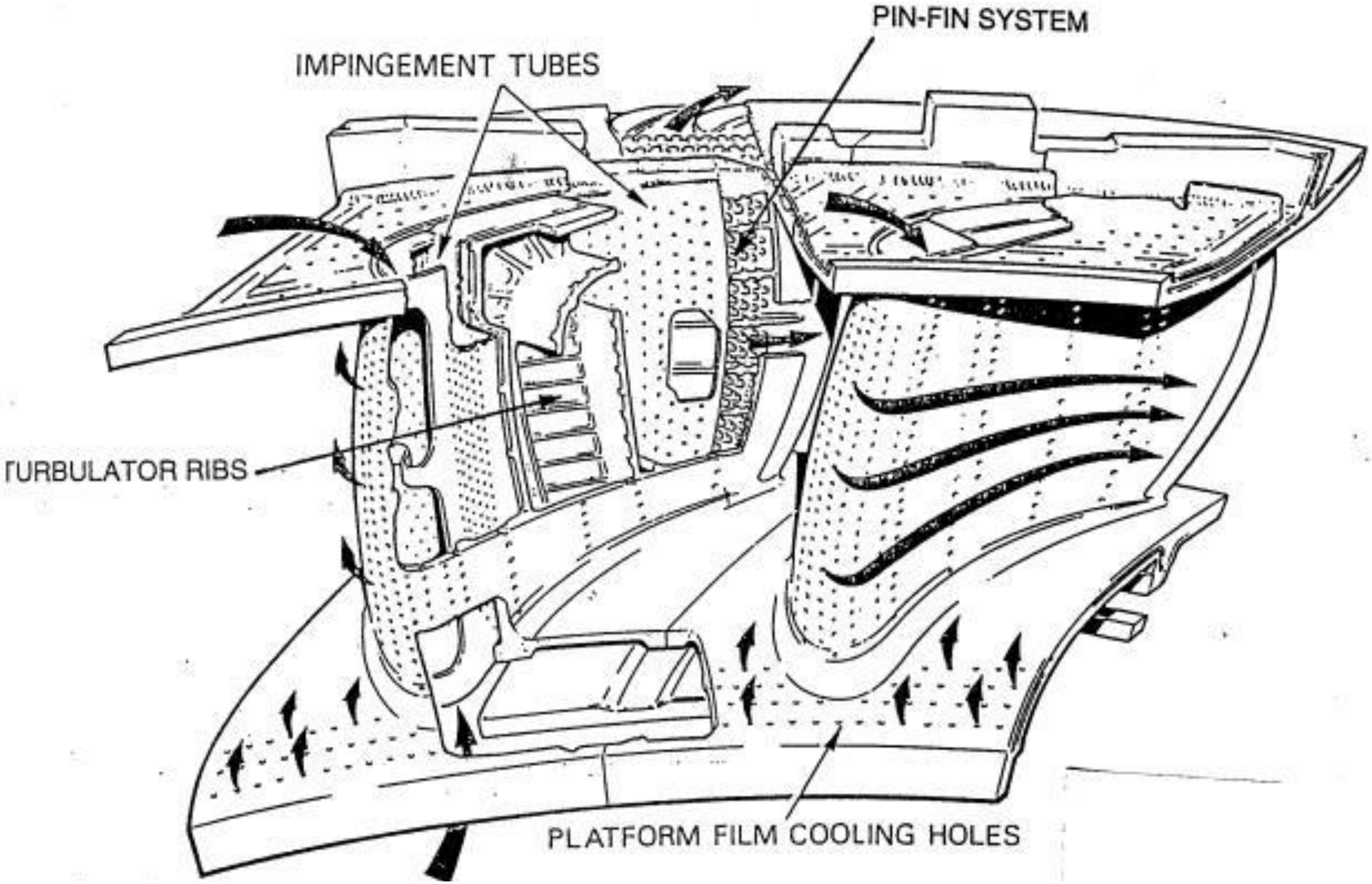
Ref. “Gas Turbine Engines for Airborne Systems”, Lokay et.al, Moscow, Russia 1991.



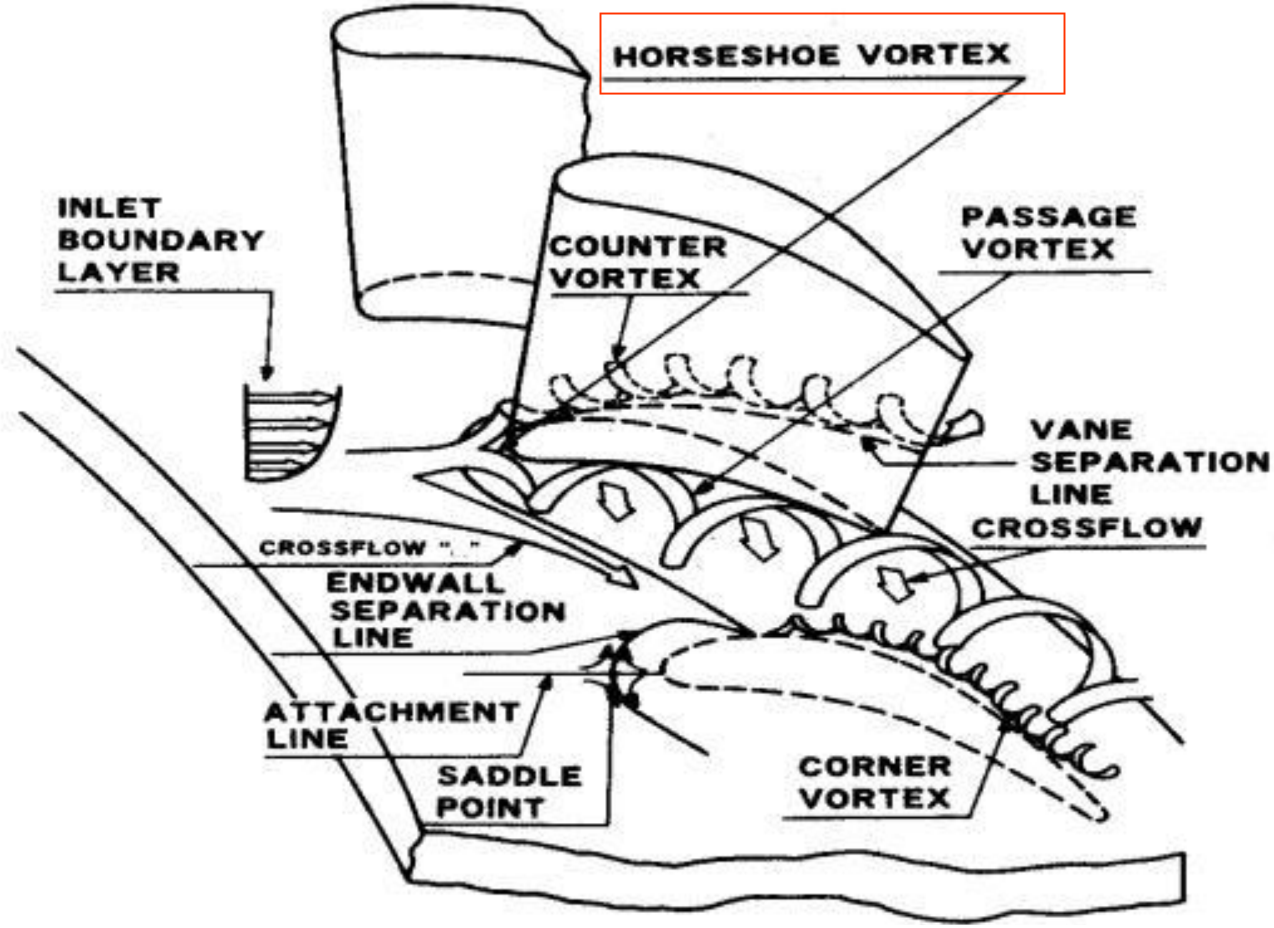
Preferable Flowpath Forming Features



Gaspath Outer and Inner Structure Sealing

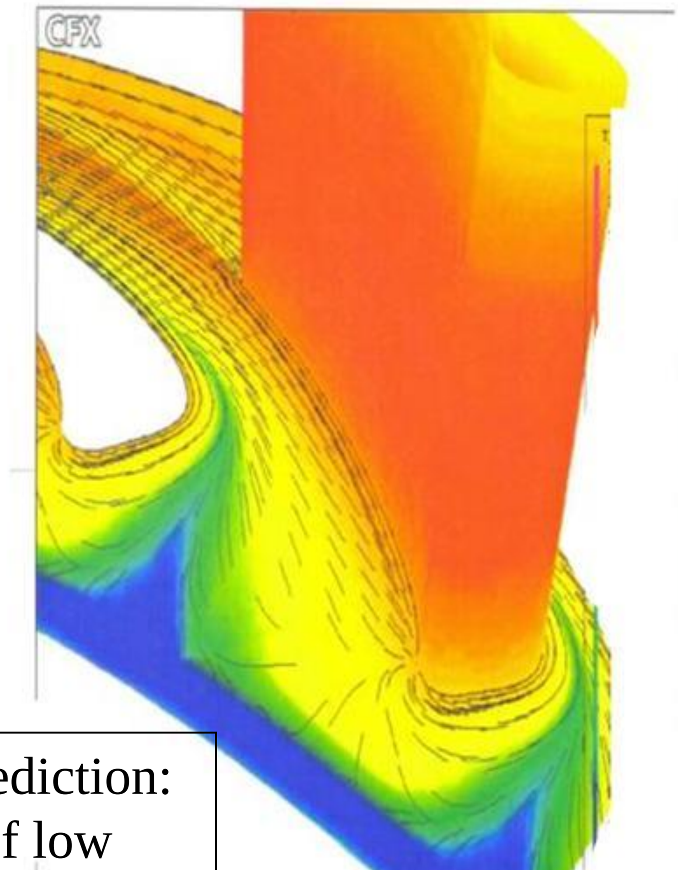


Example Of Advanced Nozzle Design

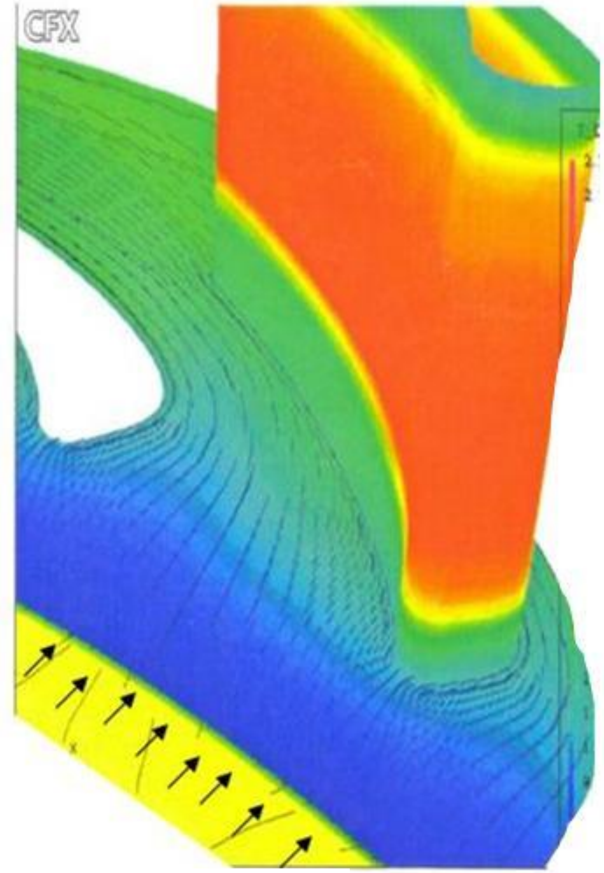


Nozzle Endwall Secondary Flows

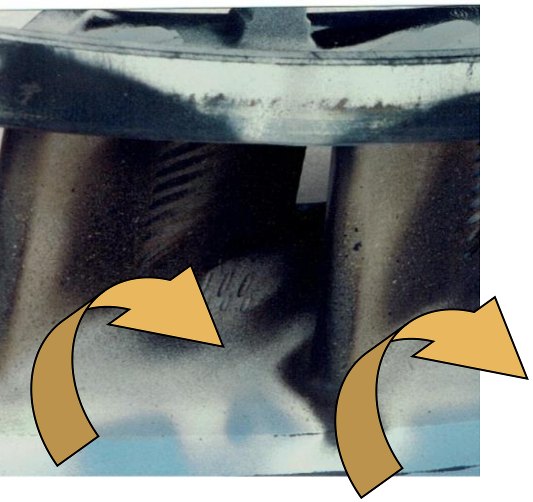
Endwall 0.8% PCD
upstream film flow



Endwall 2% PCD
upstream film flow



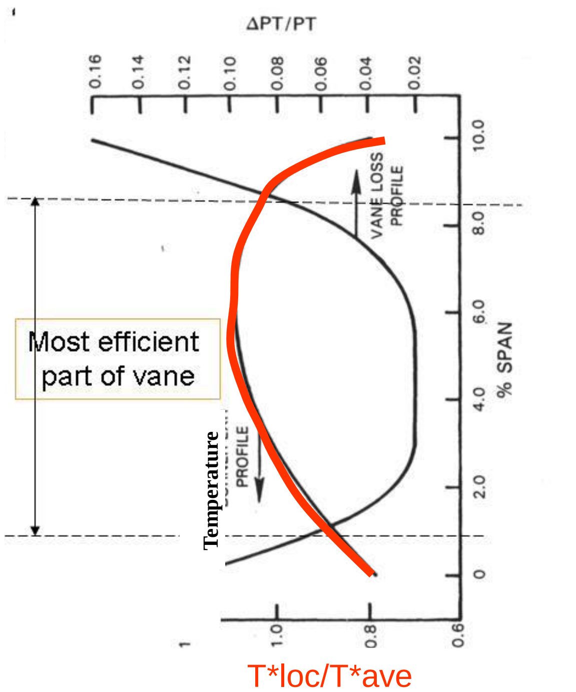
(higher film blowing ratio, steeper angle)



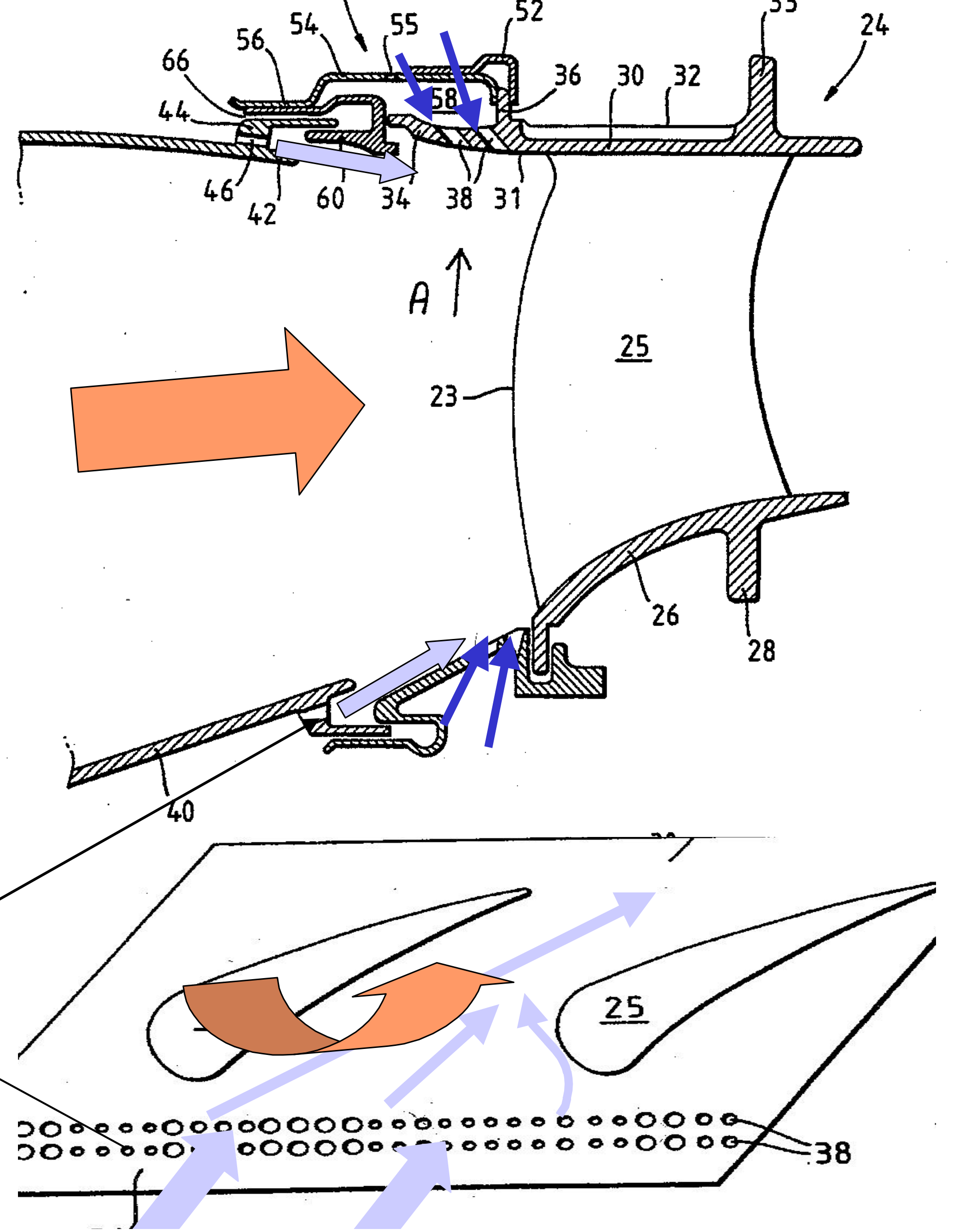
Thermal paint mapping

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

"Suppression" Of Horseshoe Vortex With Upstream Film



Vane Inlet Conditions



Proper introduction of endwall film cooling can provide full film coverage and significantly reduce stage pressure losses suppressing a “horseshoe” vortex

Combustor Transition- Endwall Cooling

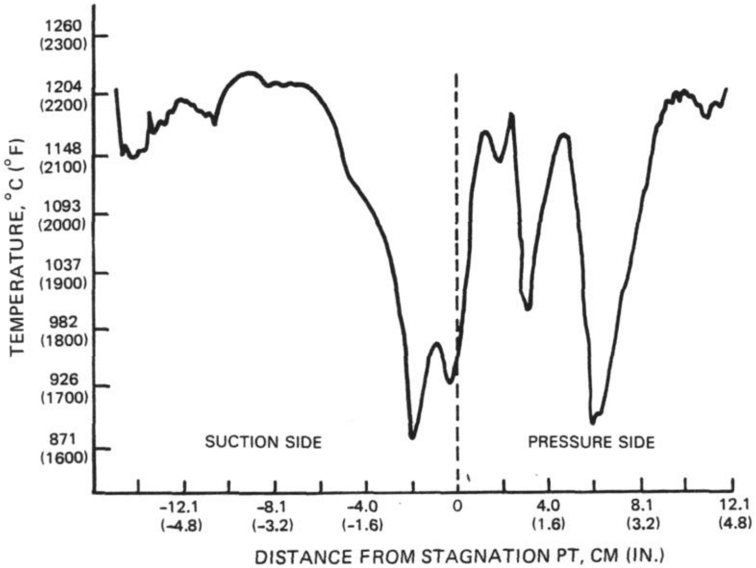
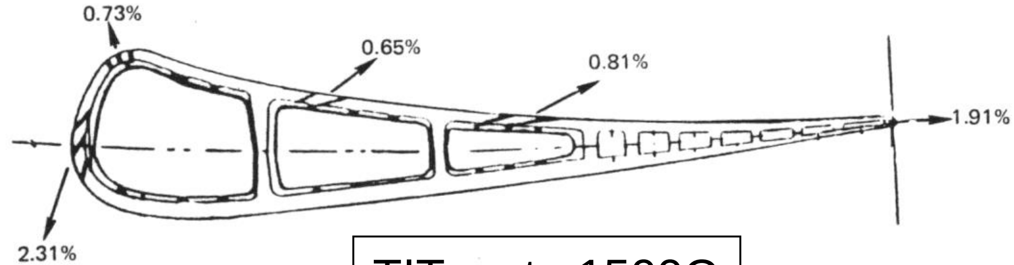
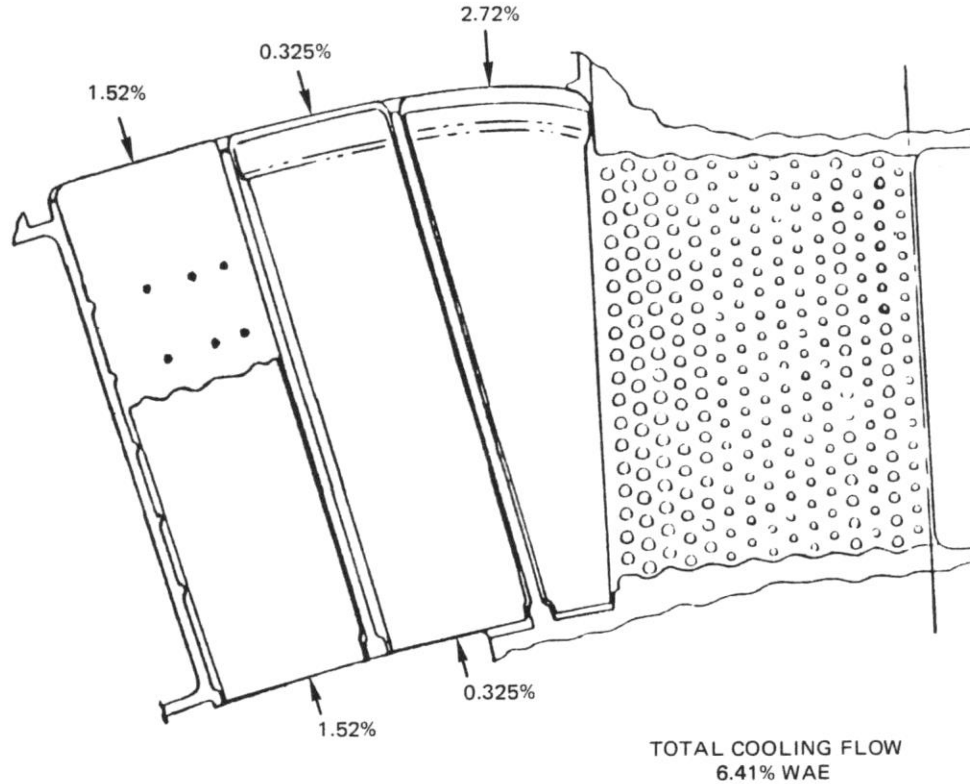
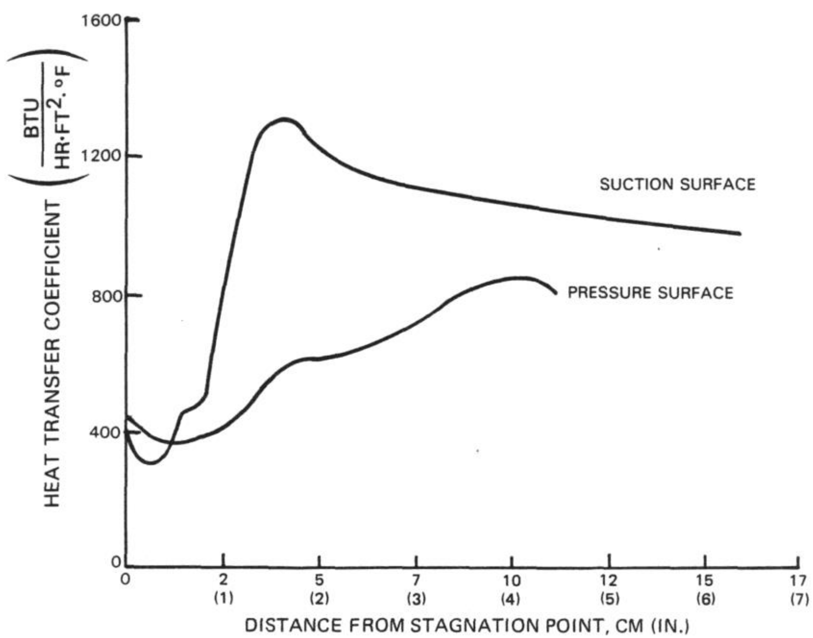


Figure 5.2.1-10 Vane Surface Temperature Profile



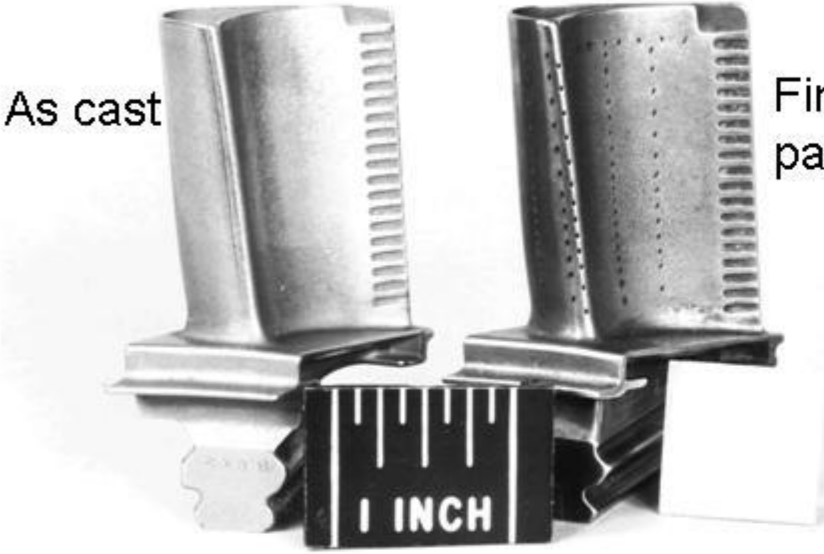
TIT up to 1500C





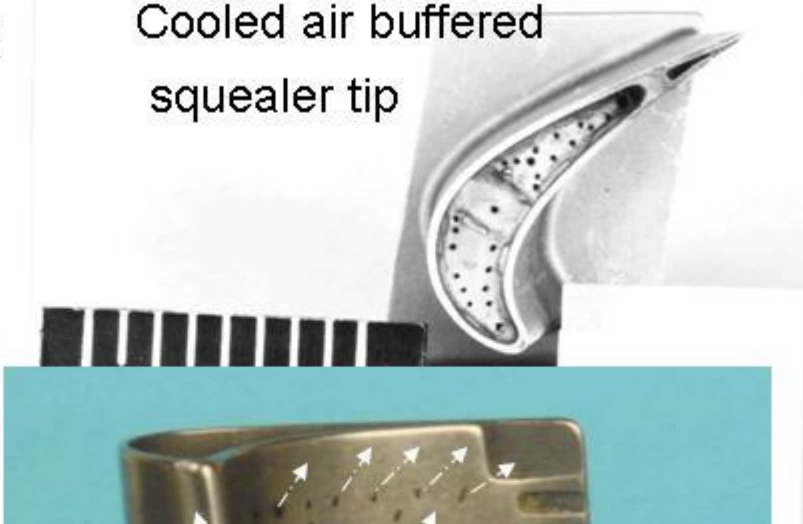
Ref. K.Leach, “E3 engine HP Turbine Component Test Report”, United Techn. P&W, 1983

Nozzle Film Cooling Design for High TIT



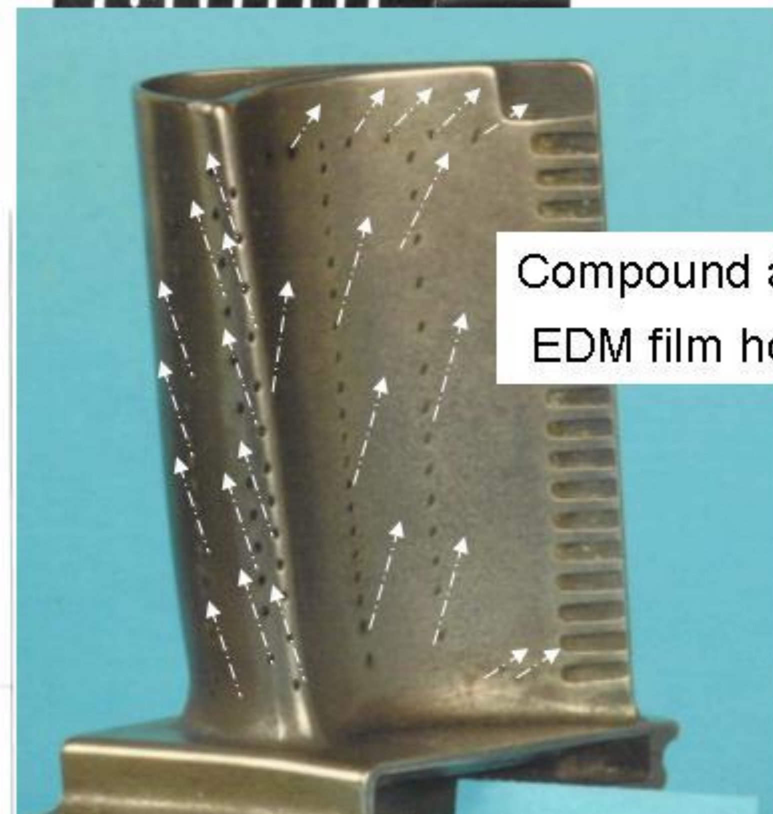
As cast

Finished part



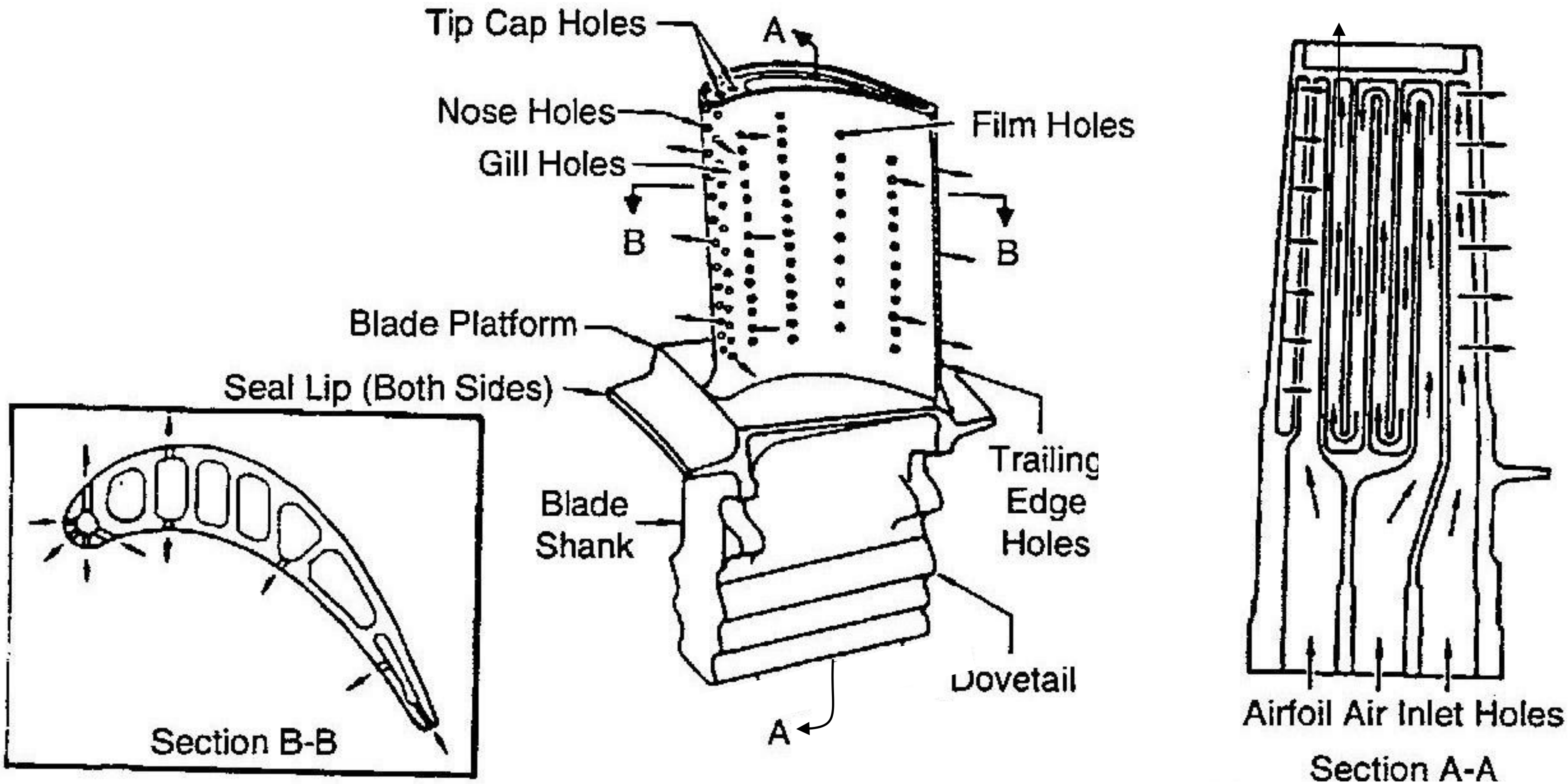
Cooled air buffered squealer tip

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



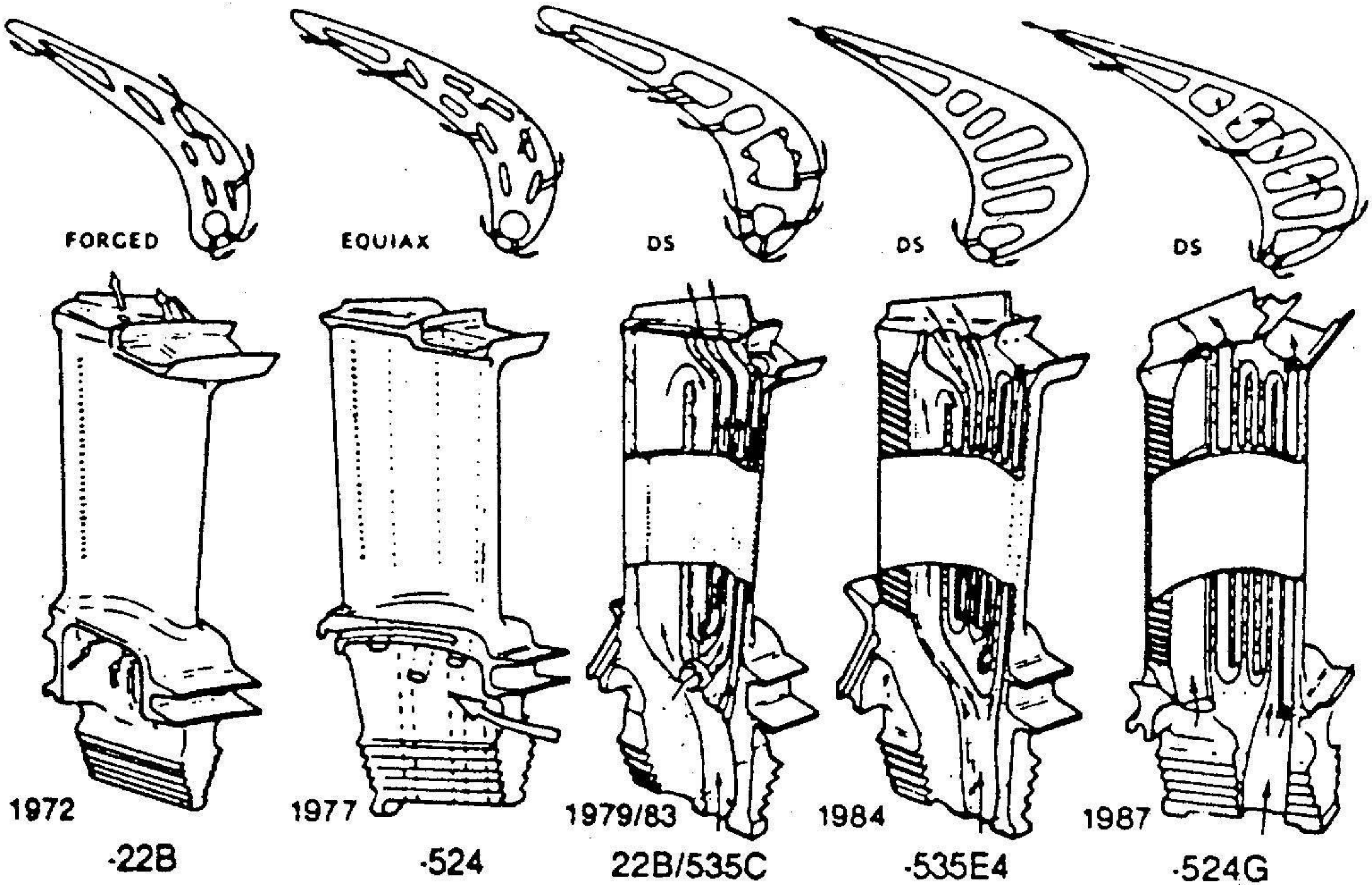
Compound angle EDM film holes

High temperature shroudless blades



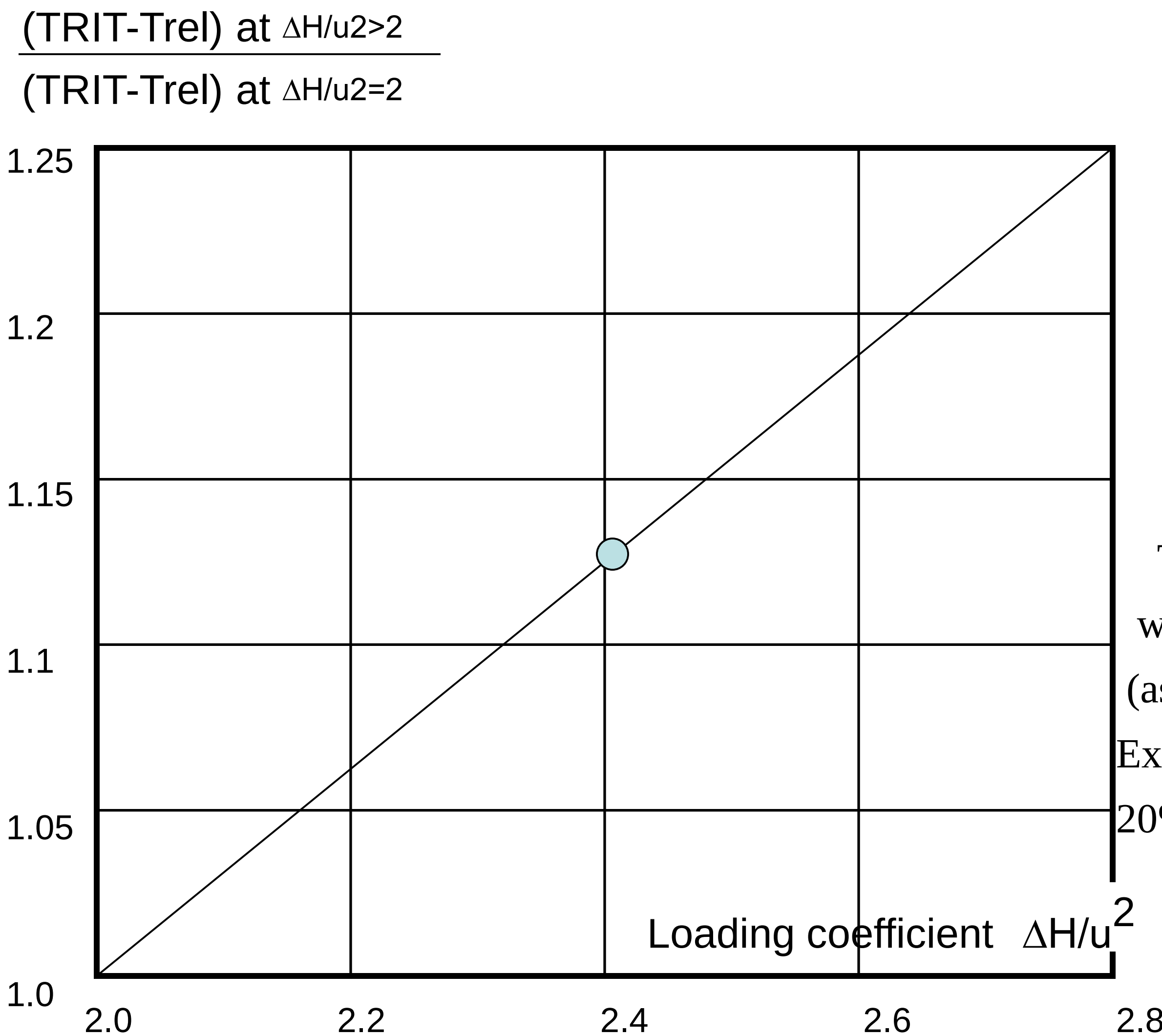
Blade Cooling With Combination of Convective, Impingement and Film

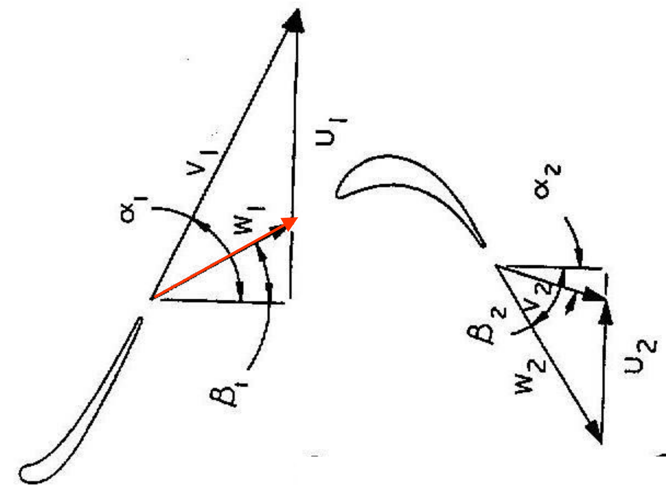
(courtesy of GE)



RB211 St.1 Shrouded Blade Cooling Design Evolution

(courtesy of RR)





$\text{TRIT-Trel} = u^2(\gamma - 1))/2R\gamma$,

where γ-spec. heat ratio and R gas const

(assumed a near 0 reaction at the root)

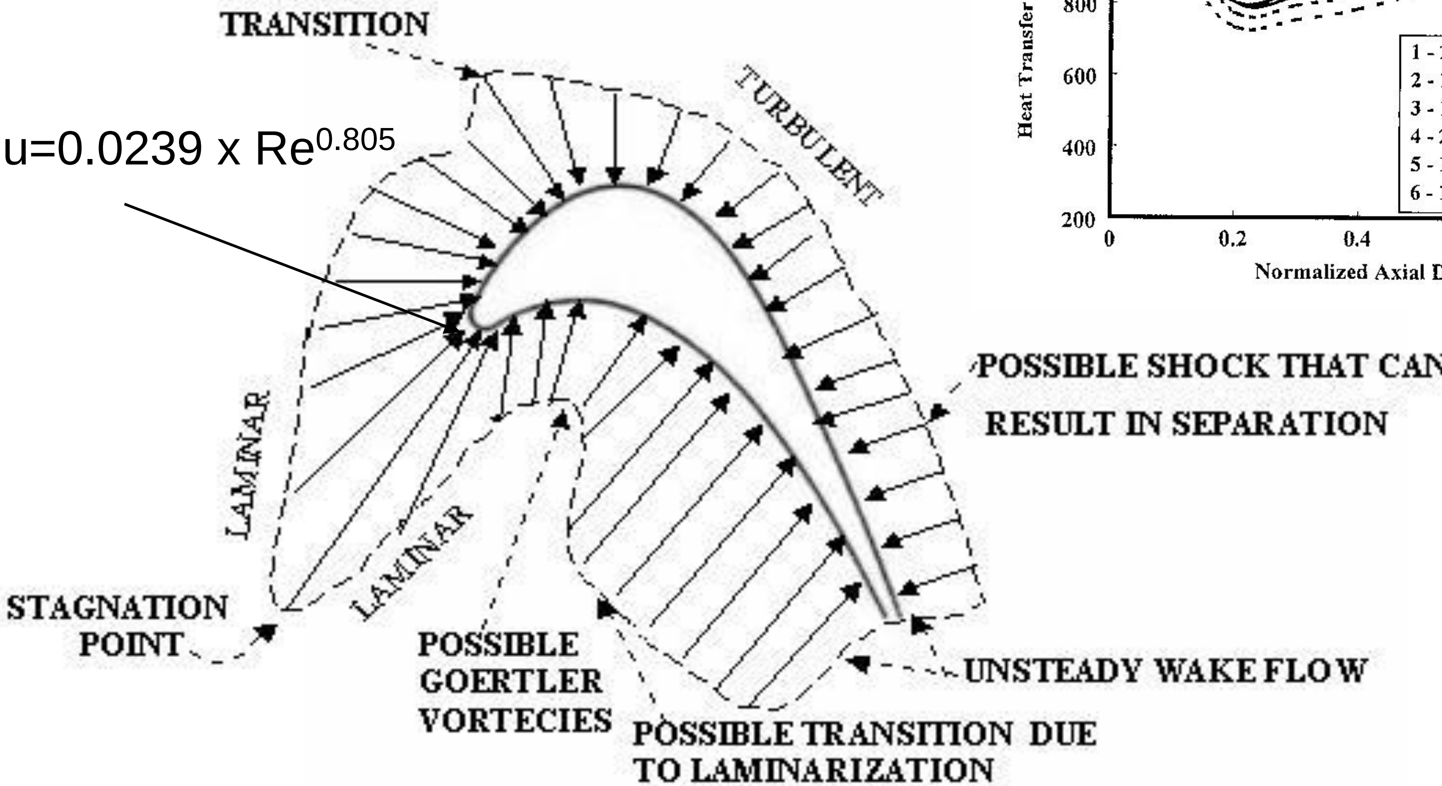
Example:

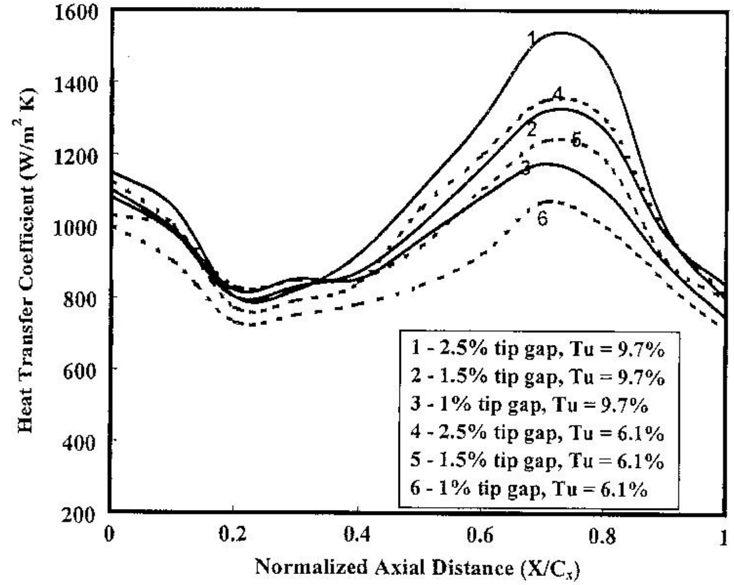
20% increase in load ~13% rise in $\overline{\Delta \text{Trel}}$

Effect of Blade Loading on TRIT_{Rel}

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

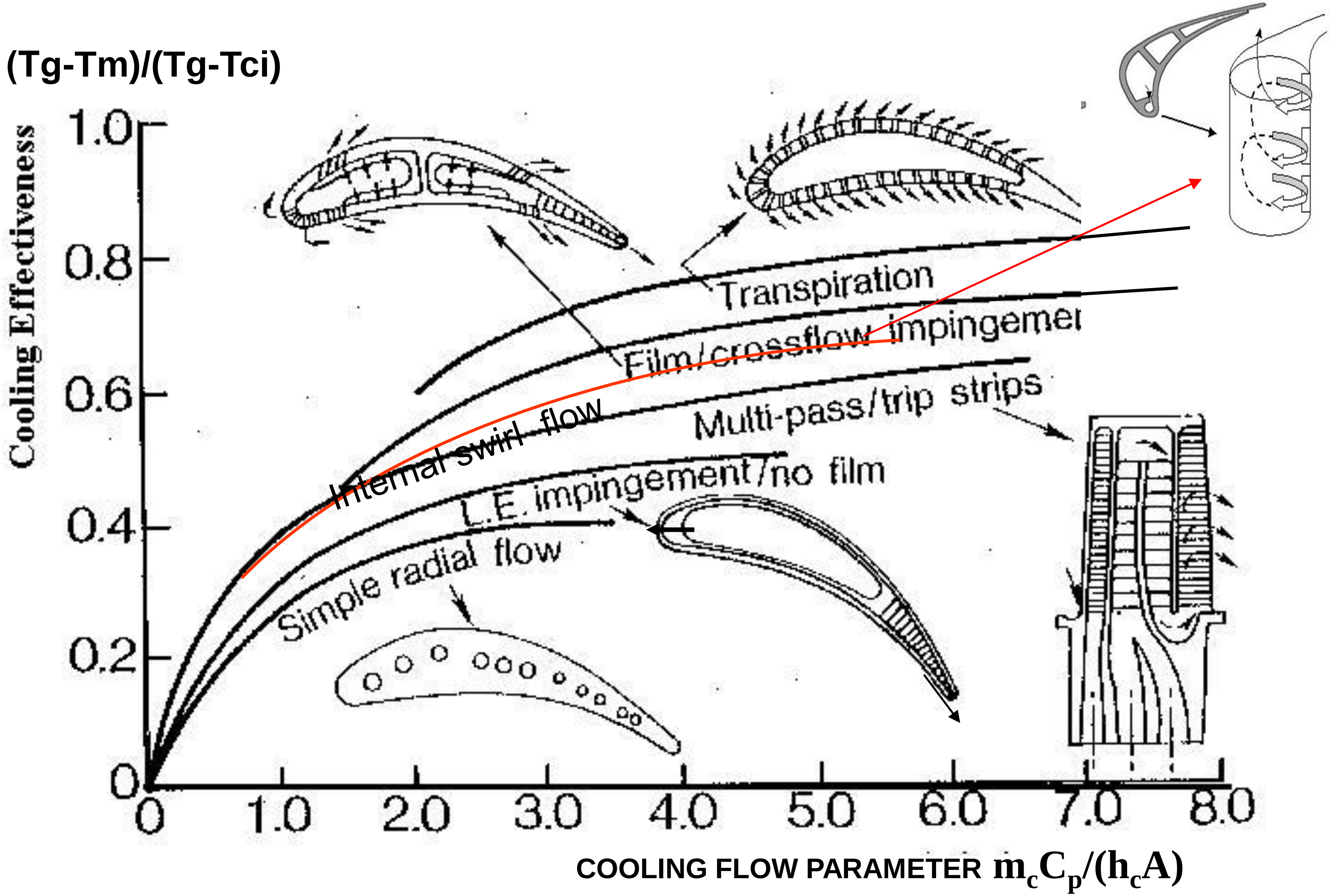
$$Nu=0.0239 \times Re^{0.805}$$



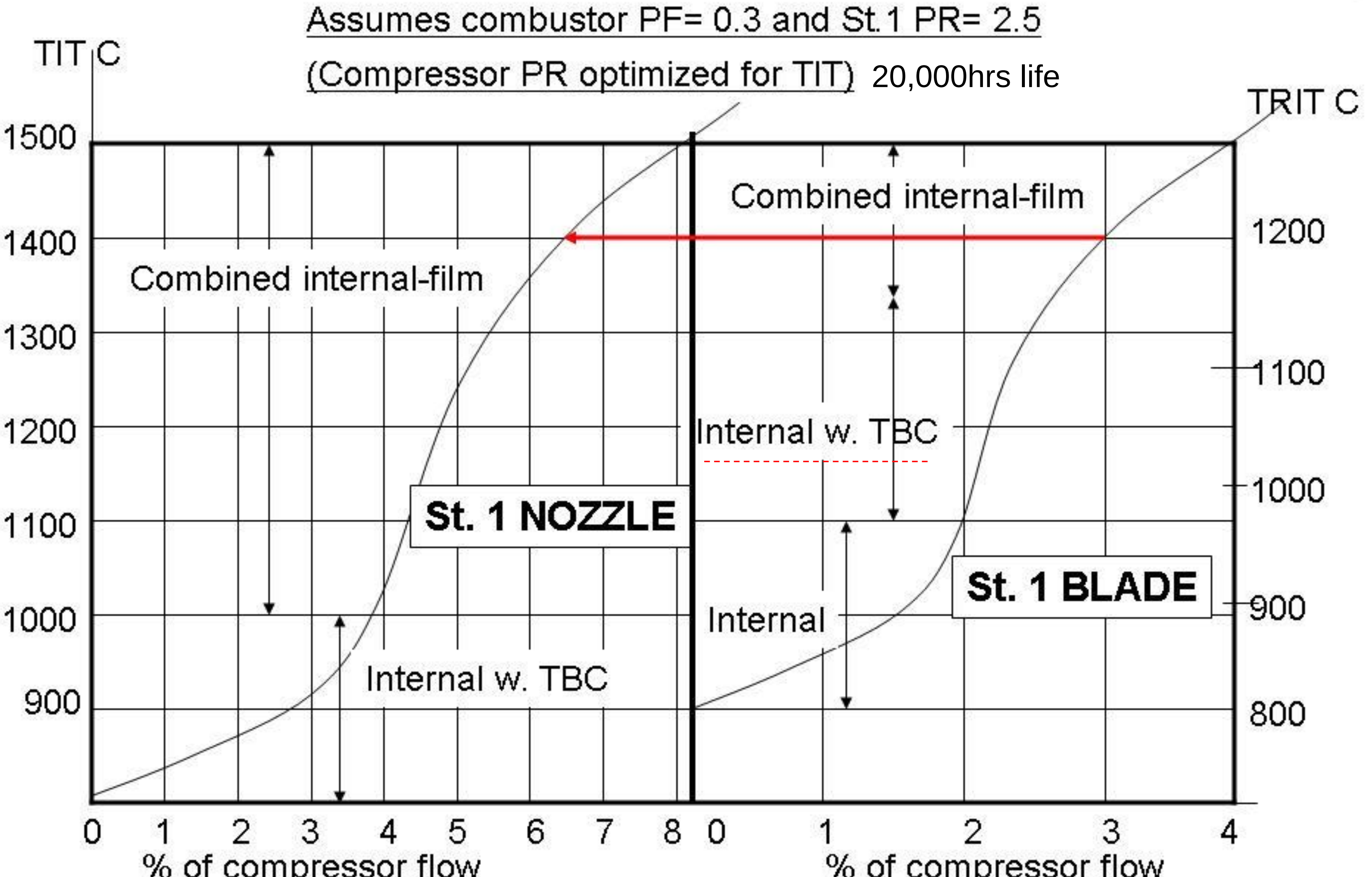


Airfoil External Thermal Boundary Conditions

$\varepsilon = (Tg-Tm)/(Tg-Tci)$

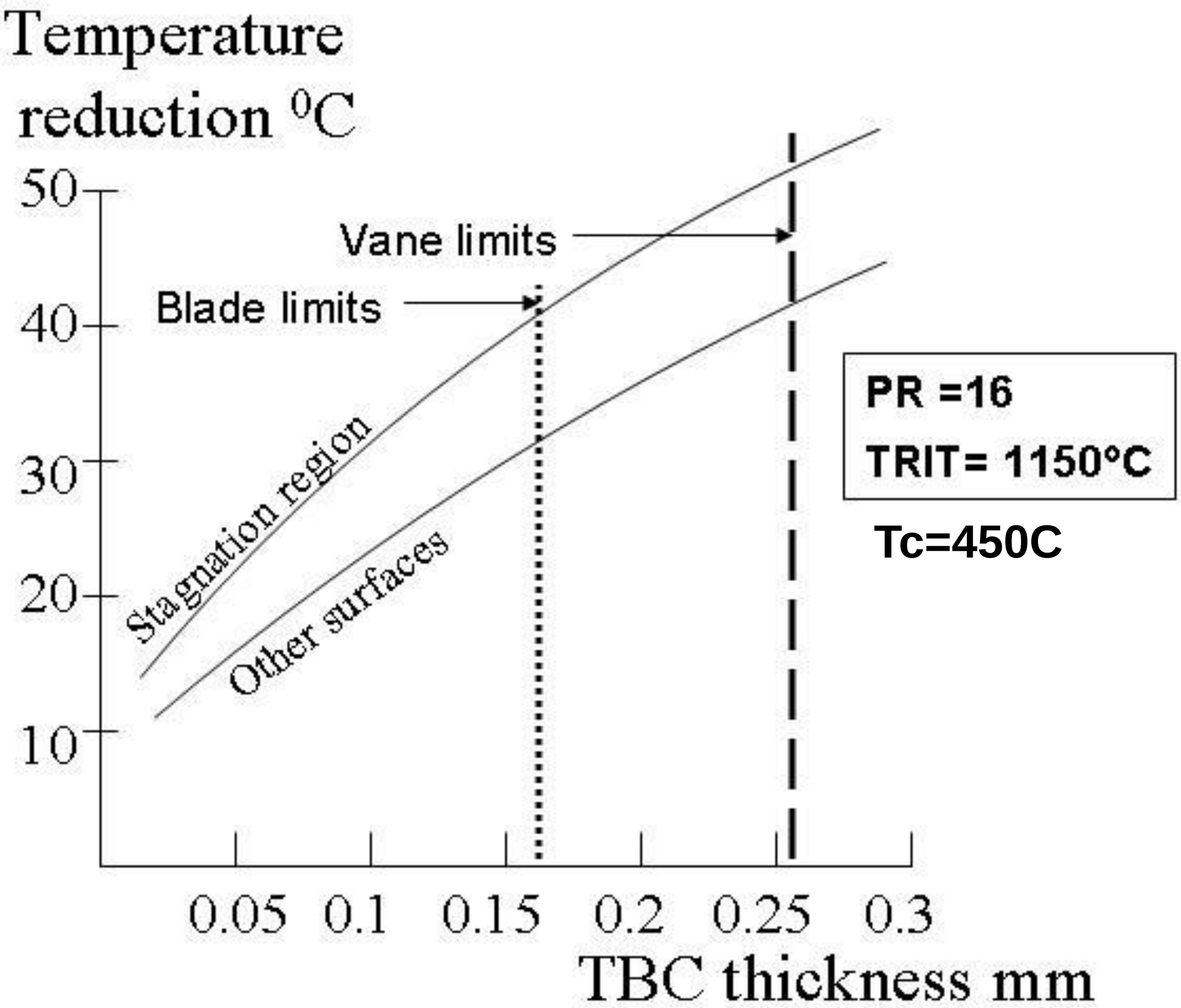


Required Airfoil Cooling Effectiveness



Approximate Guidance for Selection of Cooling Techniques for St.1 GP Turbine

Before film holes are considered



Few recommended coatings:

TBC Yttrium stabilized zirconium (YTZ) - **airfoils**

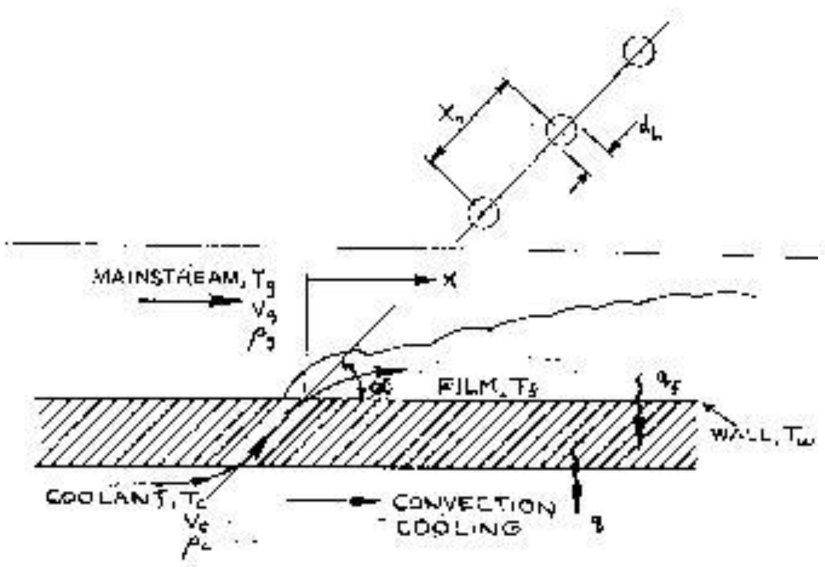
PWA 270 (NiCoCrAlY) – oxidation-erosion resistant coating- **vanes**

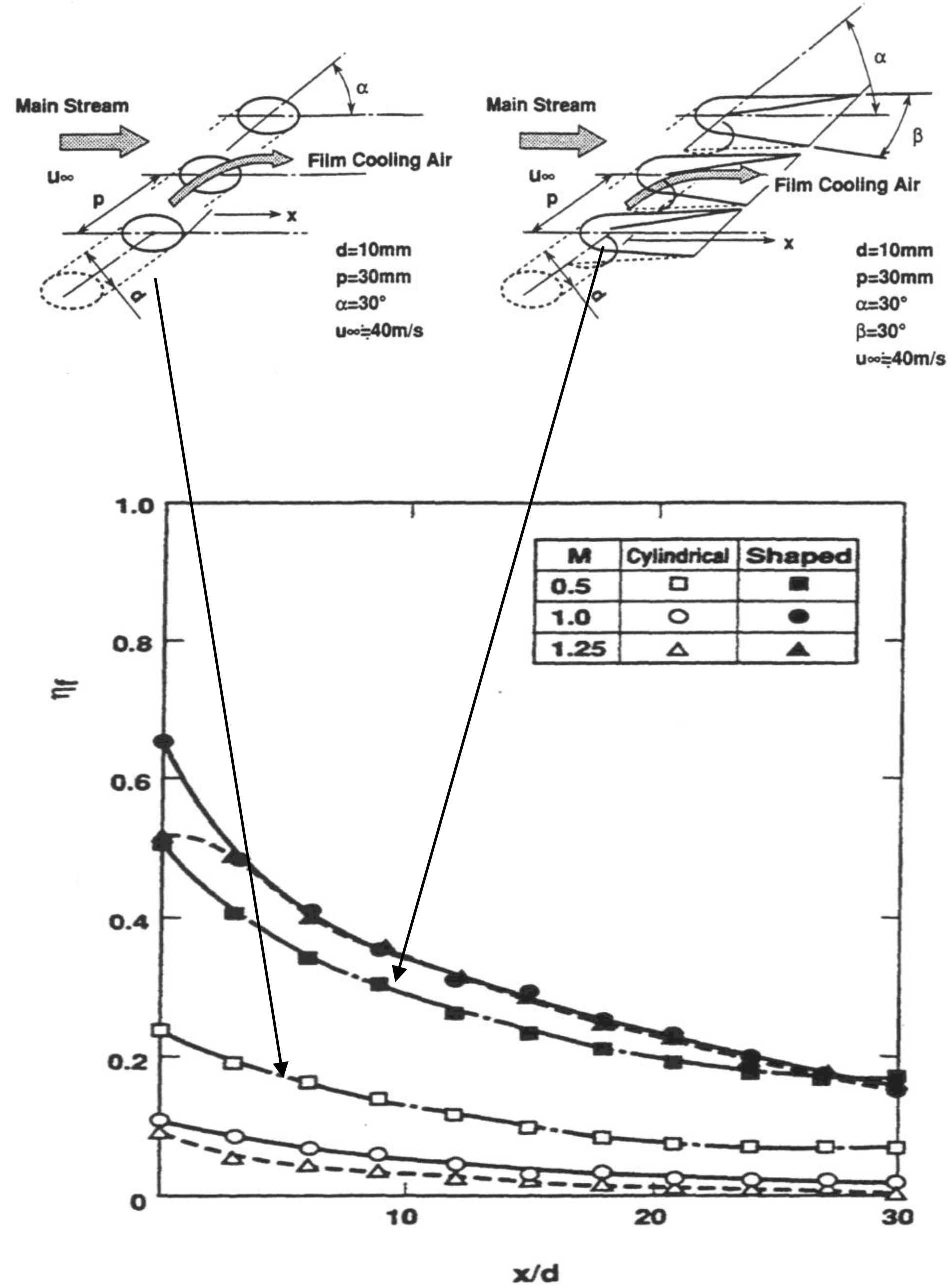
PWA 264 –TBC for endwalls

TBC 100 for **platforms**

PWA 275 – aluminide coat. for **internal vane cavity**

Approximate Effect of TBC (YTZ) on Metal Temperature of Effectively Cooled Airfoil

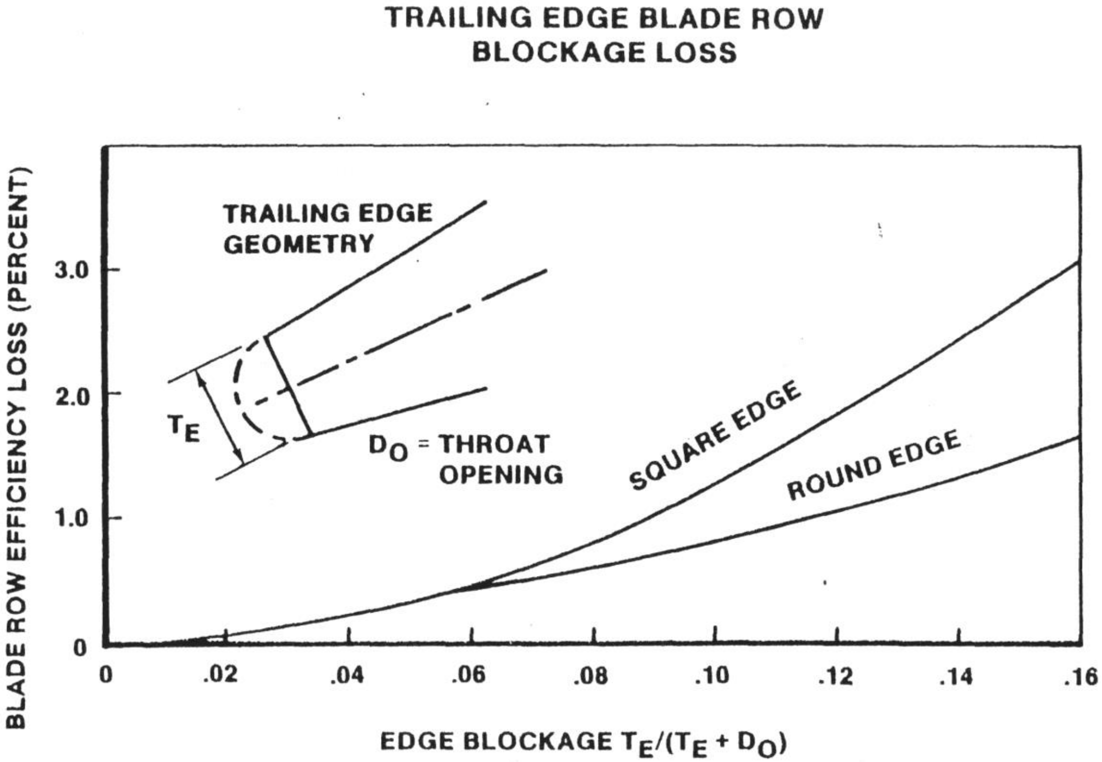


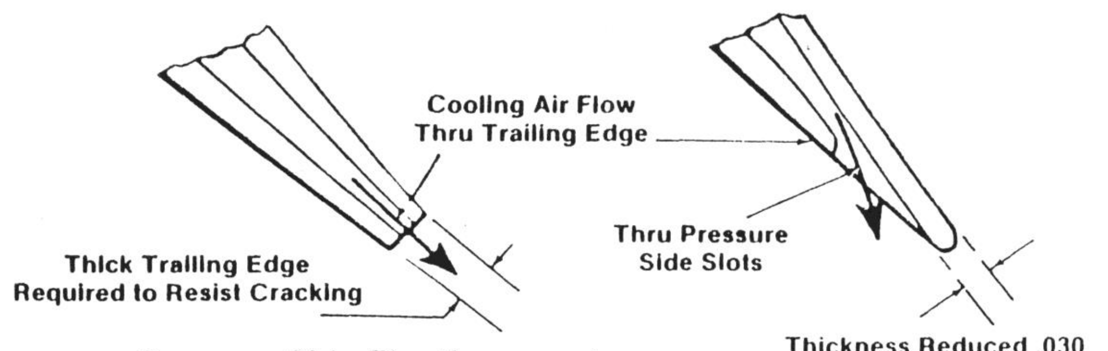


Effect of Film Cooling Hole Shaping

- **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

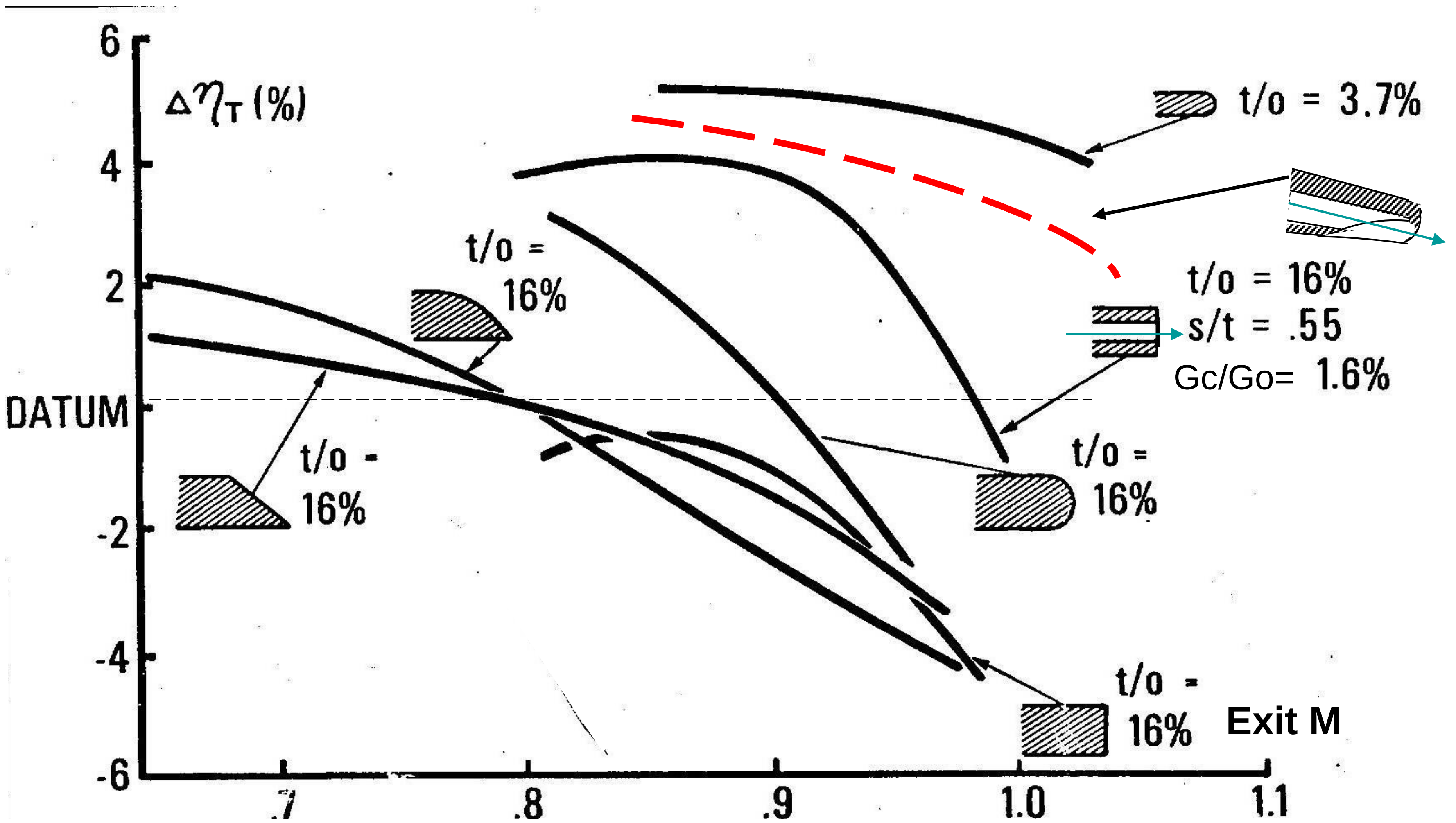




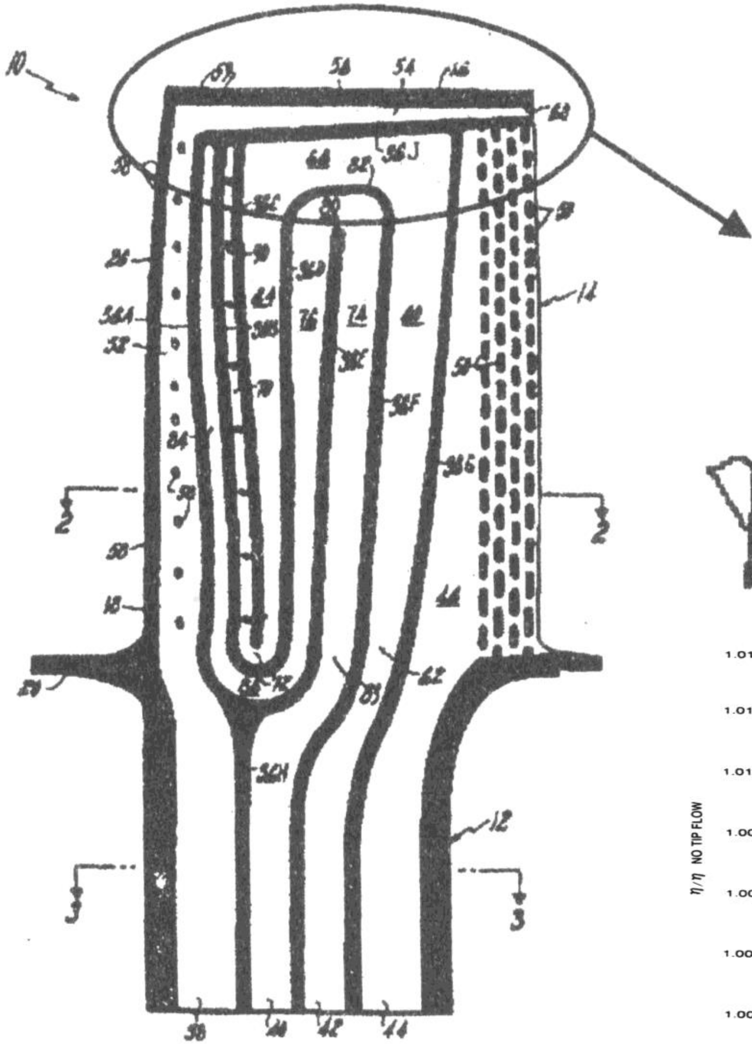
Pressure Side Ejection

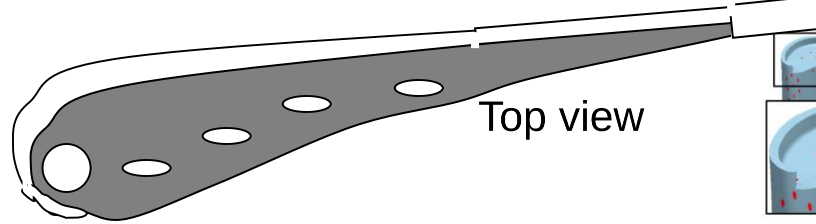
- Requires Increased Cooling Flow
- Achieves Increased Turbine Efficiency
- Net Benefit to the Engine - Reduced SFC = 0.3%

Stage Efficiency Loss Due to TE Blockage



Effect of Vane Trailing Edge Thickness on Efficiency of Turbine Stage (based on R-R studies)





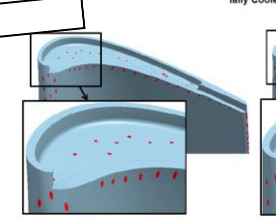
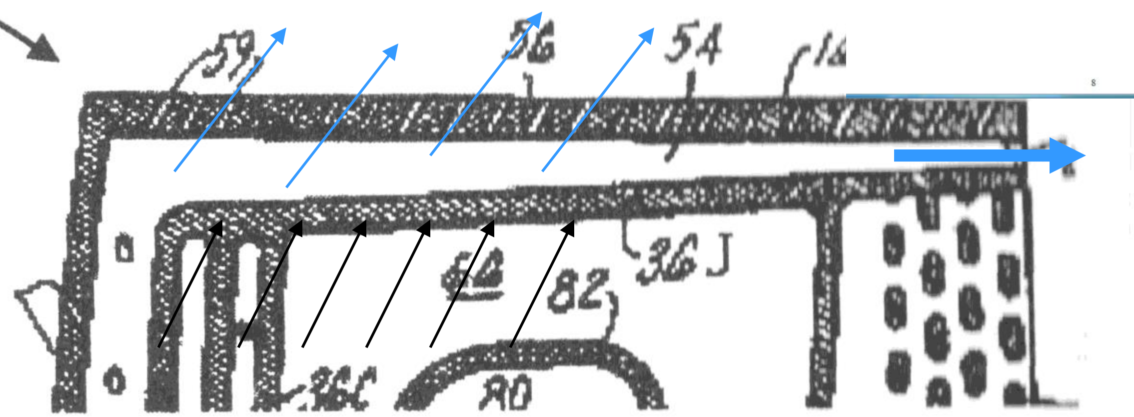
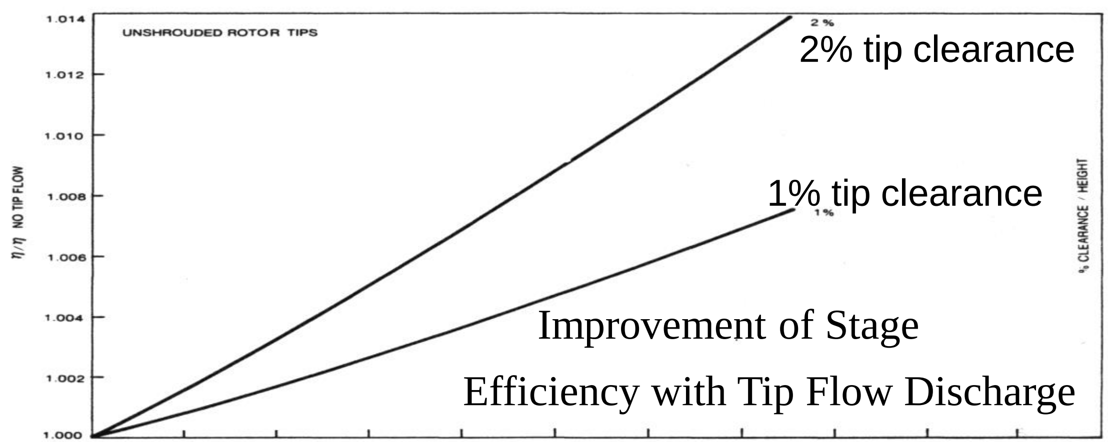


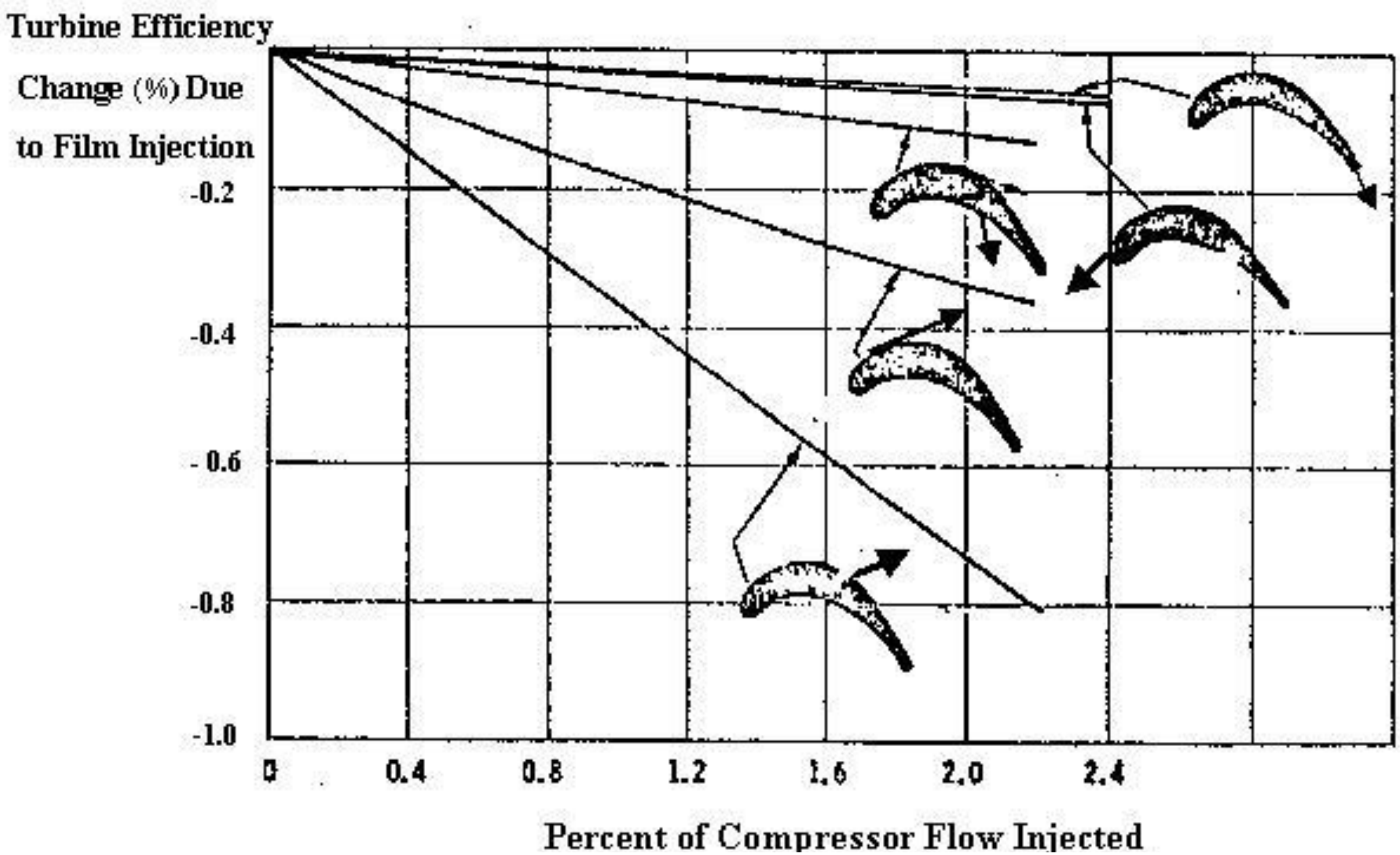
Figure 3. Blade Tip detail: No P/S squealer





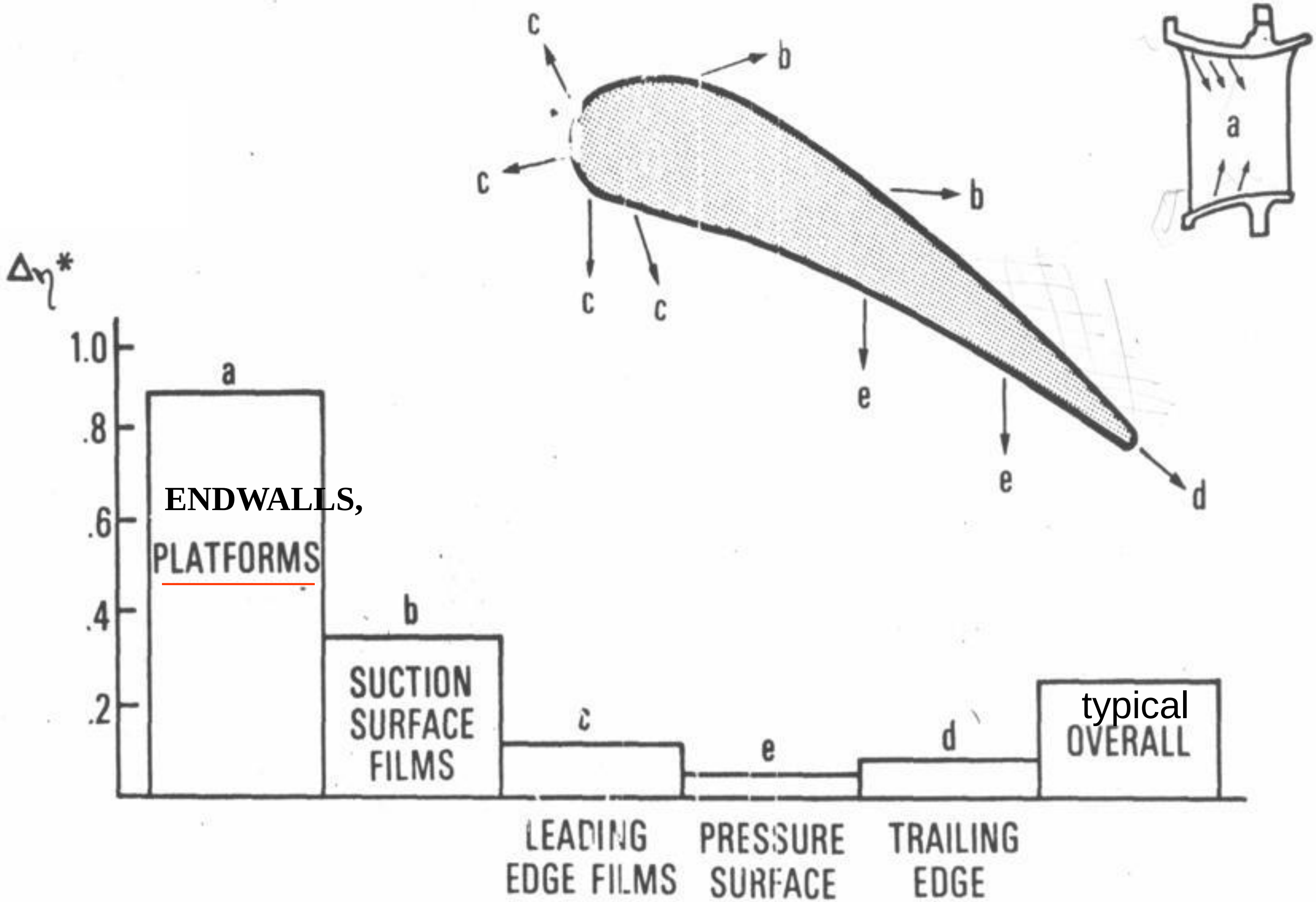
Utilization of LE Spent Cooling Flow for Tip "Flag" Cooling

Loss estimate: $P/P_\infty = \gamma\, m_c/m_\infty Ma^2/2\ (1+Tc/T\infty - 2Vc/V\infty \cos\alpha)$
Ref. ***Hartsel***, J. E., 1972,



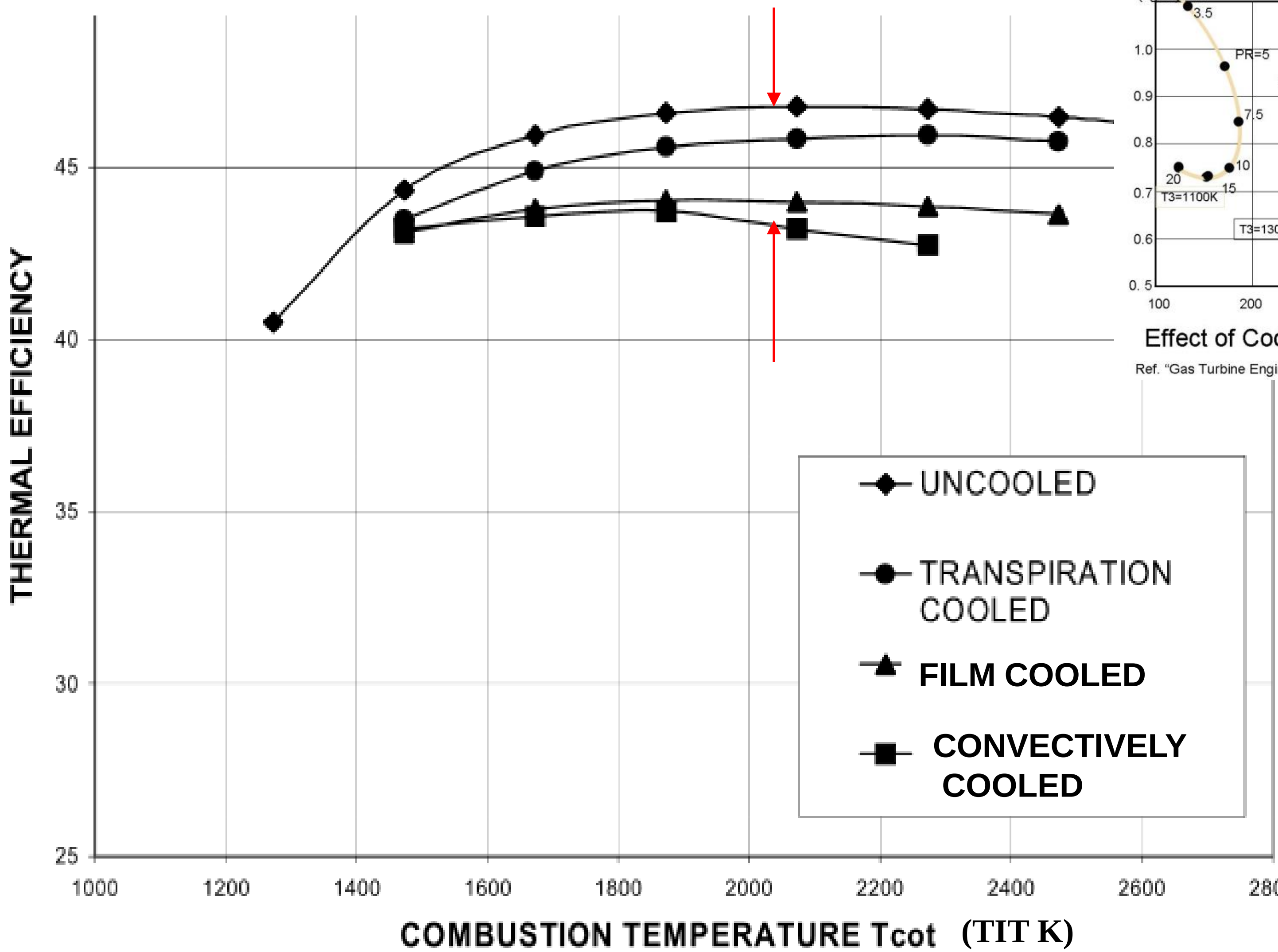
Film Related Turbine Efficiency Losses

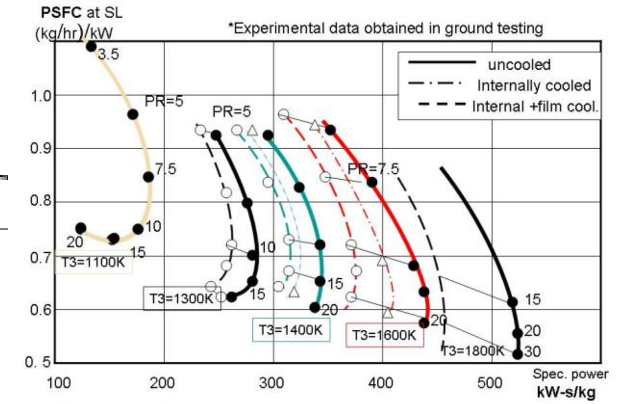
(Ref. B.Barry, 1976, von Karman LS 83)



Nozzle Film Cooling Discharge Penalties

(Ref. B.Barry, 1976, von Karman LS 83)



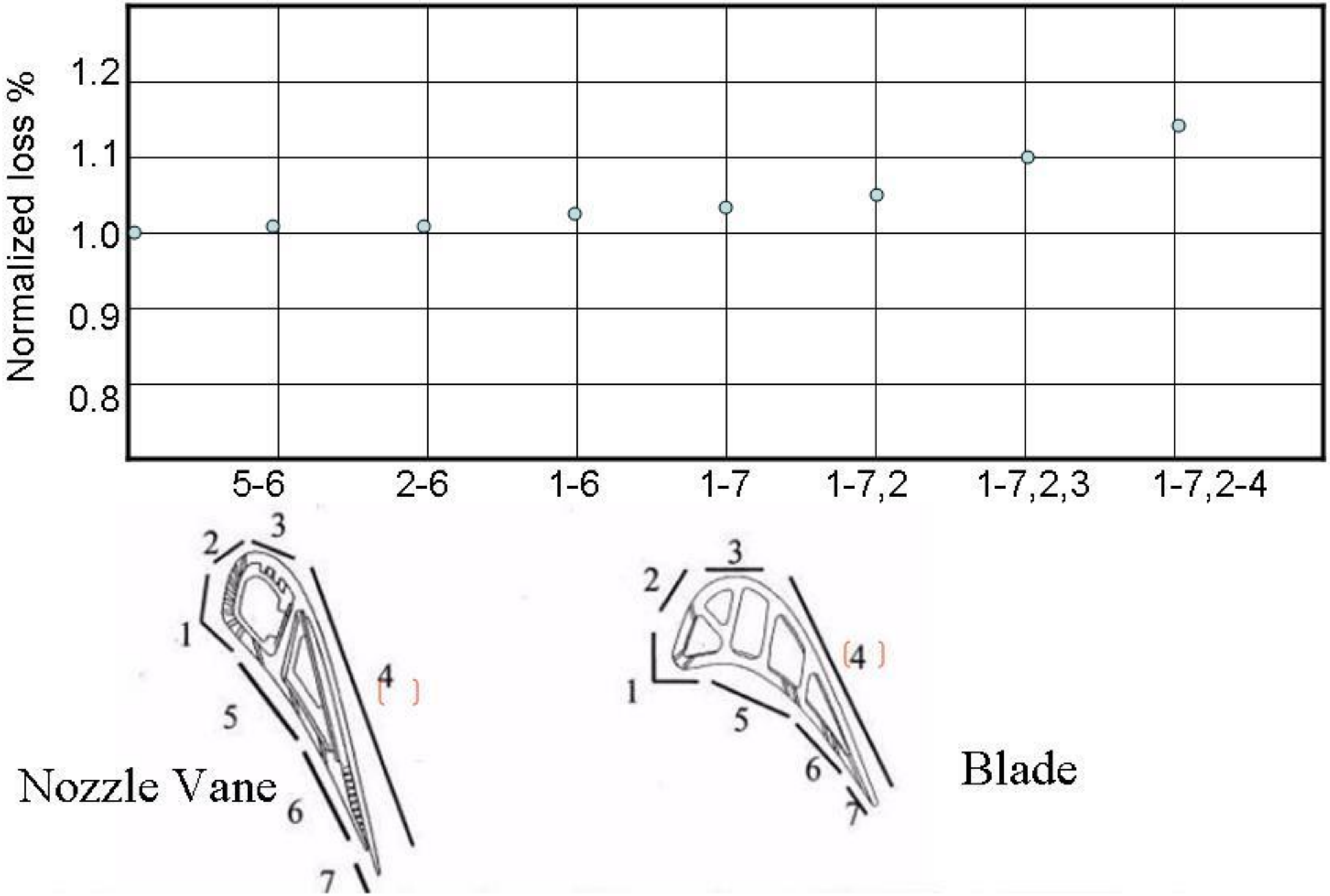


Effect of Cooling on Aero Engine Performance

Ref. "Gas Turbine Engines for Airborne Systems", Lokai et.al, Moscow, Russia 1991.

Thermal Efficiency as a Ψ(TIT) for Different Cooling Techniques

Ref. "Limitation on GT Performance Imposed by Large Cooling Flows" J. Harlok, et.al IGTI 2000-635

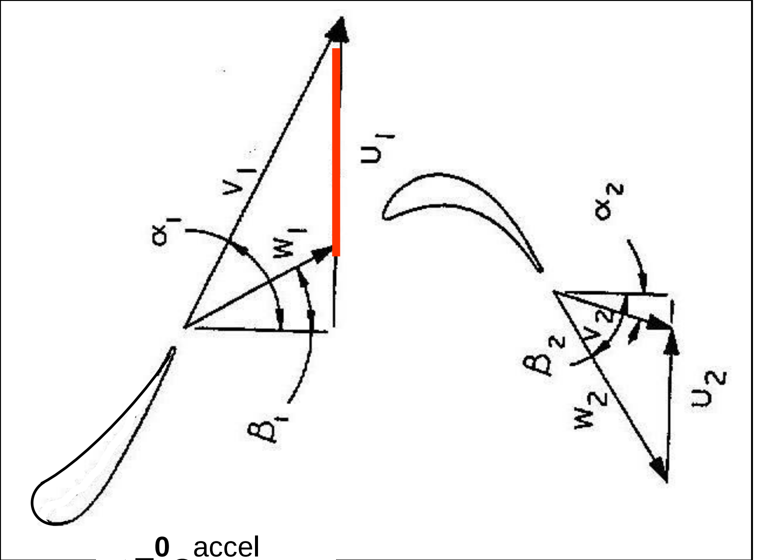


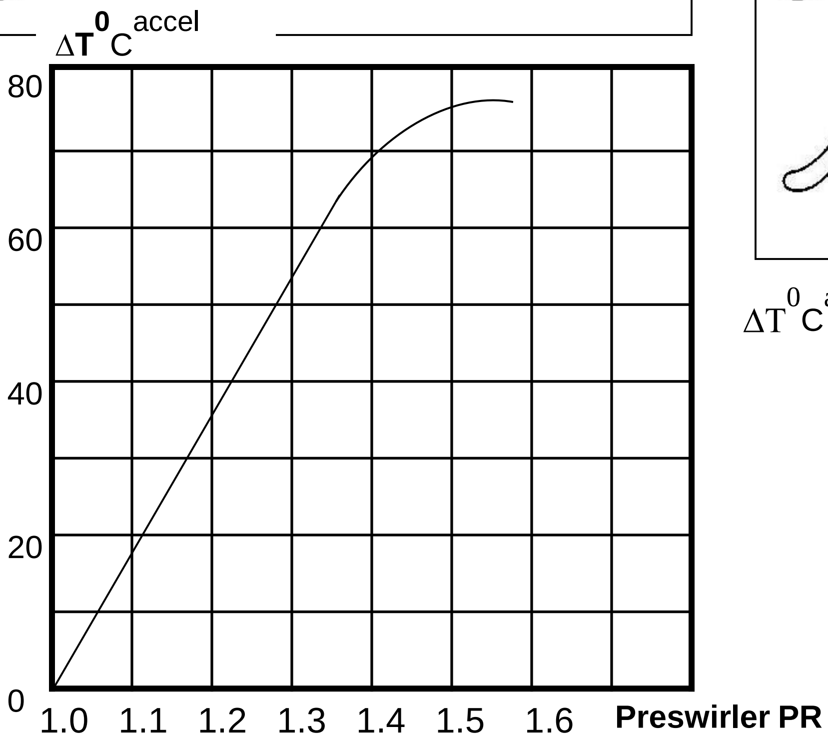
Typical Airfoil Cooling Penalties

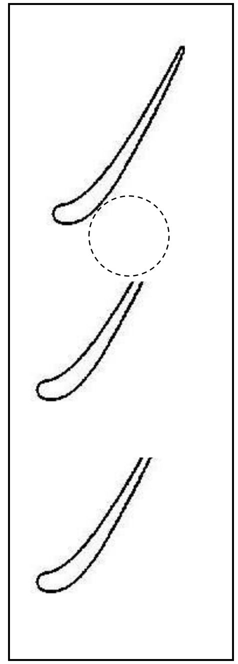
"Aero losses from film injection" Day, Oldfield& Lock, Experiments. In fluids, Verlag, 2000

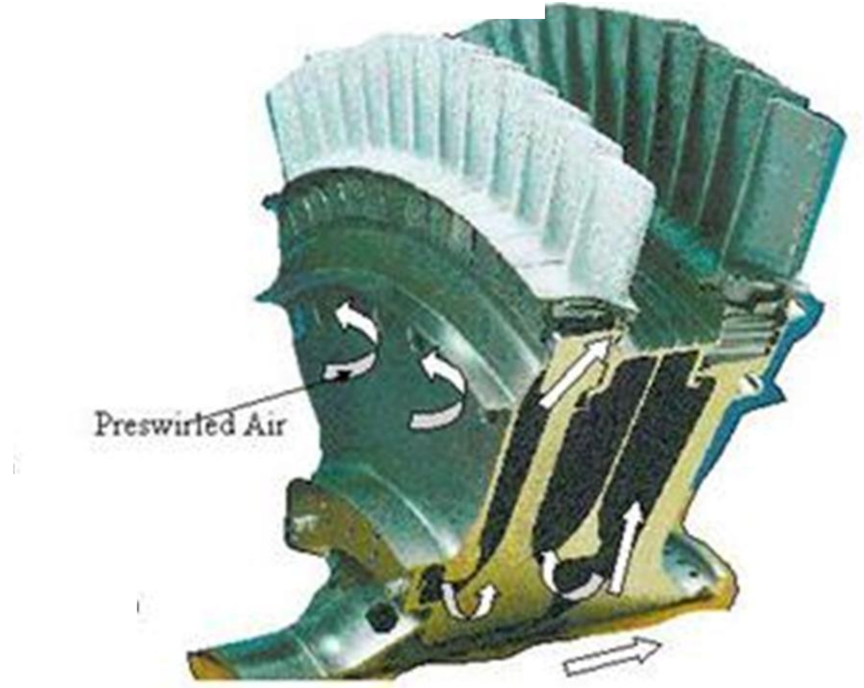
- Nozzle Vanes have to be designed for a peak hot-spot temperature anywhere in the LE except near the tip and hub (10-15% from the endwalls)
- Air flow for the shower head cooling of the LE practically does not affect stage performance but reduces effective gas temperature that has to be compensated 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

Major Requirements for Nozzle and Blade Cooling System Design

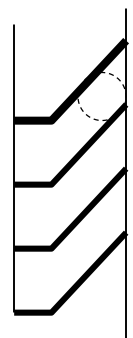




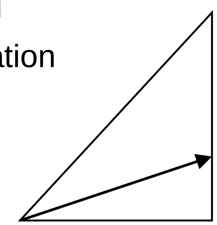




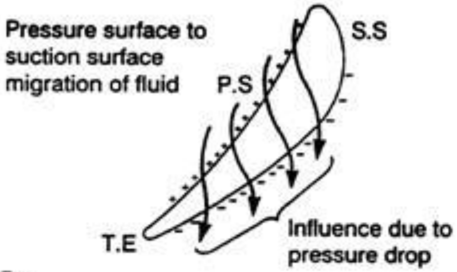
$$\Delta T^{0}C^{accel} = (2UV\cos\alpha - U^{2})/2R\gamma/(\gamma - 1) + \Delta T^{0}C^{frict} = U^{2}(\gamma - 1))/2R\gamma$$

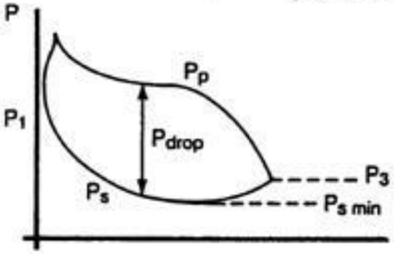


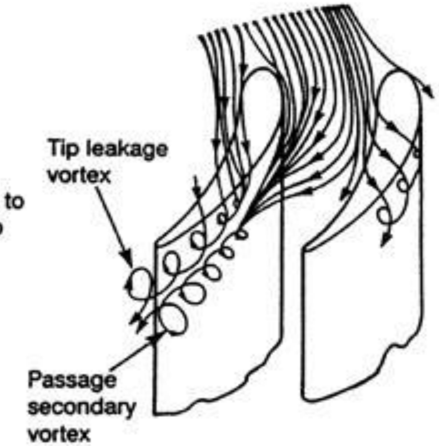
Simplified configuration

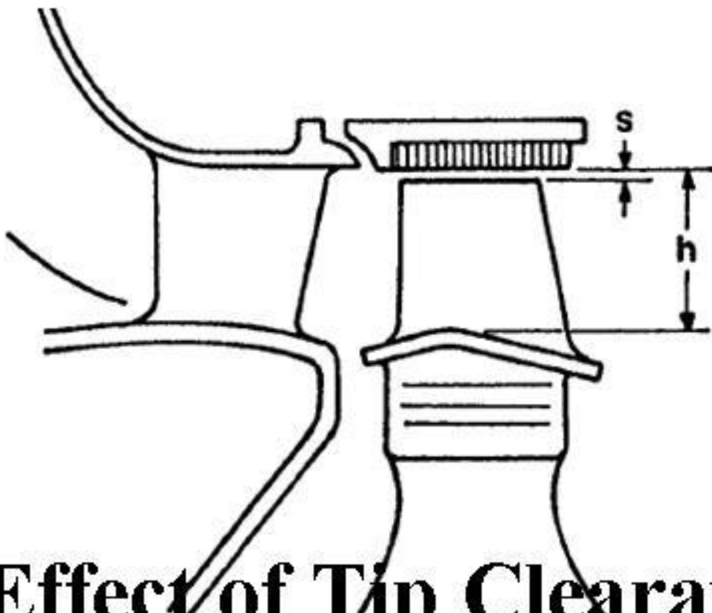


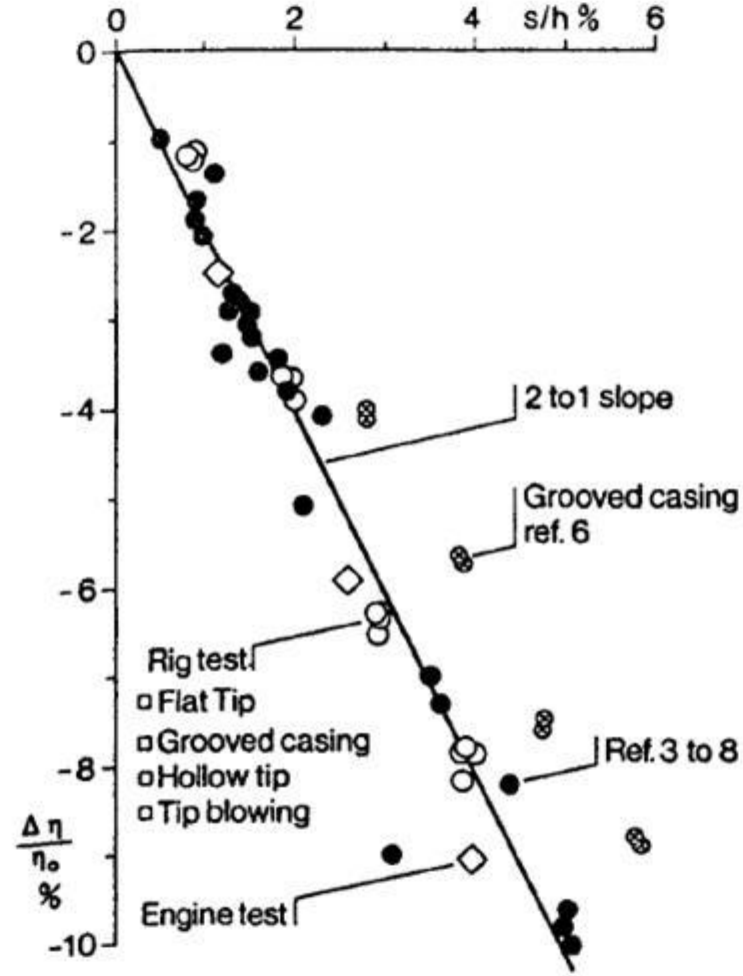
Benefits of Air Preswirling Into Disc Cavity



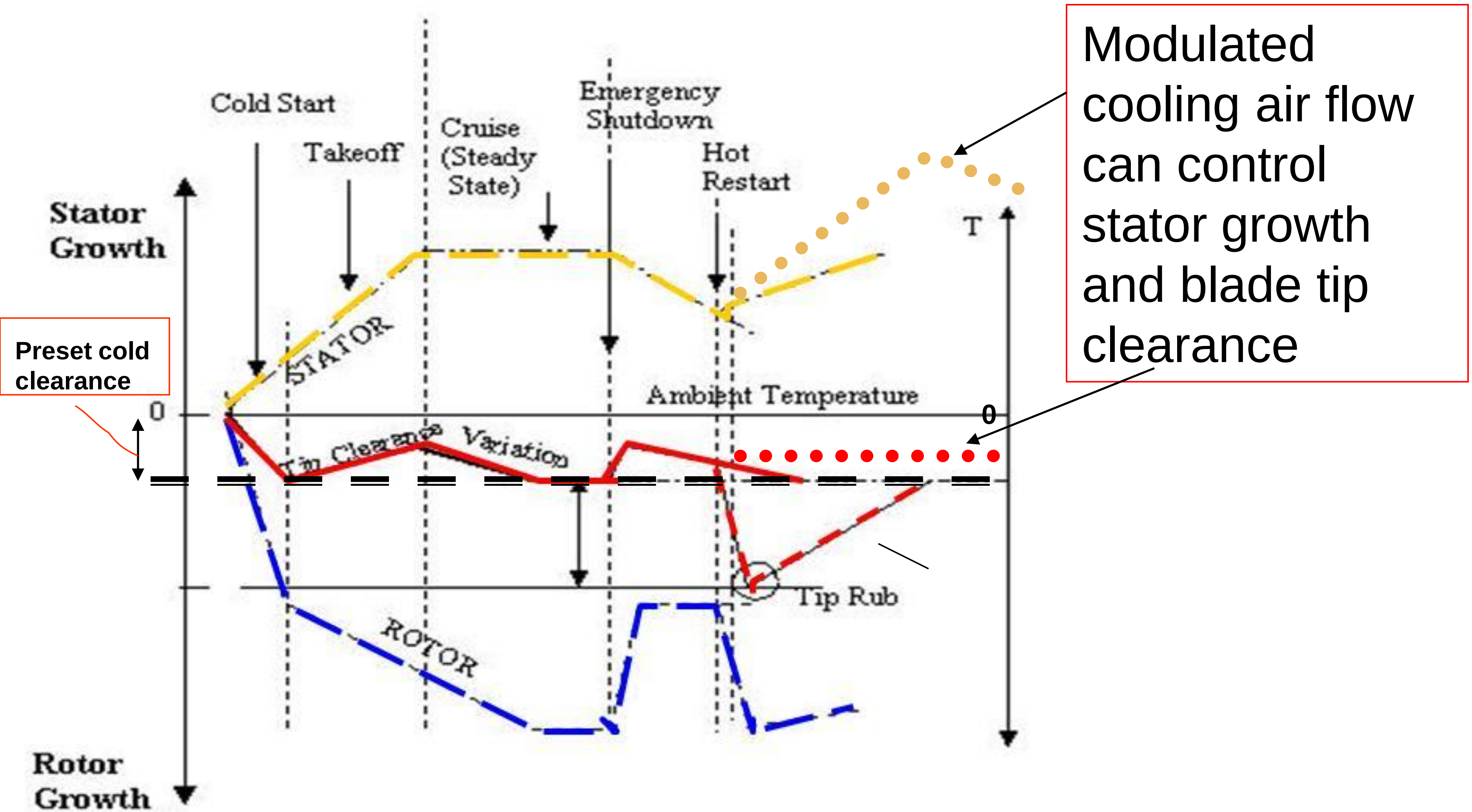




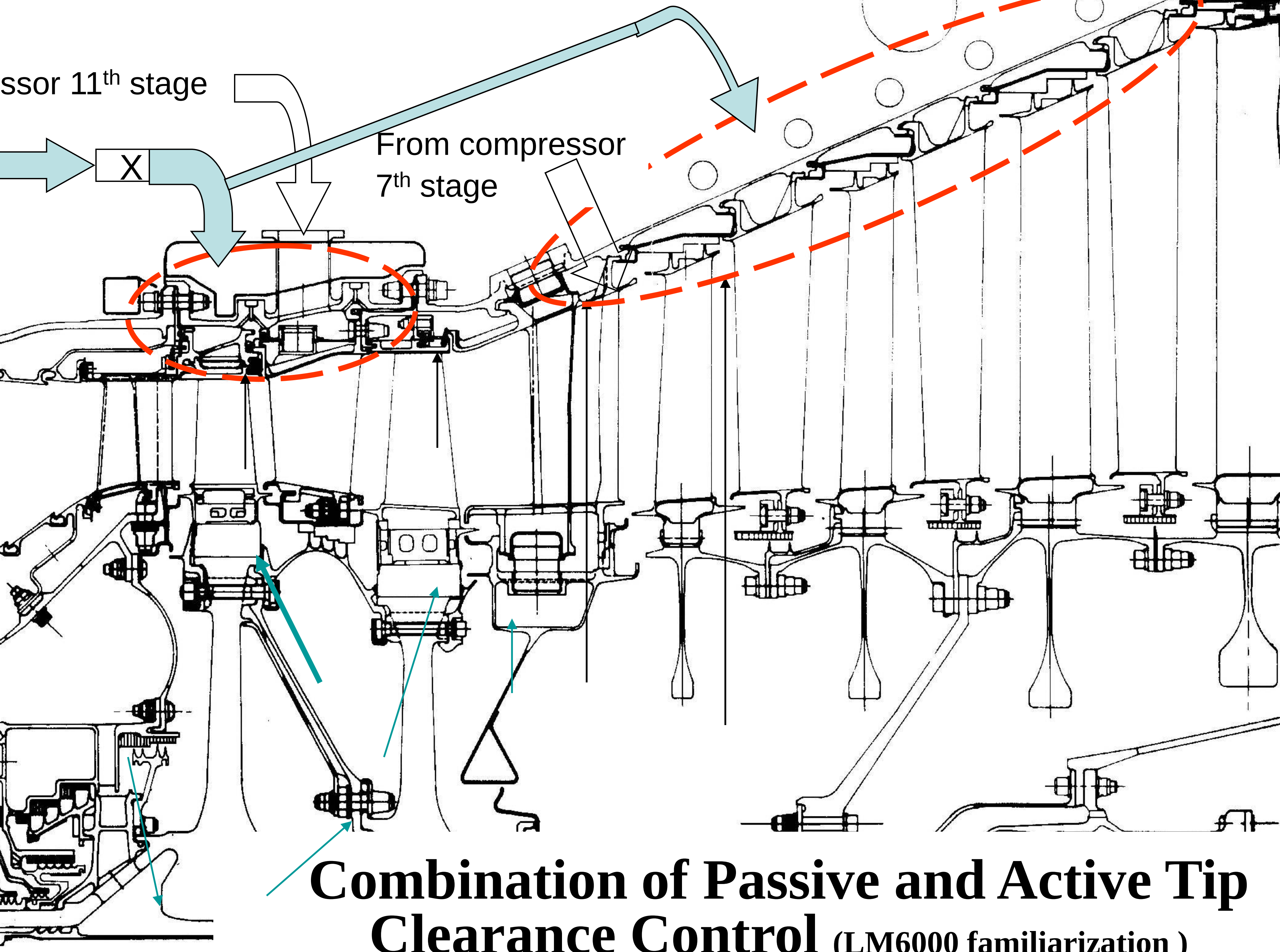




Effect of Tip Clearance on Turbine Stage Efficiency



Effect of Engine Operating Modes on 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 DISCHARGETHE 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