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EVOLUTION OF TURBINE COOLING

Ronald S. Bunker
Consultant
Placitas, NM 87043

ABSTRACT

Turbine cooling is a battle between the desire for greater hot section component life and the techno-economic demands of the marketplace. Surprisingly little separates the haves from the have nots. The evolution of turbine cooling is loosely analogous to that of the Darwinian theory of evolution for animals, starting from highly simplistic forms and progressing to increasingly more complex designs having greater capabilities. Yet even with the several generations of design advances, limitations are becoming apparent as complexity sometimes leads to less robust outcomes in operation. Furthermore, the changing environment for operation and servicing of cooled components, both the natural and the imposed environments, are resulting in new failure modes, higher sensitivities, and more variability in life. The present paper treats the evolution of turbine cooling in three broad aspects including the background development, the current state-of-the-art, and the prospects for the future. Unlike the Darwinian theory of evolution however, it is not feasible to implement thousands of small incremental design changes, random or not, to determine the fittest for survival and advancement. Instead, innovation and experience are utilized to direct the evolution.

Over the last approximately 50 years, advances have led to an overall increase in component cooling effectiveness from 0.1 to 0.7. Innovation and invention aside, the performance of the engine has always dictated which technologies advance and which do not. Cooling technologies have been aided by complimentary and substantial advancements in materials and manufacturing. The state-of-the-art now contains dozens of internal component cooling methods with their many variations, yet still relies mainly on only a handful of basic film cooling forms that have been known for 40 years. Even so, large decreases in coolant usage, up to 50%, have been realized over time in the face of increasing turbine firing temperatures. The primary areas of greatest impact for the future of turbine cooling are discussed, these being new engine operating environments, component and systems integration effects, revolutionary turbine cooling, revolutionary manufacturing, and the quantification of unknowns. One key will be the marriage of design and manufacturing to bring about the concurrent use of engineered micro cooling or transpiration, with the ability of additive manufacturing. If successful, this combination could see a further 50% reduction in coolant usage for turbines. The other key element concerns the quantification of unknowns, which directly impacts validation and verification of current state-of-the-art and future turbine cooling. Addressing the entire scope of the challenges will require future turbine cooling to be of robust simplicity and stability, with freeform design, much as observed in the “designs” of nature.

INTRODUCTION

The theory of evolution of species as amply put forth by Darwin [1], says that the provision of infinite random variations in features and functions tested over many generations will under natural selection lead to the survival and propagation of the fittest variants. This necessarily means that the majority of variants in this “design” process will either fail or be superseded over time. Also, under the influence of changing environments and evolving competition, this will require the fittest to be constantly undergoing modifications and upgrades to remain on the leading edge of survival. The comparison of natural evolutionary theory for biological systems to the development of turbine cooling in mechanical systems is an inexact parallel to be sure, but instructive none the less. Each require, or demand, continual improvements and upgrades, one having thousands of generations for testing, and the other being compressed into but few. Each faces constraints and limitations, and each is influenced by both natural and artificially imposed conditions. In the science and technology of cooling turbines, there are now several generations of advancement representing its development background, a range of variants and degrees of success forming the state-of-the-art in operation currently, and also a host of feasible, hopeful, and dreamlike new generation designs yet to be truly tested. It is the aim of this paper to guide the reader through each of these areas in some depth, and to develop an understanding for the links between them, the reasons for the preference of some cooling technologies over others, the limitations inherent in current technologies, and most especially what the future evolution may hold barring unforeseen external influences.

First the stage needs to be set concerning what turbine cooling covers, and does not cover, in the present paper. A generic aviation engine cross section is shown in Figure 1 [2]. The gas turbine engine is a thermal device and so is composed of a multitude of major and minor cooling and heating systems. Engine cooling encompasses not only the high-pressure (HP) turbine cooling, but also combustor system cooling, heat exchangers, casings, bores, compressor and turbine disks, bearings and gears, exhaust nozzles, de-icing, and even fire suppression. This list is by no means complete, but clearly indicates that turbine cooling is only a fraction of the total engine system cooling challenges. However, turbine cooling does concern the sub-system with the highest economic impact on engine development and repair costs, representing up to 30% of the total.

As a thermodynamic Brayton cycle, the efficiency and/or performance of the gas turbine engine can be raised substantially by increasing the firing temperature of the turbine. Modern gas turbine systems are fired at temperatures far in excess of the material melting temperature limits. This is made possible by the aggressive cooling of the hot gas path (HGP) components, the use of advanced materials for structural components and protective coatings, the application of high efficiency aerodynamics, the use of prognostic and health monitoring systems, and the continuous development of improved mechanical stress, lifing, and systems interactions and behavioral modeling [2,3]. The HP turbine contains the most advanced high temperature alloys and associated processing methods, which together with the combustor represent the key components that have limited life, and so tend to strictly dictate the cycles of operation and the allowable time on wing. Performance gains from the advancement of turbine cooling, for example reduced specific fuel consumption (SFC) or increased specific thrust, are of extreme value in competition (i.e. price, service costs, and market survival).

For these reasons, the evolution of turbine cooling has been lavished with constant and substantial research for several decades. The focus of this paper will hence be on the cooling of the HP turbine components, though the technologies are employed elsewhere, e.g. combustors, as depicted in Figure 2 [2]. Actively or passively cooled regions include the stationary vanes or nozzles and the rotating blades or buckets of the HP turbine stages, the shrouds bounding the rotating blades, and the combustor liners and flame holding segments. Collectively these components are referred to as the hot gas path (HGP). The turbine alone may use 20 to 30% of the compressor air for cooling, purge and leakage flows, which presents a severe penalty on the thermodynamic efficiency unless the turbine inlet temperature (TIT) is sufficiently high for the gains to outweigh the losses. With advances in technologies, the amount of compressor air used for cooling the turbine is trending downward now even in the face of further increasing TIT. This summary will also refer mainly to aviation turbine cooling, but the features and physics described are almost without exception applicable to land-based engines.

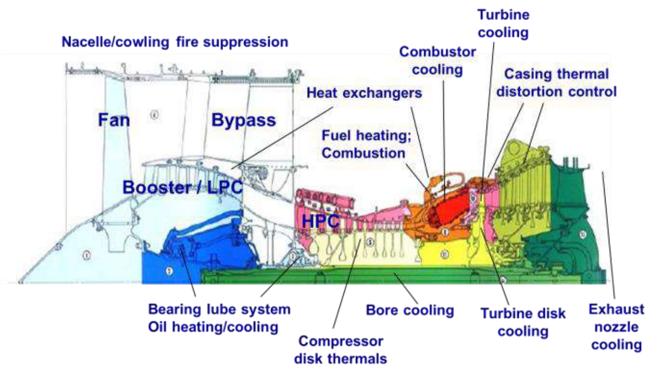


Figure 1. Engine heat transfer and cooling [2]

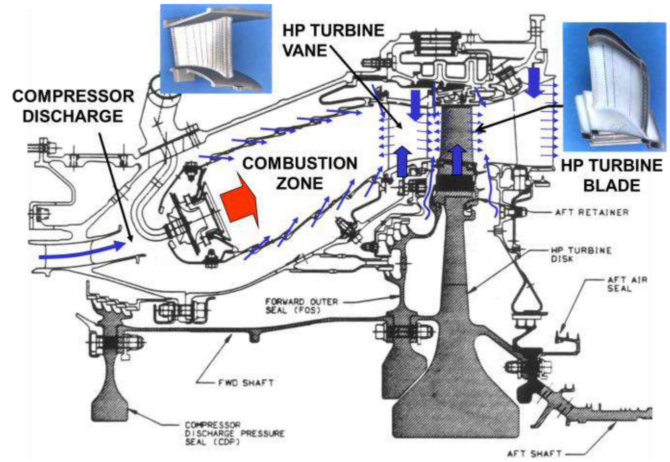


Figure 2. Turbine and combustor hot gas path cooling [2]

BACKGROUND TO TURBINE COOLING DEVELOPMENT

In general, the driving motivations for any and all developments in turbine cooling are (1) engine efficiency and/or performance increases as the hot gas temperature rises and as overall cooling effectiveness increases for a given amount of cooling flow, and (2) engines indicate when and where improved or added cooling is required for survival. In response, the cooling methods employed in gas turbine components have evolved over the years from simple smooth cooling passages to very complex geometries involving many differing surfaces, architectures, and fluid-surface interactions. The fundamental aim is to obtain the highest overall cooling effectiveness with the lowest possible penalty on the thermodynamic cycle performance. This is shown in Figure 3 in the form of notional cooling technology curves [2]. Overall cooling effectiveness charts the ability to bring the component bulk metal temperature (T_m) level to that of the cooling fluid (T_c) versus the hot gas temperature (T_g). As a component level cooling effectiveness, bulk metal temperature within the hot gas path (e.g. a blade airfoil) is most closely associated with component life, cooling fluid temperature is that supplied to the component, and the hot gas temperature is that at the inlet plane to the component. Other technology curves can be based upon other definitions, but typically these provide a common basis for comparing many differing designs. Cooling is challenged by the external heat loading level (UA_g) and the thermal capacity and amount of coolant used (W_cC_p). The ratio of these terms as the heat load parameter conveniently compares the effective total heat load applied to the hot surfaces (convective and radiative) to the maximum capacity of the coolant to accept that heat. Because the coolant supply and sink locations are separated by a finite quantity of the component to be cooled, the effectiveness never reaches unity. In practice, the incremental gains in effectiveness diminish as more coolant is used, just as in a heat exchanger, which is exactly what each cooled component represents. While the form of cooling effectiveness curves remains the same for all cooled components, vanes, blades, shrouds, and so forth will have different actual curves within each technology band based on their design needs and constraints.

The constraints common to all cooled components throughout historical development include, but are not limited to, pressure losses, material temperatures, component stresses, geometry and volume, aerodynamics, fouling, and coolant conditions. Turbine cooling technology has developed with the use of five main elements: (1) internal convective cooling, (2) external surface film cooling, (3) materials selection, (4) thermal-mechanical design, and (5) selection and/or pre-treatment of the coolant fluid. The introduction of these cooling technologies has, as depicted in Figure 3, progressed over time in the order noted here, from straight internal radial cooling passages only, onward to the inclusion of film cooling, protective coatings (i.e. thermal barrier coating, TBC), the use of improved or alternate high temperature materials, and alternate cooling fluids (the latter in land based turbines). Furthermore, within each broad technology area, there are a multitude of variations in form and complexity, such that each cooling effectiveness curve is actually composed of many differing designs encompassing a cloud of design space, some even overlapping between major technology introductions. Over the course of time, component cooling effectiveness has moved from earliest gross effectiveness of about 0.1 to current levels as high as 0.7. This progression has taken place through concurrent improvements in technology or the introduction of new technologies (vertical movement in Figure 3), and reductions in coolant usage (movement left in Figure 3). Increases in firing temperature are usually obtained with new technologies, and initially demonstrated on military engines seeking higher thrust-to-weight ratios. Decreases in coolant usage are obtained through improved designs and experience, and continuously sought for commercial engines. Most typically, advancement in both directions is sought, or required, to meet new product goals in both commercial and military applications.

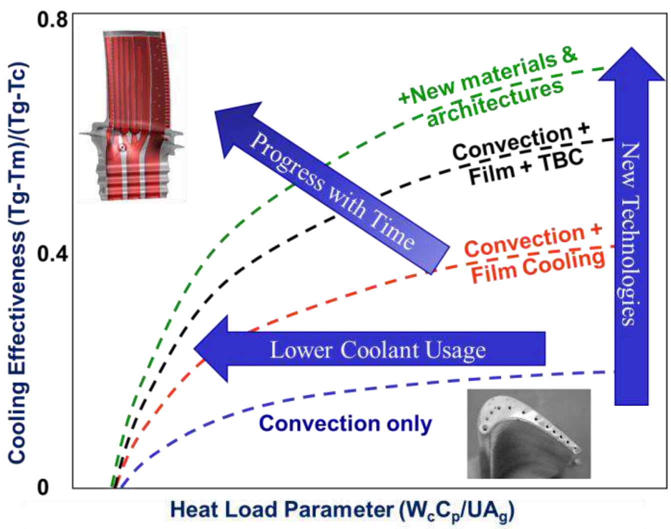


Figure 3. Notional component cooling technology curves

To add some perspective to Figure 3, the ranges of present day gas turbine conditions and temperatures are provided in Table 1 for both large commercial aircraft propulsion and ground based power production turbines. The parameter ranges reflect the fact that a broad range of gas turbines are in operation, including parts that are cooled only by convection to those that use the most recent technologies. Turbine firing temperatures, usually taken as the hot gas temperature just prior to work extraction by the first turbine stage, are for the most part above the softening temperatures of the metal alloys used in HPT components. In practice though, metal temperatures must be kept several hundred degrees lower yet in order to maintain strength and durability. The range of coolant supply temperatures shown are based on compressor discharge temperatures, which will typically require adjustment upwards, sometimes significantly, for heat addition in the coolant secondary supply circuits.

Table 1. Turbine condition ranges for products in service

Parameter	Large Commercial Aviation Engines	Heavy Frame Power Turbines
Overall Pressure Ratio	20 - 60	15 - 35
Turbine Firing Temperature (T41)	1315-1650 C (2400-3000 F)	1100-1430 C (2000-2600 F)
Coolant Supply Temperature (T3)	480-650 C (900-1200 F)	375-550 C (700-1000 F)
Metal Temperature (softening)	1260 C (2300 F)	1260 C (2300 F)
Maximum TBC Temperature	1315-1550 C (2400-2800 F)	1315-1550 C (2400-2800 F)
Coolant Usage (%Wc)	20 - 30%	5 - 20%

One over-riding constraint common to all but the simplest designs in commercial use today, is that internal cooling features, i.e. the foundational level of cooling technology in Figure 3, rely on investment casting manufacturing. This manufacturing approach is amenable to materials most commonly employed in cooled parts, namely high-temperature, high-strength nickel- or cobalt-based superalloys processed in equiaxed, directionally solidified, or single crystal metallurgical formats. Figure 4 schematically illustrates the five conventional cooling philosophies that are produced today with investment casting. The simplest uses only internal convection with no film cooling, and is common on many second or third stage vanes and blades that use very little cooling flow. All of the other means employ film cooling to various degrees. Most turbine airfoils utilize internal impingement or convection passages and a moderate amount of film cooling. Combustor systems tend to use either effusion cooling, or effusion with backside convection augmentation very similar to the leading edge cooling of airfoils, to provide a higher degree of cooling within the material and as film cooling. The final method shown is alternately known as double-wall, or near-wall, cooling and places the cooling passages inside the airfoil cast wall, thereby moving the coolant closer to the heat source. More will be said on this method later.

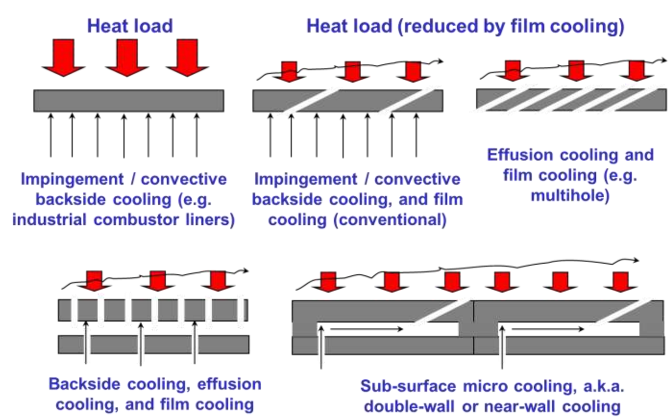


Figure 4. Conventional modes of turbine cooling

Common to all of these generic cooling configurations is the fact that the thermal-mechanical design of the components must have acceptable thermal stresses, coating strains, oxidation limits, creep-rupture properties, and aero-mechanical response. As illustrated in the design cycle of Figure 5, which applies to each individual turbine component as well as the system, the foregoing considerations and much more must be assessed and balanced with respect to the ability to manufacture the components cost effectively. The final design of a cooled component will necessarily be a compromise within the allowable parameters, though certain factors may be weighted more than others. Within this multi-disciplinary space, however, turbine cooling typically has the most powerful relationship with manufacturing, and vice-versa. The relationship of Figure 6 calls for required cooling means, e.g. complexity, to be met by manufacturing and also for manufacturing to restrict cooling capabilities according to proven high yield methods. This has resulted in a strong dual dependency in the evolution of turbine cooling. In the earlier years of development, roughly up to year 2000, cooling technologies tended to lead manufacturing, as incremental improvements to the latter were made while more far reaching cooling improvements were demonstrated in laboratories and development engines. In the more recent period since 2000, the roles have reversed to where new manufacturing technologies, e.g. various additive methods, are now outpacing the demonstrated cooling methods, at least in potential or promise.

Figure 5. Turbine component design cycle (solid lines represent core disciplines most closely tied to cooling)

Once the cooling and manufacturing limits are reconciled one to the other, thermal stresses caused by thermal gradients in the airfoil substrate are a key contributor to design limits, and also failures. Almost without exception, a design which significantly reduces the through-wall thermal stresses, and to a lesser extent the in-plane gradients, in all or most of an airfoil will result in a more efficiently cooled and longer life airfoil. Greater degrees of cooling uniformity, or greater local control over specific cooling magnitudes, can be utilized to reduce local thermal gradients. But the historical limitations of manufacturing methods, for example lower limits on film hole sizes, minimum allowable spacing between film holes, and minimum trailing edge cooling channel sizes, has restricted the ability to cool components in the most desirable ways.

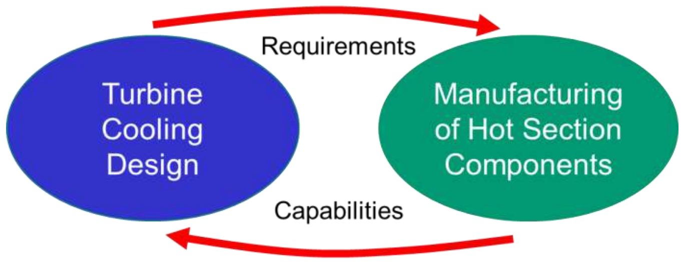


Figure 6. Dependent relation between cooling and manufacturing

All of the foregoing discussion concerns conventional/historical cooling of components. While most forms of conventional cooling can be made to work in most circumstances, there remain several limitations, as delineated in [4], which require continuing development and ingenuity to advance the state-of-the-art. These limitations include difficulties in distributing cooling where and in what quantity it is required, significant non-uniformities of cooling, highly complex fluid mechanics leading to internal heat transfer and flow characteristics that are difficult to measure or predict, serially linked cooling flow networks, investment casting complexity and part-to-part variability, and costly tooling revisions for any cooling changes that require modification of the casting.

This is not to say that historical (older) turbine cooling methods are bad, because clearly they have served very well. Instead, it has simply been a process of “evolution” and invention from the simplest means of cooling to more complex and capable means. The older cooling designs (e.g. 1980’s blade cooling) frequently continue in operation today with minor alterations despite newer technologies that might greatly improve them, though this is not an exclusive conclusion. To substantially alter or completely replace such designs would lead to a cascade of system changes, re-design, re-balancing, new hardware, new verifications, and possibly even recertification. Typically, only relatively benign changes such as the addition of some TBC, or perhaps a few strategically placed film holes, are made. It should also be kept in mind that any new technology must also buy its way onto an engine in a cost versus performance trade. In the short history of gas turbines, the proven [older] means of turbine cooling fits the niche within its environment and market.

An additional note must be made here about “failures”. The vast majority of failures for cooled turbine components do not constitute engine failure, but instead simply means that the part fails to meet its intended life. Failures can however have significant consequences in terms of SFC, time on wing, servicing costs, and warranty and concession costs. The cooling features and methods that survive and propagate are those proven to be manufacturable and of reliable performance.

A BRIEF DIGRESSION FOR HEAT EXCHANGERS

A cooled turbine component is at heart an air-to-air heat exchanger. What could be simpler? There are volumes to fill a library on heat exchanger designs and operational characterization for high volume bulk fluid processing units, compact units for electronics cooling, and micro or even nano units for special functions and future applications. Turbine components are differentiated from ordinary heat exchangers by the many cooperating functions shown in Figure 5. For the purpose of this summary however, there are two key aspects of differentiation. First, the objective in cooling a turbine part is to transfer as little energy as heat as possible, using as little coolant as feasible, while satisfying all other design constraints. This is markedly different from most heat exchangers that desire to transfer a maximum amount of energy as heat, up to the limit of both fluids being equal in temperature. Second, the cooling air used in the turbine parts must be returned to the hot air for maximum utilization of energy in the Brayton cycle. Hence the turbine component is an open heat exchanger with many bleed apertures, most of which attempt to inject coolant as film cooling on the HGP surfaces. A component with no film cooling, some shrouds for example, are essentially co-flow, counter-flow, or mixed flow heat exchangers. Components with film cooling are a combination of two connected heat exchangers, one being the standard two-temperature exchange separated by a wall, the other being a three-temperature exchange and fluid mixing "device" (i.e. film, hot gas, and wall). The thermal resistance network model is depicted in Figure 7, including the additional term of a radiation heat exchange modeled as an equivalent heat transfer coefficient.

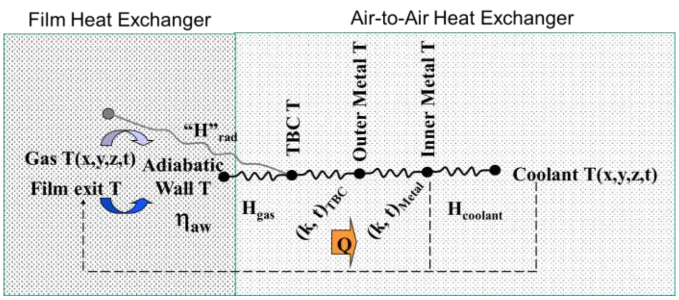


Figure 7. Heat exchange thermal resistance network

In the simplified thermal network, aside from fluid and material properties and temperatures, there are two coefficients that virtually define the art of turbine cooling modeling, namely the heat transfer coefficient H, or 'h', and the adiabatic film effectiveness η. Heat transfer coefficients (HTC) apply at both the exterior and interior surfaces of the components, as well as any other surfaces in contact with fluid in motion (e.g. seal faces, film holes). The well-known Newton's Law of Cooling, which is actually not a law of physics, is a very convenient construct that relates the heat flux at a surface to the thermal driving potential

$$Q / A = h\,(T_{\text{gas reference}} - T_{\text{wall}})$$

This is shown graphically in Figure 8 for an external turbulent boundary layer. Replacing the wall heat flux with the alternate definition of the thermal gradient conducted immediately at the wall in the fluid provides

$$h = \frac{k * (dT/dy)_{\text{wall}}}{(T_{\text{gas reference}} - T_{\text{wall}})}$$

The HTC is then seen as proportional to a ratio of the thermal gradient at the wall to the temperature difference in the fluid driving the heat flux. It is highly inconvenient to measure thermal gradients in the fluid at the surfaces, which then leads the HTC to be more or less easily determined by heat flux and temperature difference measurements. For external flows, the gas reference temperature is either the gas recovery temperature (no film cooling), or the adiabatic wall film temperature (see next paragraph). For internal cooling passages, the local mixed bulk fluid temperature is used. In practice, HTC values are measured on just about any surface in model tests, sometimes even when not appropriate to do so. For many situations, such as turbulated channels or impingement jet arrays, HTC is readily correlated over Reynolds number ranges and other parameters of interest, thus becoming very useful for design purposes. For many other situations, such as external hot gas flows over rough surfaces with boundary layer transitions, film injection, acceleration, and so forth, HTC is not amenable to correlation. However, in such cases, the several effects of parameters such as roughness upon HTC can be correlated, or used to validate predictive tools. The combination of external and internal surface HTCs with some form of dividing wall is then one of the heat exchangers requiring characterization for turbine cooling.

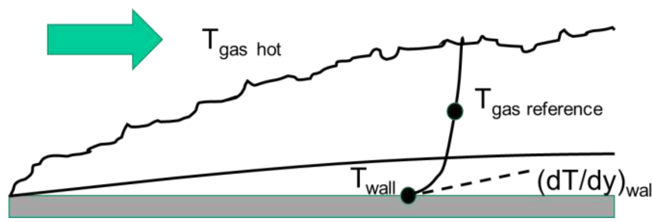


Figure 8. Defining the heat transfer coefficient

Hot surface film cooling is the practice and art of bleeding the internal cooling fluid through the walls of the components to form a thin layer of coolant on the surface, thereby decreasing the effective mixed fluid temperature (coolant mixed with hot gas) that drives the incident heat flux at the outer surface (excepting thermal radiation). This fundamental aim is represented in Figure 9 and quantified by the local adiabatic film effectiveness η

$$\eta = (T_{\text{recovery}} - T_{\text{adiabatic wall}}) / (T_{\text{recovery}} - T_{\text{coolant}})$$

The definition shown uses the local hot gas recovery temperature, and the coolant static temperature exiting the film apertures, as the sources for local mixing. The effectiveness η is simply the ratio of two temperature differences, the actual thermal driving potential from gas to wall, to that representing the maximum possible potential between gas and coolant. The effectiveness then can reach as high as 1 for complete coolant coverage and no mixing, and also will eventually become 0 as the fluids are completely mixed and hot gases dominate. Consider film cooling to be a heat exchanger without the dividing walls, with most configurations being co-flow and some being mixed co-flow and counter-flow. One further note, the term adiabatic film effectiveness may be confusing considering that cooled HGP components are not adiabatic. This convention arises from the need to measure and correlate film effectiveness for use as boundary conditions in design, yet do so in manageable environments, i.e. in lab testing rather than actual engines. A common and somewhat convenient basis is found by testing with adiabatic walls, rather than having data from various researchers use many different conditions. The adiabatic wall also serves as the ideal and limiting condition.

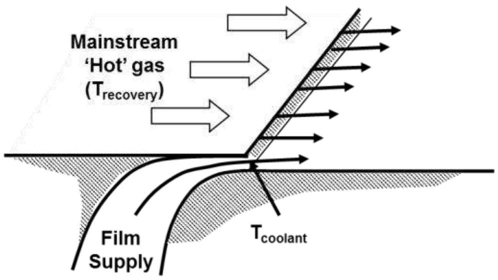


Figure 9. Defining adiabatic film effectiveness

Determining the correct heat transfer coefficients and film effectiveness values are the priority design objectives for turbine cooling, along with other necessities such as discharge coefficients and friction coefficients. Historically, nearly all values of HTC and h have been developed and re-developed over time as correlations with few parameters (e.g. Re, Pr, x/D, relative roughness, mass flux ratio, etc.). Dozens of correlations, at least, have been required to represent the many fluid-surface interactions, design alternatives, and transitional flows. The definitions above still form the basis of all design boundary conditions, at least until such time as validated and verified computational models supersede further need for these characterizations.

CURRENT STATE-OF-THE-ART (SOA)

The overall status of conventional HGP cooling, as well as the main challenges associated with cooling technology moving forward, were summarized a decade ago in the review of Bunker [5]. Those main challenges were then identified as including:

- Uniformity of internal cooling
- Ultimate discrete hole film cooling
- Secondary flows / leakages as prime HGP cooling
- Contoured, non-axisymmetric HGP surfaces
- Component cooling architecture (e.g. micro cooling)
- Thermal gradient reduction (i.e. thermal stresses)
- Combustor-turbine system integration
- Reduction of incident heat flux (e.g. reflective coatings)
- Controlled or adaptable cooling
- Regenerative turbine cooling (energy re-use within engine)

Progress in advancing each item can be measured in the space of a few years for fundamental research, but actual validation and engine verification of performance or life improvements is a matter of more than a decade, and usually longer for commercial fleet applications. During this past decade from 2005-2015, all of the items listed have seen progress in some measure, but few or none have been implemented in engine operations on any widespread basis. The definition of SOA used in this paper is whether an advancement area is used in commercial fleets for more than temporary or in-service testing. Another way to state this is that the technology is in essence considered prime reliable for production and operation. Certainly, military development and demonstration turbines are another matter, frequently involving cooling technologies that are classified and hence will not be discussed here.

The present summary of the SOA is not intended as a complete compendium or bibliography, nor is it a recommendation of design practices. Only a high level description of the SOA is presented here, leaving the details of design information to the thousands of published papers, and several books, covering dozens of topics in turbine cooling. For example, the compilation of Han et al [6] briefly summarizes the content and findings of many of the individual cooling technology studies published up to about 1998. The edited volume of Shih and Yang [7] presents good summary of the SOA for the main HP turbine design disciplines (not only cooling) with one or more chapters focused on each discipline. Another source of in-depth cooling related summaries are the von Karman Institute lecture series publications, e.g. [8, 9]. The present SOA is very briefly summarized next, followed by snapshots of the many individual SOA techniques in turbine cooling.

A typical SOA cooled HPT nozzle is depicted in Figure 10, and a typical cooled HPT blade is shown in Figure 11 [2]. The nozzle is largely cooled using internal impingement jet arrays, an abundance of discrete rows of film holes, and axial cooling channels within the thin trailing edge. A SOA turbine inlet nozzle uses between 7 and 13% of compressor discharge air for cooling and leakages of airfoils and endwalls, depending on the combustor exit temperature, airfoil count, and loading. The blade is cooled mainly with a serpentine channel having regularly spaced turbulators on the interior pressure and suction side walls, impingement in the high heat load leading edge region, trailing edge channels with pressure side bleed slots, and strategically placed rows of film holes. A SOA turbine blade uses between 5 and 8% of compressor discharge air for cooling of the airfoils, platforms, and tips. The blade's opposing shroud, or blade outer air seal, may use 1 to 3% air. If the HP turbine has two stages, then another nozzle, blade, and shroud may require cooling, though usually with earlier compressor stage bleed air, and lesser amounts. Many specialty cooling areas exist on these components, such as the inner and outer nozzle endwalls, the blade platform, and the blade tip. In truth, for the component designer, every cooled region no matter how large or small is special, requiring in-depth knowledge of past engine experience and present knowledge of design boundary conditions.

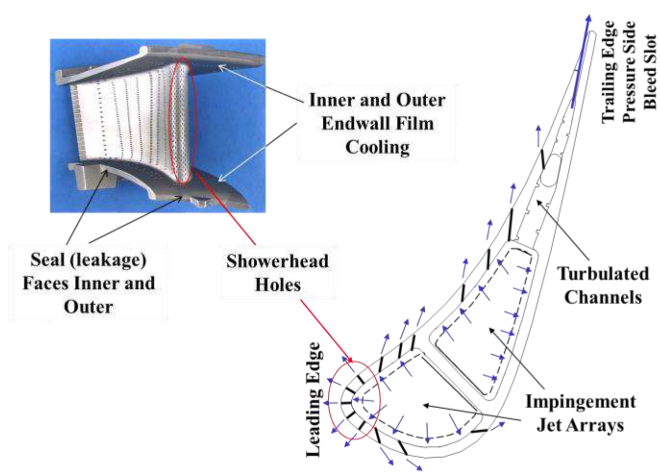


Figure 10. Typical cooled HPT nozzle [2]

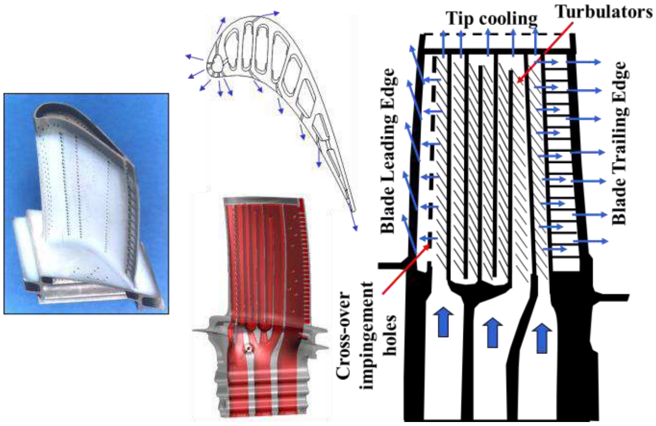


Figure 11. Typical cooled HPT blade [2]

One key aspect of turbine component cooling is that no single cooling method is both sufficient and necessary in all localities, nor for all components. So for example, while the nozzles and blades all constitute aerodynamic airfoils, the means for cooling them can be very different. The adage common amongst turbine designers is that the ideal aerodynamic design requires each airfoil to be as thin as possible, with sharp leading edge and vanishingly thin trailing edge, while the ideal structural design prefers a straight or tapered solid cylinder for load bearing, and the ideal thermal design wishes for a generously hollow airfoil with blunt leading edge and thick trailing edge to allow ease of cooling with any of several methods. But these three ideals are largely mutually exclusive if the turbine is to be at all efficient or competitive. The typical compromises are the nozzle and blade designs shown here, though these are not universal. It is possible to force a nozzle and blade to have the same basic cooling design, but one or both will sacrifice efficiency. For example, turbine inlet nozzles use impingement cooling, and require greater airfoil thickness, because the available pressure driver for the coolant is the lowest of all components. The use of serpentine cooling in this nozzle would lead to hot gas ingestion. In contrast, the blade design, as the rotating work extraction airfoil, must place a higher premium on aerodynamics, which rules out the use of nozzle-like impingement cooling because the required internal volume would sacrifice the aerodynamic efficiency.

Integrated Cooling Designs

Turbine cooling constitutes many dependencies and inter-relationships for each component. Figure 12 depicts the thermal design knowledge (boundary conditions) that is required in a fully distributed format for each part and each surface [2, 3, 7]. To the left and right are listed the many critical functions or objectives or requirements that are directly influenced by the cooling design. Some requirements are localized issues and others have a more global engine impact. Below the cooling design box is again the unavoidable relationship with manufacturing, but this time concerning the tolerancing and variability of manufacturing any one or more feature that impacts cooling, rather than simply whether the part can be made cost effectively. This means explicitly incorporating into the design, preferably in a probabilistic manner, all of the variability in the many boundary conditions. The factors noted at the top of the figure all concern engine operation and experience, including the transient change in boundary conditions with any shift from one cycle point to another, the alteration of boundary conditions and actual component degradation with time and environmental/operator differences, and any experiential factors that may encompass reality as yet not fully understood. It is absolutely critical in the understanding of turbine cooling to realize that everything shown in Figure 12 is inter-related. This can lead to somewhat non-intuitive realities in some circumstances, for example desiring lesser cooling capability, rather than greater.

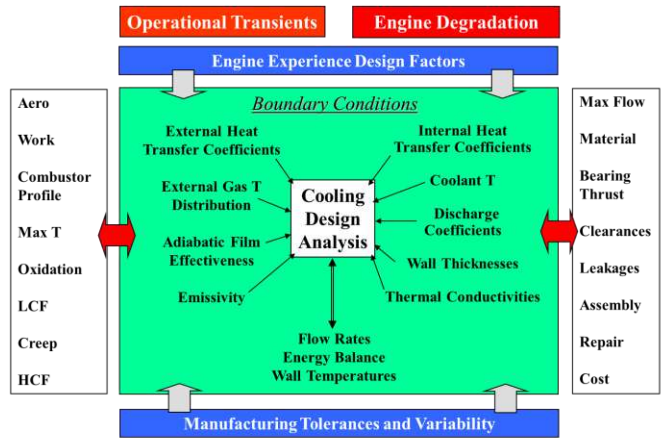


Figure 12. Integrated detailed component cooling design

External Surfaces and Flows

Before enumerating active turbine cooling methods, a note is required concerning the reason for cooling, namely the external HGP heat load on these components. The external heat load is a combination of convective and radiative heat transfer, with the majority of the convective portion being from turbulent or transitional flows. The radiative heat component is mainly non-luminous flux from the hot gases to the HGP surfaces (e.g. TBC surfaces), with the exception of the HPT nozzle leading edge region which has a direct line-of-sight to the flame zone. Obviously, the combustor walls are the most impacted by radiative flux, but fortunately tend to have much lower convective HTCs. Regardless, for turbine and combustor, there are many variables that directly impact the magnitude of the total external heat loading, including aerodynamic loading, surface roughness, freestream turbulence, swirl, rotational effects (blade), boundary layer disturbances (e.g. film injection), hot gas temperature and pressure profiles, hot streaks, periodic unsteadiness (e.g. wakes), secondary flows, and film effectiveness.

Several design factors have come to be commonly referred to as cooling technologies, even though they do not use active cooling flow. A few examples will be described here, but the reader is encouraged to seek out more specific references for further information. First, and usually of the most impact, is TBC type (porous or dense) and thickness. Increasing TBC thickness can lead directly to lower metal temperatures or the use of less coolant, but only if the maximum allowable TBC temperature, as well as the bond coat temperature, limits are not exceeded. As the TBC thickness is increased however, there will arise a limit beyond which the risk of coating loss increases, with the severity of consequences increasing as thickness increases. Second, the external surface roughness can be a strong driver for convective heat transfer [10-19]. Reducing roughness by initial polishing will lower external heat load, at least in the short term. Just how much to polish and in what regions the benefits will be longest lasting is not entirely clear even today. Third, while the use of non-axisymmetric contoured endwalls and platforms may have a thermal benefit [20, 21], at least in design predictions, this technology is generally driven by aerodynamic efficiency gains rather than cooling needs. The reason for this is that stage aerodynamic efficiency increases will, hopefully, propagate through the entire turbine and engine. At a minimum, any such aerodynamic gain due to contouring should also do no harm to the cooling design. Lastly, film cooling is the one technology that clearly impacts both cooling and aerodynamics. For the majority of any component, the aerodynamic mixing losses incurred do not outweigh the cooling benefits. But for high Mach number regions of component surfaces, the mixing losses can be high, thereby forcing a real design conundrum, whether to add more cooling flow than needed in lower Mach regions in order to reduce mixing loss, or to endure the aerodynamic penalty in favor of a more efficacious cooling design.

Internal Cooling

The fundamental mechanisms used for all HPT component cooling are turbulent convective duct flow and turbulent impingement flow. Isolated circumstances might employ laminar flows, but these are not the norm. Internal convective cooling SOA has relied largely on the enhancement of flow surfaces for the augmentation of otherwise smooth surface heat transfer, for example through the use of turbulators and pin-banks. Surface enhancement methods continue to play a large role in today's turbine cooling designs with new, or newly re-discovered, mechanisms and methods such as those summarized in [22]. Heat transfer coefficient augmentation factors span the range from 1.5 to 5, and total heat flux capability augmentations may reach higher when increased surface area is considered. The basic mechanism for turbulated channels or walls, also called trip strips or rib rougheners, is shown in Figure 13. The protruding elements are transverse or at some angle to the bulk flow, thereby creating a disruption to the near wall fluid that aids the mixing of heated near wall fluid and cooler bulk fluid, bring cooler air to the surface. The disruptions also create flow separations and re-attachments, with associated lower and higher HTC. The average enhancement to HTC depends on many parameters including Reynolds number, angle of attack, relative turbulator height, turbulator spacing, edge sharpness, channel shape and aspect ratio, and rotation. The lower image in Figure 13 provides a typical example of the degree of local HTC distribution on the surface, where the local magnitude can easily vary by a factor of three. Many, many other configurations are conceived and in use, including segmented turbulators, corner wrapped turbulators, chevrons, W-turbulators, and more [6, 23-34].

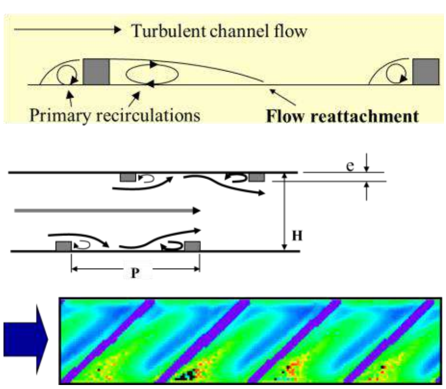


Figure 13. Basic thermal-fluid mechanism for surface turbulators

Turbulators have their limitations, for example yielding insignificant HTC enhancements if the relative height is too small, and on the other end resulting in massive pressure losses when too high. In such situations, the alternative of pins and pin banks, sometimes called pin fins, becomes attractive. Figure 14 shows that the pin fins are typically cylindrical or oblong obstructions oriented mainly transverse to the bulk flow. Wake generation and unsteady flow provides the HTC enhancement by scrubbing the walls, while impinging flows on the pin fins further augment the total heat flux. Again, the many flow and spatial parameters come into play for a wide range of design alternatives [35-42].

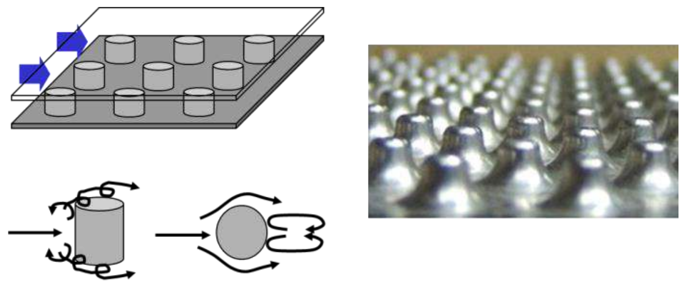


Figure 14. Basic mixing mechanism for surface pins and pin banks

Internal impingement jets and arrays of jets found in the SOA tend to use only smooth surfaces, though augmentation methods have been studied and quantified. Impingement without surface augmentation provides a relatively high enhancement over smooth duct flow heat transfer, up to about a factor of 4, making additional augmentation unnecessary in most cases. Another disincentive to combining impingement with surface features such as turbulators (e.g.) is that the multiplication of parameter combinations and resulting effects, and uncertainties, for design space can become unmanageable. A summary of jet impingement cooling methods can be found in the edited volume of Amano and Sundén [43]. Figure 15, redrawn from Martin [44], displays the fundamental mechanism of a jet impinging on a surface, with consequent stagnation flow region followed by a developing wall jet. The image of local HTC distribution for an axisymmetric jet impinging normal to a flat surface indicates that as much as a factor of 10 can separate the highest from the lowest HTC in a very compact space. Jet Reynolds number and target spacing Z are two primary control parameters for the outcome in the absence of cross flows [45-48]. Basic impingement of this type is common in critical cooling regions like the blade leading edge depicted in Figure 16 [49-53]. The realm of geometry parameters is almost limitless as well, evidenced by the use of trailing edge angled impingement also shown in Figure 16. The other common usage for impingement is in the form of arrays, such as shown in Figure 17 [54-59]. Here the individual jets may still retain their resulting HTC characteristics as seen in the image example, or the spacings, flow rates, etc. may serve to generate highly modifying cross flows even to the degree of forming channel flows.

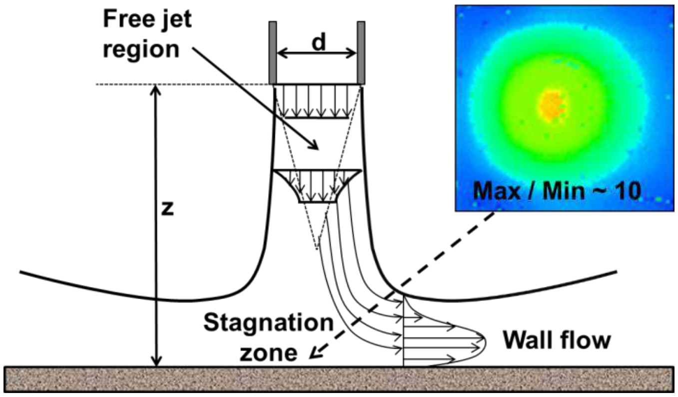


Figure 15. Fundamental single axisymmetric jet impingement

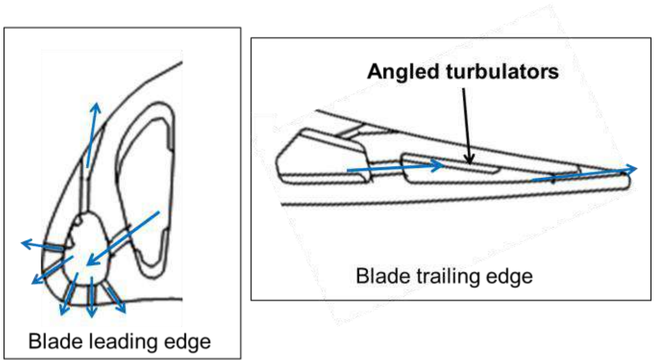


Figure 16. Typical blade leading and trailing edge impingement

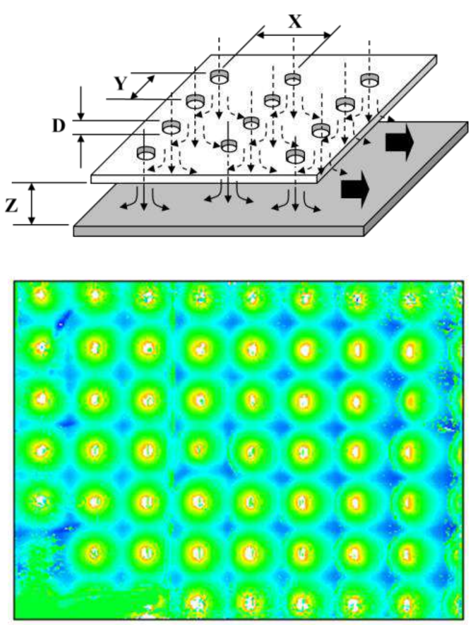


Figure 17. Jet array impingement HTC distribution example (same color representation as Fig. 15)

Figure 18 summarizes the heat transfer coefficient enhancement capabilities for the primary techniques in use today along with the associated coefficient of friction coefficient augmentations [2]. The Nusselt number Nu is simply hD/k, where D is some relevant characteristic length (i.e. jet diameter or channel hydraulic diameter). Nu is then sometimes referred to as a surrogate for HTC. The normalizing values Nu_o and C_{fo} are defined by fully developed turbulent flow and heat transfer in a duct. The reason for this comparative format is that every cooling augmentation comes with a penalty, namely the loss of coolant pressure expressed as a friction coefficient. Expressed with respect to the principle of Reynolds Analogy (i.e. the equality of turbulent thermal and momentum diffusivities in a boundary layer [60]), the great majority of practical methods cause higher friction augmentations than heat transfer coefficient augmentations. This is primarily due to the nature of these methods, meaning that most rely on complete disruption of the boundary layer by flow obstructions of the same order of size, or greater, than the boundary layers. This is critical information because the use of too much pressure drop could lead to local failures, or the potential for failure, in the form of hot gas ingestion into the components. To clarify this point further, commonly published optimizing parameters seeking the highest HTC for the lowest pressure loss are myth when it comes to actual component design. A design must satisfy temperature limitations, but this does not mean that the highest HTC is always desirable, in fact the generation of higher thermal gradients is frequently equally detrimental to high temperatures. A design must also guard against gas ingestion, or backflow, at any location on the HGP, but use of the lowest pressure loss mechanism usually leads to inadequate HTC. Also, to balance internal cooling with good film cooling, it is sometimes desired to burn excess pressure. So it is never a simple case of optimizing a function of HTC/Cf based upon model tests. The full extent of each design must be taken into account.

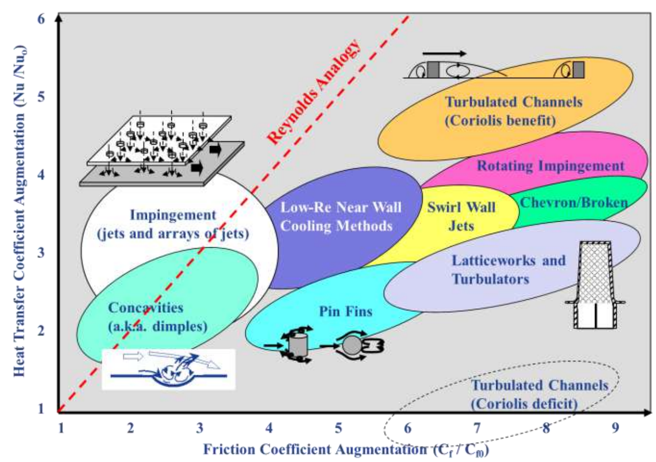
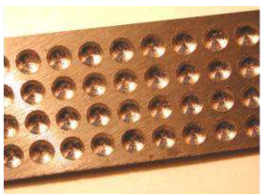


Figure 18. Comparison of various internal cooling SOA methods

Several things should be noted about this summary. First of all, nothing is said about natural enhancements that abound within cooled components. Natural enhancements include mild turns, 180-degree bends, accelerating or decelerating flows, suddenly contracting or expanding passages with flow separations and attachments, and more. These are all special internal cooling configurations that generally rely on the broader literature for heat transfer knowledge rather than only gas turbine literature. These enhancements can be significant, but are also usually localized effects. More important is how these natural enhancements combine with the methods shown in Figure 18. For this last point, there are few answers in the SOA beyond simplified cases of turbulated serpentine bends, because virtually all configurations are unique in design. Testing becomes a requirement.

Second, some of the methods shown in Figure 18 may not be familiar even to seasoned cooling designers, nevertheless they are SOA. These include the so-called swirl or vortical flow methods of surface concavities, wall jets, tornadic flows, and some forms of latticeworks. These methods were well developed and used in the turbine designs of the Former Soviet Republics [61, 62]. These methods do not rely on material flow obstructions, but instead induce organized periodic vortical flows that transport the cooler bulk fluid to the walls. Figure 19 shows the mechanism of self-organized vortical flow developed as a bulk fluid moves over a generally hemispherical concavity. Flow swirls inside the concavity and expels a somewhat unsteady vortex that penetrates the bulk flow. Arrangements of arrays of these concavities in varying geometries and depths provide a wide range of HTC enhancements [63-75]. Figure 20 shows just two examples of wall jets in the cooling of airfoil leading edges. Rather than impinging on the surface, the wall jets create a tornado-like flow of intense Reynolds number on the surface of interest [76-79]. Figure 21 depicts the basic design of latticework cooling in which two alternating layers of cooling channels are open to the flow. At boundary walls the fluid is forced to switch directions and move to the alternate layer, thereby creating high augmentations from turning that last a considerable distance [80-82]. Variations of latticework geometries have been established for many differing regions of the components [61].



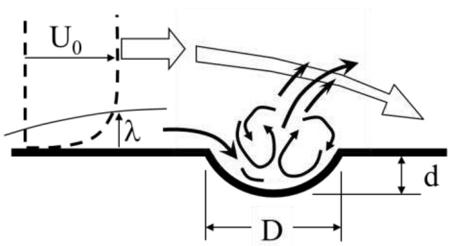


Figure 19. Self-organized vortical flow generated by surface concavities

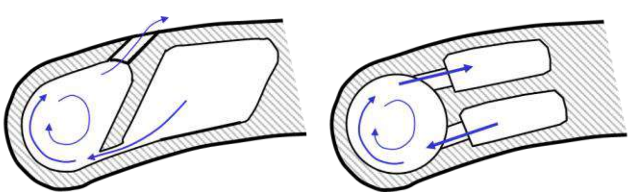


Figure 20. Examples of wall jet or tornado swirl flows

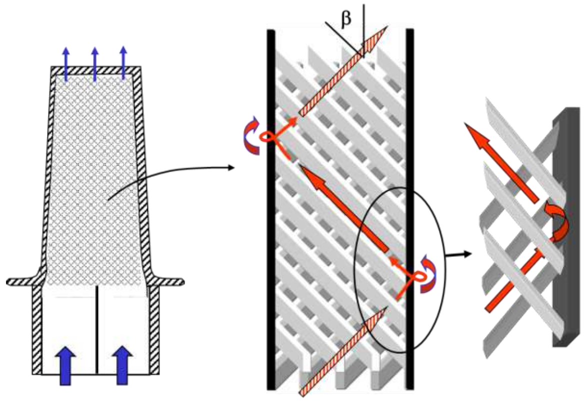


Figure 21. Basic structural and flow path form of latticework cooling

Third, the subset of enhancements dealing with rotating blade cooling clearly stand out as both the best and the worst in magnitude due to the additional factors of Coriolis and buoyancy forces [6, 83, 84]. Rotational effects represent a small but important portion of the SOA for turbine blades, but the overwhelming majority of research in the topic concerns turbulated channels only. Methods such as impingement and latticework cooling have been shown to resist to a good degree the extremes present in turbulated channels. Fourth, while impingement is relatively devoid of frictional losses, it must be remembered that a significant pressure loss is required at the jet orifice. Lastly, while the SOA contains an impressive array of internal cooling methods, not all methods are suitable to all components or compatible with other requirements. As noted by Figures 10 and 11, certain methods find homes in nozzles and others in blades as dictated by the many relationships in Figure 12.

With the advancements in materials and manufacturing technologies of the last decade, a drastically larger realm of cooling enhancement techniques has become cost effective for conventional use in the cooling of turbine airfoils as noted in Bunker [85]. More will be discussed on this later.

Film Cooling

The image in Figure 9 depicts an idealized two-dimensional layer of coolant film being injected tangentially on the HGP surface. The image of Figure 22 shows discrete film holes injecting coolant at an angle to the surface; the colors represent differing species concentrations as coolant and gas mix [86]. The ideal condition is achievable under very restricted conditions of slot flows, and will still result in fluid mixing and effectiveness decay for the dominant turbulent flows in turbines. The use of discrete film holes represents the majority of the SOA, due to material structural concerns (e.g. stresses) and manufacturing limitations. Actual design, as noted by Figure 12, must account for the real conditions through the complete cooling analysis of each component.

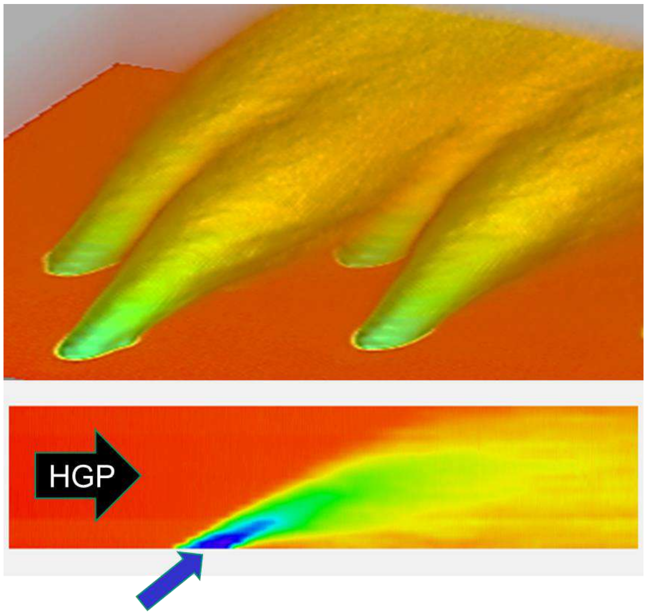


Figure 22. Discrete [round] film hole cooling [86]; three-dimensional coolant jet images above, and centerline cross-section below

As indicated by the images of Figure 22, discrete film hole cooling has a significant degree of complex, three-dimensional fluid mixing that is initiated immediately at the coolant exit location. A jet in crossflow is the typical fundamental flow visualization. It is a universal SOA truth in actual HP turbine environments that a complete understanding and quantification of film cooling cannot be achieved for all possible parameter combinations, and perhaps not for any practical turbine conditions. An appreciation of this view point comes from even an incomplete list of the parameters involved, listed in Table 2, all compounded by the fundamental issue of turbulent flows. The factors are noted as being associated with the HGP or the cooling film holes, but are intimately co-dependent in very complex ways. This dependency is what lends so much diversity to actual film cooling designs throughout the turbine, and consequently results in unique behaviors for virtually all locations and scenarios.

Table 2. Factors Influencing Film Cooling Effectiveness

HGP Factors	Film Hole Factors
Stagnation regions	Directional injection
Local Mach number	Horseshoe rollup vortex
Transitional boundary layers	Film hole geometry/shaping
Local boundary layer thickness	Film hole spacing
Destabilizing concave surfaces	Film hole size
Re-laminarizing flows	Film hole orientation
Separation and/or shocks	Film/gas blowing ratio
Freestream turbulence	Film/gas momentum ratio
Carryover film cooling	Film/gas density ratio
Leakage flows	Film/gas velocity ratio
Film migration	Hole inlet flow effects
Vortex scrubbing	Turbulence generation inside holes

To gain even a small appreciation for the distributed surface nature of film cooling, Figure 23 shows an infrared image of the surface temperatures for a well behaved row of diffuser shaped film holes on a region of a model turbine nozzle with essentially flat surface and adiabatic condition (flow is left to right). Each film hole exit footprint can be seen, and also the warmer surface regions between the film hole exits and downstream between the coolant streaks. The film effectiveness distribution will follow that of surface temperature since the gas recovery temperature in this case is essentially constant. Super imposed on this image is a representation of the laterally averaged adiabatic film effectiveness, starting from a relatively high value and rapidly decaying with non-dimensional distance X/D downstream. The density ratio, or inverse of temperature ratio, of coolant exiting the holes to hot gas is about 2 in this example, much as in an engine. The coolant jets are at an angle of 30 degrees to the surface tangent. Though all of the holes are nominally cleanly drilled in a plastic model, differences in the coolant distribution are still apparent in the thermal image. All in all, even this simple example is not so simple after all.

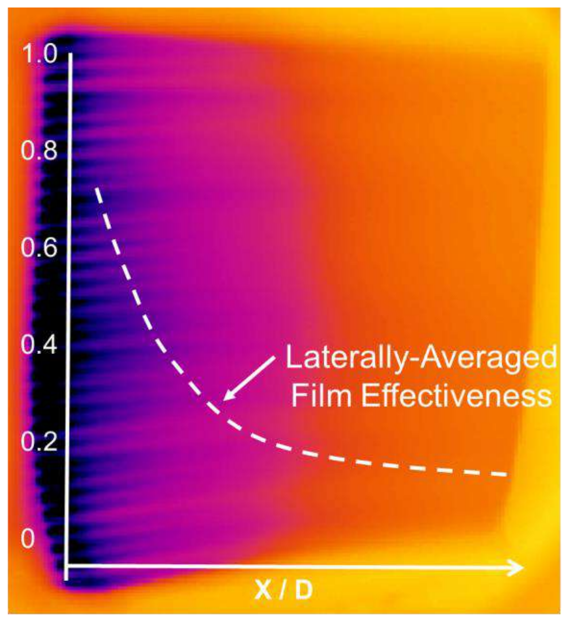


Figure 23. Example of local [adiabatic] film effectiveness distribution superimposed over surface thermal map (black holes are coolant)

The behavior of most well defined film rows, as opposed to individual holes and groups of holes serving more as local bore cooling (through-wall convective cooling), can be characterized with the help of two parameters comparing coolant conditions to gas conditions at the point of injection. One is the mass flux ratio of coolant to gas, more commonly called the blowing ratio M, and the other is the momentum flux ratio I, defined as

$$M = (\rho V)_{coolant} / (\rho V)_{gas}$$

$$I = (\rho V^{*}V)_{coolant} / (\rho V^{*}V)_{gas}$$

Figure 24 shows an example of centerline (center of row rather than laterally averaged) adiabatic film effectiveness for two rows of axial holes on an airfoil, one on the pressure side and the other on the suction side [9]. Each film row has several sets of data varying M, plotted against another commonly used parameter, x/Ms, where 's' is the equivalent two-dimensional slot width for the film row. This parameter is derived from theoretical considerations concerning the relationship between the real film row and the ideal slot [9], and is related to a ratio of the gas and coolant Reynolds numbers. As Figure 24 indicates, the film "curve" behaviors tend to begin at a peak value near the injection point and asymptote to some slope far downstream. The peak level of effectiveness can be correlated in most cases to I, since this is the dominant parameter affecting the initial jet penetration into the gas. If I is too large, then the coolant jet will completely separate from the surface and may fail to re-attach, causing low effectiveness. If I is too small, the coolant may be dispersed quickly by the gas and cause a rapid decay of effectiveness. Hence, a moderate momentum flux ratio is generally desirable. Far downstream the dominant correlating parameter is blowing ratio M, in which the amount of coolant to be mixed with gas is most important. Higher M leads to higher downstream effectiveness as long as the jet can be kept from complete separation at injection. A proper balance between M and I for any given film row and location is indeed a tricky matter. In many instances, such as large scale freestream turbulence relative to the film boundary layer, a good balance may be quite impossible.

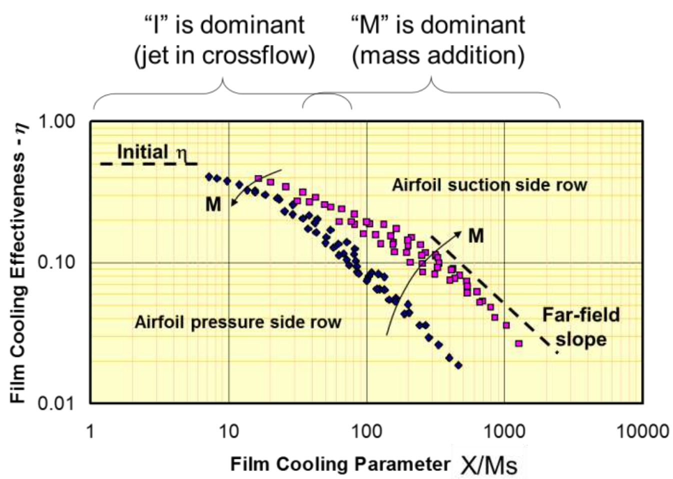


Figure 24. Typical film curve behaviors

Though the task of full characterization looks impossible, it is also true that film cooling represents the most effective means of cooling most turbine components today. The reduction of the effective hot gas temperature at the surface serves to reduce the entire convective heat flux load that the cooling design must accommodate. As a result, the research publications devoted to film cooling from the earliest date to the present total about 5000. But has the actual SOA in film cooling advanced? Figure 25 attempts to condense the history of film cooling deployment in turbines along-side a few major research milestones [9], charting the rise in effectiveness with time. Overall effectiveness in this plot is a relative and approximate surface average value only. While early research focused on the more ideal forms of tangential slot and porous plug film cooling for the purposes of rocket nozzle cooling and flame tube cooling, serious work on turbine film cooling using discrete holes did not begin until the 1970's [87-89]. The basic and still most foundational advancement from round holes to diffuser shaped holes was known, albeit under fairly benign test conditions, in the early 1970's [90]. Implementation required manufacturing advances and these did not mature to commercial use until about 1985. Advances in film cooling beyond this time were driven by the objectives of performance improvements (e.g. SFC), emissions reduction, and lower costs.

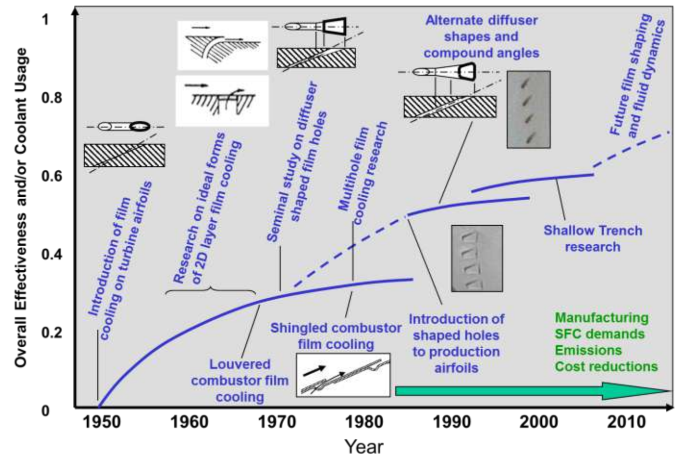


Figure 25. Film cooling development leading to SOA

But more than a history, Figure 25 also represents the current SOA for film cooling. Slot film cooling was and still is only employed in some combustors, and in natural slot-like situations such as the interfaces between components. More importantly, as delineated in [91], the basic diffuser shaped film hole remains the most robust and reliable improvement in film cooling to this day. The reasons are clear, that a properly manufactured diffuser, which is within certain flow physics limits defining stable diffusion in turbulent mixing environments, provides a reproducible and effective layer of coolant that is more resistant to separation over a broad range of M and I. Actually achieving this outcome under the multitude of turbine conditions, unsteady flows, and manufacturing constraints presents a constant challenge and new learnings.

As a consequence, SOA film cooling is limited to variations on the basic diffuser film hole, as shown in Figure 26 [92]. These variations include diffusion in both the lateral direction and into the surface (A), diffusion in only the lateral direction (B), diffusion only into the surface (C), and conical diffusion (D). Round film holes are still very much a part of the SOA, as there are many locations where exit shaping is simply impractical or of no substantive benefit. Consideration of improvements on diffuser shaping will be deferred to a later section. Given the enormous extent of a full film cooling bibliography, the reader is directed to the summaries noted above and the reference lists contained therein. One significant, self-consistent set of studies [93-101] is noted here as it represents a good source for multi-parameter effects concerning basic diffuser shaped film holes used in the SOA. The reported research extends from adiabatic film effectiveness to include the associated effects of film injection on discharge coefficients and HGP heat transfer coefficients.

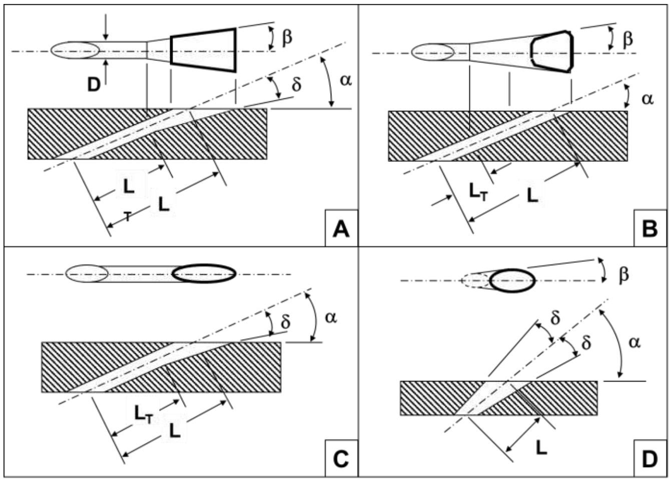


Figure 26. SOA diffuser shaped film holes

To complete this very brief summary of SOA film cooling, the augmented HGP surface heat transfer coefficients must be considered. An ideal slot injection of film cooling, if of sufficient size relative to the upstream or approach boundary layer, would present a full boundary layer restart, as well as a very high surface HTC that decays going downstream (i.e. like a sharp edged flat plate boundary layer). But since the majority of film cooling is achieved with sets of discrete film rows and holes, there is no simple full boundary layer restart. However, as the injecting jets do present varying degrees of disruption to the local boundary layers, there is an augmentation to the HTC on the hot gas path surface. Figure 27 shows a compilation of data for diffuser shaped film rows from several sources indicating that the local HTC augmentation depends at least on the relative approach boundary thickness [to the film hole diameter] and the blowing ratio M [92]. In most instances, the augmentation increases HTC, which then offsets some of the benefit of the film cooling. When the approach boundary layer is relatively small, the HTC augmentation is high and extends over a surface well downstream. When the approach boundary layer is relatively large or similar to the jet size, the HTC augmentation can actually be less than 1. But the larger point to be made here is that the approach boundary layer conditions for any particular location on a component can have a significant effect on the resulting HTC and heat load. By extension, the same can be said for the film effectiveness, yet until very recently the approach boundary layer conditions have not been part of the research and discussion concerning film cooling. Research using full airfoils and components in properly simulated environments goes a long way to alleviating this knowledge gap, though even so the conditions will not be correct without a full combustor-turbine flow field. The SOA understanding of film cooling unfortunately rests primarily on the implicit assumption that approach boundary layer conditions are not as important as the jet injection itself.

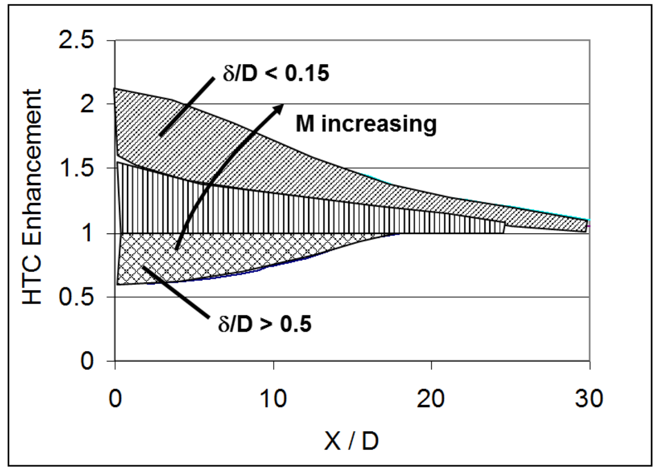


Figure 27. Film injection augmentation of HTC [92]

Feature-Based Cooling Design

The SOA is today at the point of having enough design boundary condition data, e.g. heat transfer coefficients and film effectiveness, which when combined with engine validation data, allow the majority of the boundary conditions for a new design to be predicted from correlations and modeling, and/or computational fluid dynamics (CFD). As such, turbine cooling design is sometimes treated as a feature-based exercise. This means that a designer might pick a certain design for a row of film holes, place that "feature" at a location on the component, and in short order have an updated prediction of material temperature distributions.

The description just given is loaded with challenges and assumptions, and is made somewhat tongue-in-cheek. At first glance, the computational capabilities (i.e. speed for productivity) are very tempting for the use of vast raw databases in conjunction with methods such as Bayesian hybrid modeling to rapidly project outcomes in design spaces where no data exists. It is a virtual guarantee that any new design, or even a limited modification to a design (e.g. adding a film row), will at best only be approximated by existing data, and at worst will bear no resemblance to existing data. Yet numerical tools are promising to fill the gaps and reduce or eliminate the need for validation testing, i.e. jump straight to engine verification. The same can be said about detailed CFD predictions, at least that the speed and content of the predictions are highly tempting for design productivity improvements. Prime reliant accuracy, however, should be a matter of great concern.

This author subscribes to the belief that no prediction should be accepted as correct without validation data obtained through experiments or testing. To put this another way, no numerical prediction should ever be called "data". Considering the previously noted list of parameters just for film cooling, it is true that validation data will not be obtained for all possible scenarios. This is not required however. The best possible outcome is that each feature, and its actual new or modified use in the context of component cooling design, be validated by testing. The risk in moving to fully feature-based cooling design of HGP components lies in what is unknown about the flow and thermal physics. This is compounded by the fact that most feature testing is performed under vastly reduced lab conditions, not at engine conditions, and the transfer function from one to the other is replete with assumptions. The SOA in deciphering the knowns and unknowns in turbine cooling is ultimately provided by the engines themselves. What do the engines teach by virtue of distress and failure, and can it be interpreted correctly? In this author's opinion, feature-based cooling design should remain within the precincts of preliminary design rather than being extended to detailed design optimization. Even so, since preliminary design is where the cooling architecture is decided, the CFD and modeling tools do have a valuable role to play in focusing cooling methods and options down to a select few for validation.

Materials Effects on Turbine Cooling

The turbine cooling designer does not normally get a vote in the material used to form the component substrate, nor the coatings, but does have direct input or recommendation over the thickness and location of "thermal" coatings such as TBC. Current SOA is composed entirely of metallic turbine HGP parts, though that will be changing in the very near future. For the present discussion, most materials related considerations that affect cooling are encountered in the manufacturing processes, and so will be addressed in the following sub-section. Certainly, material properties that affect resulting thermal and mechanical stresses do impact the final design decisions, but are mired in the details of the complete design and so will not be treated here.

There are however several important aspects of materials that the cooling designs must consider. One concerns the effects encountered as a result of drilling film holes through the applied TBC, or alternately the effects of applying coating on surfaces already having drilled film holes. Characterization of actual prototype parts, and preferably also some periodic production parts, in terms of the resulting film hole sizes, shaping, discharge coefficients, and film effectiveness is of high importance. Characterizing the distributions of these quantities even down to each film row or hole is also key. While the great majority of research publications use geometrically clean, nice film holes, these represent only a small fraction of holes used in practice. The use of TBC is a further complication. For parts where the TBC is applied after hole drilling, geometries such as those of Figure 28 may result [102]. This example is extreme for an aviation turbine since it uses a thick bond coat and TBC more appropriate for power turbines. But the visualized effect is clear that coatings do at least partially coat-down inside film holes, and also alter the exit shaping. For both of the film holes shown in profile, the discharge coefficients and film effectiveness were reduced by significant magnitudes. Some coating methods are more benign than others in this respect, but it is common sense to obtain data for all cases. For parts where the film holes are drilled after coating, the issues may include the use of different drilling methods, first to penetrate the TBC and then to drill through the substrate, or the use of a single drilling step through all materials, and consequent non-ideal or even unintended film hole exit geometries within the coating layers. As a ceramic, the reaction of TBC to drilling (e.g. cracking, de-bonding, loss) must be examined by both the manufacturing engineer and the cooling designer, the one concerned with durability and the other with cooling effectiveness.

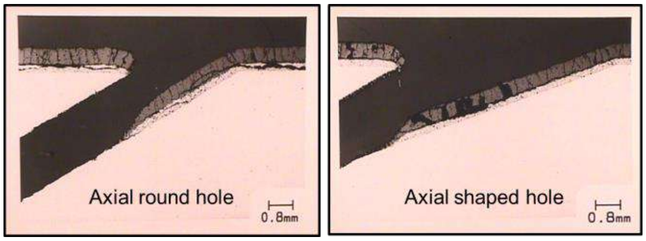


Figure 28. Implications of coatings inside film hole exits

Other materials concerns of note are those that arise during turbine operation and result in modifications to the component geometry which affect cooling. These modifications are all directly or indirectly due to distress accumulated over a number of operating cycles. Virtually all HGP components experience one or more regions of thermal distress (i.e. the region simply got too hot), either in the initial introduction of new designs, or due to changes made elsewhere within the overall system. Components that routinely show little or no distress over their planned lifetime are generally very mature, well characterized parts within engine families having broad operational experience. The catalogue of distressed regions across many engines families, designs, uses, and time is too numerous to recount here, so two recognizable examples must suffice.

Blade tip degradation is common across most designs to some extent. Many contributing factors are involved and intertwined including transient tip rubs that remove material, heat loading that is present on the exposed tip sealing surfaces and the entire tip periphery, wickedly unsteady tip region flows, and the difficulty of cooling the complex geometry under such conditions. The flow and thermal aspects of blade tips has received increased attention in the last twenty years [7, 103] due to the need for greater durability and the loss of turbine efficiency in the absence of durability. Figure 29 shows an example of not atypical squealer tip distress on an HPT blade [104]. It should be stressed that such distress does not constitute a "failure", but does impact SFC as tip clearance increases. The distress exhibits loss of material due to rubs, erosion, and oxidation. In some places, this may alter cooling holes or even make them inactive, further impacting the cooling capability. What should be clear, but is little studied or understood, is that the modified tip geometry will change the heat loading and the film effectiveness, usually in the undesirable direction leading to reduced life. Most designs attempt to provide sufficient cooling to maintain the initial geometry, but few if any deliberately plan for the material changes. In part this is optimism drawn from painful modifications made to other designs after the engine has shown its reality. Also, given the premium on SFC, cooled components are frequently introduced with only as much cooling flow as is felt required, with the understanding that thermally distressed regions will be modified only after the engine teaches the reality.


Figure 29. Example of common blade tip distress

TBC spallation is another fact of turbine life, at least until such time as TBC can be demonstrated to be prime reliant, i.e. will not spall during the intended service interval. Spalling may constitute several degrees of severity. Complete loss of TBC over wide spread areas of a surface is usually associated with some manufacturing or material supply issue. Instances of significant TBC loss are now uncommon and are generally caught in quality control or as infant failures in operation. The SOA concerns smaller TBC spalls associated with either difficult regions for coating application, or extreme heat loading regions on the components combined with interfacial strains. Edge areas are also prone to bond coat attack and TBC spalls if not designed properly. An additional cause of TBC degradation is due to the infiltration of so-called CMAS (calcium-magnesium-alumina-silicate constituents) on the hot surfaces [105]. Small local spalls represent shallow cavities in the HGP surface and depending on their characteristics may alter the heat transfer coefficients and the film cooling, again in the undesired direction. For known or suspect regions prone to TBC spalls, cooling designs should account for modified boundary conditions. When to begin such changes in the life of the part, or how to account for gradual degradation over time, is a matter for probability and experience to decide.

Manufacturing Effects on Turbine Cooling

Unlike the material effects just discussed, which mainly exhibit their impact during turbine operational experience, the manufacturing effects on turbine cooling can be quantified and assessed as part of the design of the components. Every heat transfer coefficient, coefficient of friction, film effectiveness, and discharge coefficient is in reality a distribution about a mean value, having a standard deviation, in which the manufacturing tolerance of some one or more measurable quantities is the variable. For example, investment cast turbulators within cooling channels may range from some minimum acceptable value of height to a maximum value based on the drawing specification for the part. This allows for a maximum value that presents higher heat transfer coefficients, but also means a greater coolant pressure loss, or a minimum value that presents a lower limit on the cooling heat transfer coefficient. The range also allows for wear of the core die over time. Given representative test data for heat transfer and friction coefficients covering the range of turbulator heights, as well as the mean and standard deviation of the manufactured turbulators in sample parts, the cooling design of the component can account for (1) the most conservative outcome in terms of temperatures, sometimes referred to as the worst case, or (2) a probabilistic outcome of temperatures.

There are several challenges in this seemingly simple picture. As should be apparent by now, the number of manufacturing features that might affect cooling is huge. Multiply this by the number of cooling features in the design, as well as the local nature of many (e.g. treat each film hole individually or as a row), and the possible effect distributions quickly become unmanageable. Fortunately, to the best of current SOA knowledge, only certain parameters are key, or dominant, in their effects on turbine cooling. The study of Bunker [106] examined 32 features related to the cooling of turbine airfoils to assess the effects of manufacturing tolerances on component temperatures. The main categories included external aerodynamics, component wall construction (e.g. TBC thickness), film cooling, impingement cooling (e.g. leading edges), turbulated channels, internal cooling passages, internal pin banks, and trailing edge cooling. Typical manufacturing tolerances were set and the effects of each upon the associated key thermal boundary condition were assessed from available literature data. A simplified thermal model for a turbine blade was then used to quantify the effect of each feature and its tolerancing on resulting metal temperature. Figures 30a and 30b show the results in a pareto of the maximum metal temperature increases as delta from nominal for the 27 features that had any effect. Some features, such as film hole geometry, have the potential for much higher thermal impact if tolerances are not controlled, or more positively could help reduce variations in temperatures by using tighter tolerances. Some features have relatively small impacts by themselves, but could become contributors if several feature tolerances were to stack up in an adverse manner.

The key in assessing the effects of manufacturing tolerances is data. Statistical data is required on the physical features being measured, e.g. those determined to matter, and boundary condition sensitivity data is required on each of the affected cooling geometries or categories. The former is SOA today, but the latter is not.

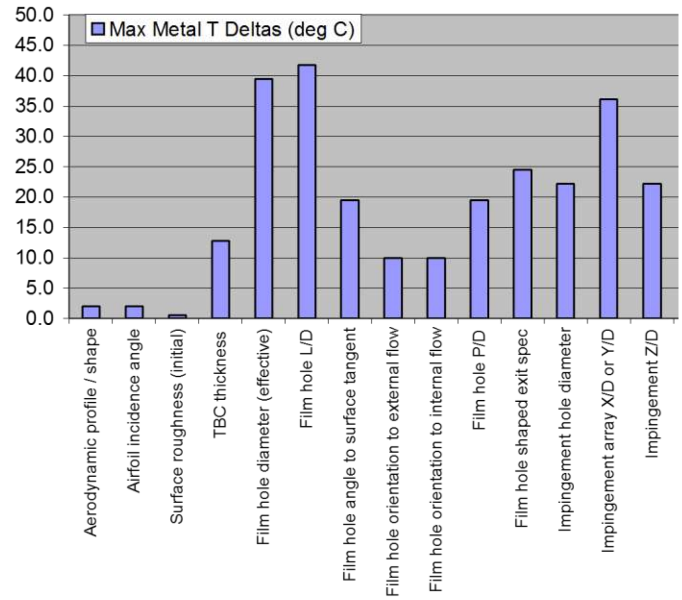


Figure 30a. Pareto of tolerancing effects of cooling

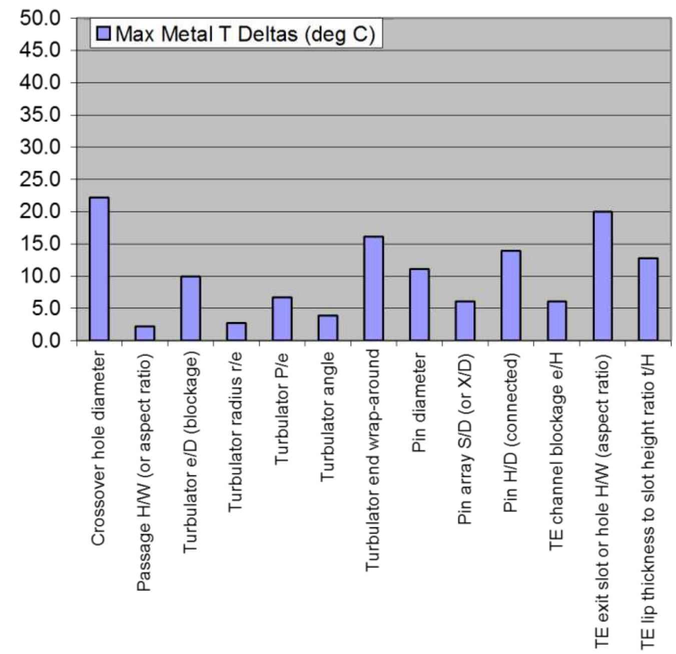


Figure 30b. Pareto of tolerancing effects of cooling (continued)

Summary of SOA

The current SOA in turbine cooling is the culmination of several decades of both development and experience. The actual mechanisms of internal cooling and film cooling, the two main methods in use, have in fact been demonstrated for roughly 45 years now. What has changed over that period of time is (1) an increase in our fundamental knowledge of the cooling methods, (2) an introduction of several variants upon such methods as turbulators, pin fins, and diffuser shaped film holes, (3) incremental improvements in super alloy temperature capabilities, (4) the introduction and widespread use of TBC, (5) the introduction of new manufacturing methods (e.g. laser drilling) and the advancement of investment casting technology, and (6) experience. The change driving all of these advancements has been the constant increase in turbine inlet temperatures plus the demand for lower SFC. Cooling techniques have not made any revolutionary changes, but rather the methods and their many variations have been forced to keep up with components requiring greater amounts of cooling air in order to survive. Testing and experience, applied to the same fundamental methods, have contributed the most over time.

This is not to belittle the vast amount of research and development in turbine cooling. That work has advanced our understanding, thereby allowing those cooling methods to be applied with less conservatism and design margin (i.e. realized SFC gains). Those common cooling methods are the work horses of the designs in operation today because over time they have become reliable, known, and to a certain degree even forgiving. Turbine cooling has been on an evolutionary track, enabled to a great degree by materials and manufacturing. At the present time, SOA has reached to a nearly flat portion of the technology curve plateau in overall cooling effectiveness (Figure 3). SOA includes and continues to use in practice all of the technologies and designs below this plateau. Fundamentally, the heat exchanger design that is a turbine component must next undergo a more revolutionary change to progress further. That is the subject of the next section.

THE FUTURE OF TURBINE COOLING

Before attempting to describe the requirements and possible approaches for the future of turbine cooling, one point must be made clear. The current SOA in turbine cooling must be consolidated and retained in the design practices. Put another way, the design community must not forget the lessons learned from the first 50 years of design evolution and experience. It seems an obvious point, but there is a real danger present in the digital age of fast processing and design turn-around times to cease looking at history. As reliance on computing increases, there is a strong tendency to set aside comparisons with historical experience, to de-emphasize uncertainties and unknowns, and to treat numerical predictions as truth. As the generational change in turbine design engineers takes place (i.e. now), this tendency is strengthened. The task of learning about past lessons becomes an ever larger one, frequently with little time devoted to it.

There are several challenges going forward, over and above those noted in the SOA. Some of these are evolutionary, some revolutionary, and some represent changes in the environment that must be accommodated:

- New engine operating environments and associated new failure modes
- Component and systems integration effects on turbine cooling
- Revolutionary turbine cooling
- Revolutionary manufacturing
- Quantification of unknowns

New Operating Environments

Engine operating environments come in several forms. For the current discussion these include actual atmospheric environment changes, the economic environment of both customers and competition, and the operational environment of how the engines are cycled. All of these have direct impact on turbine cooling.

The most recent challenge to turbine cooling for aviation engines is the increase in dust and particulates in the atmosphere within the so-called dust belt of the world, with predominance from northern Africa across the Middle East and on through China [107]. These regions also happen to be where increased global aviation growth is taking place, and consequently the numbers of takeoffs and landings. Generically referring to “dust”, without getting into the chemistries involved, the particulate matter can range from the sub-micron level up to 100 micron size. This dust differs from the sand and volcanic ash issues documented in prior events like Desert Storm and Mt. St. Helens. Increasing quantities ingested into the engine pose problems throughout, but can especially deposit inside and outside on cooled HGP components, thereby increasing heat loads and decreasing cooling effectiveness. Attention to this issue is apparent in some recent conference publications [108-112], and even more so in recent public patent applications. The impact on turbine cooling, if dust cannot be eliminated entirely from the cooling circuits, is either designs that offset dust effects (e.g. higher cooling flows), or designs that accommodate the degradation effects of dust (e.g. lower life).

The economic environment changes are spearheaded by the use of contract service agreements (CSA) for the guaranteed maintenance of engines in customer fleets. The profitability of engine manufacturers lies in the servicing of engines (i.e. replacement parts and repairs), and many of the highest cost components are those of the combustor and turbine, which are also the lowest life parts. The concurrence of cost and life factors drives intense scrutiny on the reliability of cooled parts, and consequently a constant need to improve life through better and more reliable cooling for new and repaired parts. Complicating matters are issues like dust, which can have very different impacts for differing fleet routes throughout the world. One cooling design may be quite sufficient for North America routes, yet experience a much lower life in the Middle East. These combined requirements mean that it is no longer sufficient to simply design a cooled component for introduction into service as a new part. The life cycle operational conditions must be understood, as well as any in-service changes to the turbine cooling, and any lingering effects on cooled parts after repair.

The changes in regional and world development are also manifested in new modes of operating gas turbines. In the aviation arena, increased traffic in many regions has led to different flight envelopes. For example, what used to be a relatively brief full power takeoff followed by a long climb, may now be replaced by a longer takeoff power period and/or a higher power climb, putting more burden on turbine cooling. Faster descents and landings may also add to the burden. In other words, the definition of a cycle is changing. In the power industry as well, cyclic operation sometimes on a daily basis is now common, with increased demand for very fast start times to reach full power. There is also an increased requirement for grid flexibility with renewable power sources. All of these changes lead to adjustments in the requirements for turbine cooling in order to achieve allowable component life.

Component and Systems Integration

The majority of published research concerning turbine cooling has focused on single cooling methods and variants, or on cooling designs for individual components. By necessity of limited resources, most published work even on complete component models is performed using simulated systems conditions (e.g. high freestream turbulence to mimic a combustor exit condition, or low Mach wind tunnels to look at leakage effects). In general, integrated design and testing between components or within a system has been the realm of the engine manufacturers and government agencies, and as such has had restrictions due to proprietary rights or export control.

But it is in these integrated systems and sub-systems that a great deal of future advancement is to be realized. Examples for turbine cooling abound and can be seen in even a quick glance at Figure 2. These include all major inboard and outboard interfaces between components circumferentially and axially, such as leakages and wheelspace purge flows, and also notably between the combustor and HP turbine. It includes the interactions between airfoil and endwall film cooling, between upstream wakes containing coolant and downstream components, accumulated cooling carried over to downstream surfaces, such as multihole liner film into the nozzle, and more. Some systems effects, for example hot streaks, clocking, and blade tips, have been studied for integration effects mainly with respect to aerodynamic performance rather than cooling performance. Of course there are exceptions in these examples where systems research has been performed specifically for turbine cooling, but too few. The engine is the final arbiter of turbine cooling and so more integrated systems research is required.

Revolutionary Turbine Cooling

Before the development leading to the current SOA, there was an ideal cooling methodology that was held up as the ultimate goal in cooling effectiveness. That was transpiration cooling. Up to the present day, many people still refer to transpiration as the goal for turbine cooling, though there has been no actual proof of this beyond theoretical considerations. Despite the many forms of manufacturing that might produce a transpiration cooled airfoil, and there are at least a dozen, none have been able to satisfy the complete design requirements as set out in Figures 5 and 12. The primary limitation is the strength of final material.

Yet transpiration, sometimes also called porous wall cooling, is a convenient place to begin looking at future cooling architectural changes. Figure 31 depicts the ideal of airfoil transpiration cooling in which air is bled through the structure, not uniformly, but with local metering (e.g. porosity changes) that match cooling needs to the external heat loading. In this way, both the internal cooling inside the walls and the film cooling can be set and balanced for each region [113-119]. Though the ultimate goal may be a uniform temperature part to eliminate thermal stresses, this is not an absolute requirement. The study of [5] provides a brief summary of the history of development towards transpiration-like cooling of airfoils.

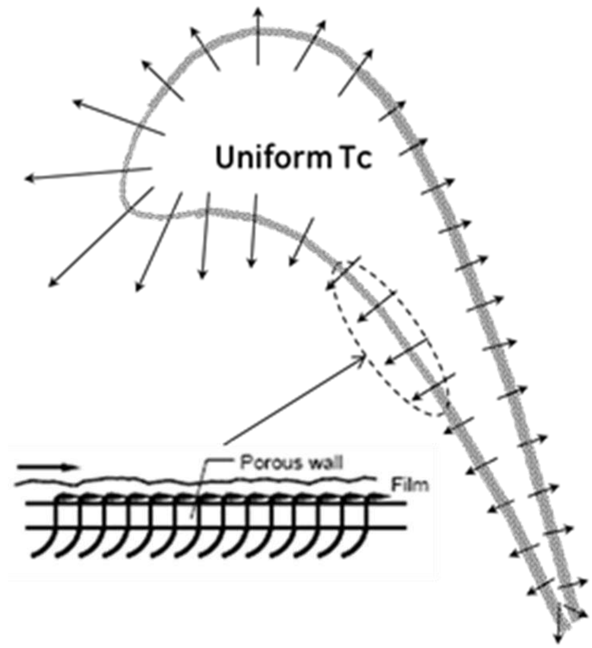


Figure 31. Idealized transpiration airfoil cooling (arrow lengths represent notional local cooling flow strengths)

However, while the many forms of manufactured porous materials have not been suitable for the high temperature, high stress HP turbine components, there are still several intermediary cooled architectures that may achieve a portion of the ideal, or even become a stepping stone to the ideal with help from manufacturing. The current roadmap for future turbine cooling is shown in Figure 32. This roadmap is presented in a very general format without attempting to show specific or preferred designs. In fact, just as with the SOA, there are many design alternatives that will satisfy each type of cooling architecture. Starting from the current SOA, two branches are shown, one a low risk and the other a high risk, where risk refers to the as yet unknown factors of development cost, production yields, durability, reliability, and life.

The low risk branch includes enhanced SOA and double wall hybrid architectures. Manufacturing of the low risk approaches may be according to present or improved investment casting, the use of additive manufacturing to produce intermediate cores for casting, or eventually the use of direct additive methods to produce the end components; the next section will say more about additive manufacturing. Enhanced SOA designs will be evolutionary in nature, accounting for many of the factors previously discussed to incrementally improve cooling effectiveness and reduce design margins. A good goal for enhanced SOA cooling is a 10% reduction in the amount of cooling air required. Much of this reduction will be derived from better understanding of the operating environments and good statistical data built from experience.

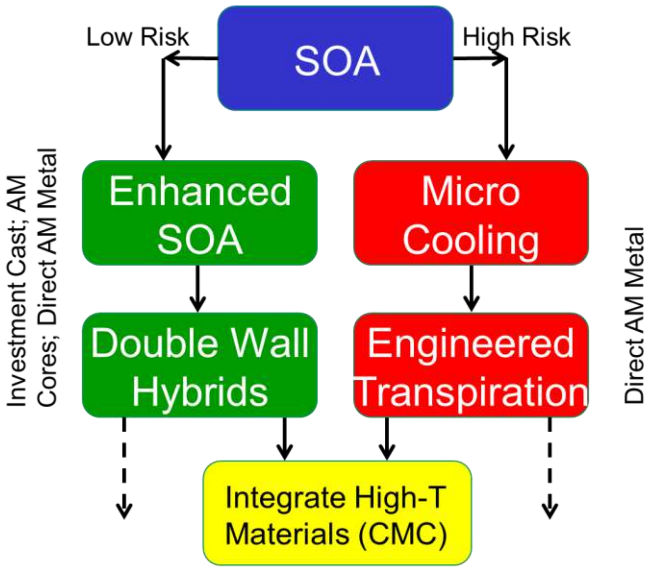


Figure 32. Turbine cooling roadmap looking forward

Double wall hybrid cooling, also known as near wall cooling, places the conventional methods of cooling inside the component walls, or between inner and outer walls as shown in the example of Figure 33 [5, 120]. Patent literature, both applications and granted patents, is rife with examples of double wall structures for HGP parts, some more fanciful than others. There is also quite a range of physical cooling passage or sub-chamber sizes discussed in literature, some of which border on micro cooling. In all cases though, the intention is increase the cooling effectiveness by moving the coolant sink closer to the heat source, and by improving the distribution of cooling within the part, i.e. a step towards transpiration-like. The inner structure remains close to the coolant inlet temperature and can therefore withstand higher stresses than the warmer outer wall. There are exceptional regions, such as airfoil trailing edges, that cannot employ this architecture. Overall, double wall cooling may be expected to achieve a 15 to 20% reduction in coolant consumption relative to the SOA.

The high risk branch of Figure 32 includes micro cooled and engineered transpiration architectures. Manufacturing of the high risk approaches will almost certainly be through the use of direct additive methods to produce the components prior to finishing processes. Micro cooling is distinguished from double wall cooling in several ways, although the sizes and manufacturing ability are somewhat subjective. A rule of thumb is that a micro cooling passage is 1 mm or less in hydraulic diameter. Micro cooling passages are typically individual passages, or a network of passages, rather than two walls separated by a gap and connectors. As detailed in [5], micro channels may be interior to the substrate wall, or for maximum cooling effectiveness may be in the surface of the substrate, perhaps even in the coatings. These latter versions are sometimes known as skin cooling by analogy to how blood vessels cool our skin. Figure 34 shows a cross section photograph of an example of skin cooling in which the cooling channels are directly covered by a coating, leaving film cooling slots as a byproduct of the manufacturing method [121]. The manufacturing approach is in fact what separates the several forms of micro cooling, with an incomplete list of processes that includes refractory metal cores, abrasive liquid jets, tomolithography, and etching.

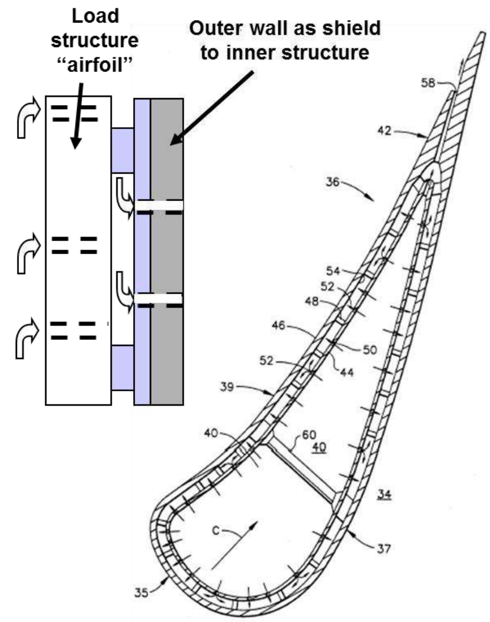


Figure 33. One example of double wall cooling

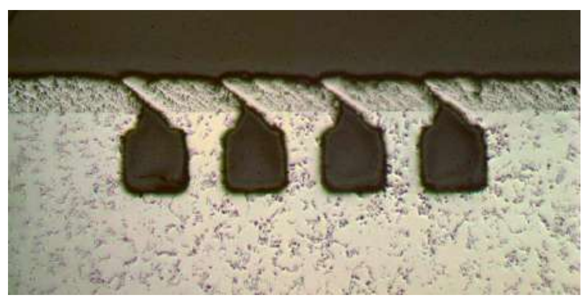


Figure 34. Example of skin cooling

Micro cooling has distinct advantages, at least conceptually. It allows for a flexible cooling design that places coolant only where it is required and in the strength desired. It also represents a “tunable” form of cooling that can be modified more readily to adjust for what the engine teaches us. Micro cooling in formats like that of Figure 34 can be extremely simple in terms of fluid mechanics and heat transfer, as basic as laminar or turbulent flow in smooth ducts. Some manufacturing methods for micro cooling are divorced from investment casting with consequent savings in hard tooling. The potential for coolant savings, as noted in the analysis of [5], are in the range of 25 to 30%, also with respect to SOA.

Engineered transpiration is the marriage of the intent of porous wall cooling with the manufacturing power of additive methods. But rather than designing or producing randomly porous walls, the additive methods promise to allow cooling apertures and “channels” of almost any desired shape, size, orientation, spacing, and routing, thereby making the wall an engineered and specified cooled structure. This extends to internal features of the component, micro cooling features in the walls, and film cooling. In theory, the cooling geometries are limitless, but in practice will depend on the material properties obtained from additive methods and any post processing. The potential for re-writing and redefining cooling designs also brings with it a host of unknowns that must be thoroughly researched. To name just one obvious example, how will new film cooling "holes" behave when no longer restricted by conventional design rules? The goal for this advanced turbine cooling is roughly a 50% reduction in cooling flows.

The final advancement, for now, noted in Figure 30 concerns the introduction of new HGP materials, such as ceramic matrix composites (CMC), monolithic ceramics, and metal matrix composites (MMC). Composites and ceramics represent new materials for the HGP with increases in temperature capabilities several hundreds of degrees beyond current super alloys. Such materials are not "new" in the sense of research and development, but are new to production HP turbines. These new materials bring other challenges however, such as much lower strength, forcing another re-balancing of the overall design of each part. New materials can be used with current SOA cooling methods, but may benefit even more from incorporation of the future cooling methods, if feasible. Even simple SOA tasks such as drilling film holes, will at a minimum require re-examination and re-characterization for these new materials. It is beyond the scope of the present summary to dive deeper into this area, however, nearly all aspects of cooling will need validation and verification again.

Hybrids of the several architectures of Figure 32 may also be feasible. For example, as shown in Figure 35, the capturing and bonding of non-structural porous materials within double wall or micro cooled formations [122] may serve to meld cooling and strength together. Another example would be the possible integration of micro cooling into CMC structures, which as matrix and impregnated layered formats would seem natural for this cooling technique [123].

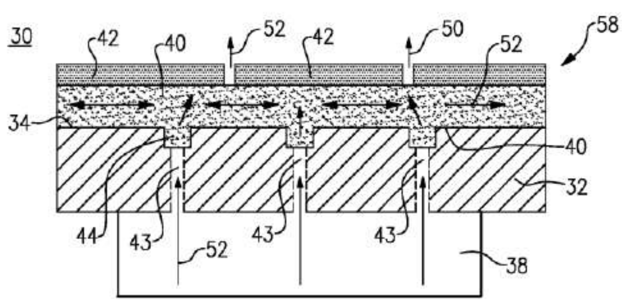


Figure 35. Example of hybrid near wall cooling

Finally, no matter how far these future cooling methods may reduce the consumption of engine air, there will also be a need to control the flow of coolant, or to adapt the flow of coolant. Control refers to the use of one percentage of compressor air under high power conditions (takeoff and climb), with the ability to reduce the percentage for low power conditions (e.g. cruise). Passively cooled HP turbines, meaning the SOA, rely on fixed metering orifices at various points in the flow supply circuits and the components. This is great for knowing what range of flow is allowed, but it cannot turn the wick up or down with the cycle point. Adaptable cooling is control for another purpose. Generally this means the increase in cooling flow in response to some degradation in conditions within the engine, for example component issues, changing operator demand, or harsh environmental conditions.

Revolutionary Manufacturing

As noted earlier, the advancements in turbine cooling have come with parallel work in manufacturing, sometimes the one pushing or pulling the other. The traditional process of investment casting has incrementally improved in core materials, directional solidification, mutli-core green state assemblies, and the use of various rapid prototyping methods for tooling, dies, and cores. The recent revolutionary step within the last decade is in the use of additive manufacturing (AM). Additive methods in general refer to even age old means such as welding. But in the present context, AM encompasses several variants upon forms of layer-by-layer, or even voxel-by-voxel, melting of metals in a controlled and programmable fashion to build a component or repair a component. These variants include selective laser sintering (SLS), selective laser melting (SLM), direct metal laser melting (DMLM), direct metal laser sintering (DMLS), electron beam melting (EBM), and at least another dozen not spelled out here.

Certain non-structural metallic engine parts are already being produced and introduced into service. But as has been amply pointed out, the HGP cooled turbine components have very special and stringent requirements on dimensions, strength, temperature capability, and compatibility. The future view of AM is that many of these parts will be manufactured directly using powder alloys of appropriate chemistry. In one version, the final net shape is directly produced with only the post addition of coatings (e.g.). In another version, the nearly net shape part is produced by AM, followed by post processing, for example the drilling of film holes to more exacting specifications than available by AM. This will be a matter of development of AM capabilities in trade-off with the economics of AM manufacturing. The most immediate challenges for AM of turbine components will be on current SOA designs. While various aspects of AM, such as build directions using current flat powder bed methods, may be overcome, the ultimate proof will still lie in the strength and durability. Even the mundane tasks of building statistical data and models for the resulting AM materials will take years.

Patenting activities are awash with AM methods of manufacture, components using AM, methods of cooling using AM, and even the ever popular "formed by formation using AM". The fever over AM in the turbine community is evident. AM has opened up vast territory at least for dreaming of new turbine cooling geometries, both at the macro scale and the micro scale. But just as material properties must be quantified, so must the changes to cooling boundary conditions. For example, will a turbulator or the interior of a film hole have stepped surfaces, or will processing bring these back to SOA knowledge? Unlike the SOA paradigm of Figure 6, the future must merge the disciplines of turbine cooling and manufacturing as depicted in Figure 36, due to the intimate relationship and capabilities of AM. This assumes of course that AM will develop sufficiently to be used for HGP components.

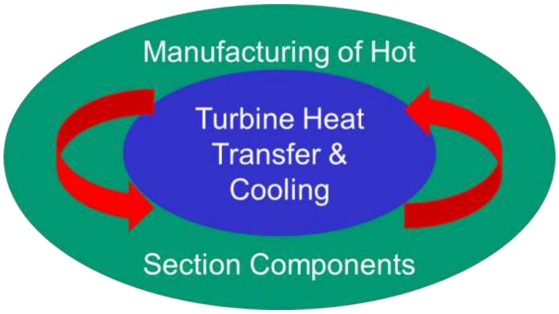


Figure 36. Merging of disciplines for future turbine cooling

Quantification of Unknowns

There are several categories of unknowns concerning any discipline or combination of disciplines in design and execution of machinery. Turbine cooling, including the requirement to quantify the external heat loading, is immediately made complex by the dependency of momentum and energy fundamentals in predominantly turbulent and transitional flows. Knowing this much alone sets a very wide expectation on the unknowns. In this expectation are both known unknowns and unknown unknowns:

- Quantifiable uncertainties on measured cooling (and heating) boundary conditions
- Non-validated scaling of boundary conditions from test conditions to engine conditions (e.g. T, P, combustion)
- Unverified physics modeling (e.g. turbulent Pr number)
- Optimal feature geometries for each flow (e.g. film hole shaping)
- Interpolation and extrapolation of boundary conditions (e.g. between takeoff and cruise)
- Geometry differences between test article features and engine features (e.g. film holes)
- Changeable boundary conditions (e.g. degradation)
- Uncontrollable flows (e.g. highly unsteady)

As with previously noted aspects of turbine cooling, the list of unknowns is daunting but for the fact that turbine operation is not just a theoretical exercise, it is practical experience. As experience, the complete accumulation of unknowns can be at least bounded, though even the setting of upper and lower bounds is in many cases doubtful due to the convolution of parameters, flows, and components. The typical means to combat this convolution is to set certain boundary conditions and make a number of assumptions, then perform a data matching analysis to bring design predictions into agreement with limited engine measurements (e.g. temperatures and pressures).

The practice of data matching side steps most unknowns out of necessities tied to schedule and economics, and in many respects dampens or diminishes the requirement for short term and long term resolution of the unknowns. In other words, data matching to what the engine is trying to teach can lead to complacency and a false comprehension that unknowns have been eliminated. Design must be continuously on guard to recognize and mitigate such false positives.

Some of the unknowns listed above can be reduced enormously through the increased gathering and use of statistical data, whether that means metallurgical evaluations of temperatures from operational parts, optical scanning measurements of film hole sizes and shapes, detailed component flow checks down to the per hole level, or even installed measurements such as infrared imaging of all turbine blade surfaces.

Some unknowns are not immediately amenable to aid from statistical treatment. For example, the flow fields and resulting thermal boundary conditions in highly unsteady flows (e.g. vane wakes on blades, or blade tip gap flows) are such that the unsteady envelope of results is of magnitude equivalent to the average. True, this can still be treated by a time-averaged condition, but measuring that averaged quantity, and therefore also predicting it, is another matter. In cases where the component temperature is insensitive to the boundary condition the argument is mute. But in areas prone to durability issues, such as leading edges and tips, the impact is as great as the unknowns.

Turbine cooling, both SOA and future, has several broad areas that demand significant attention to the reduction of unknowns. To name three key areas of high impact: (1) film cooling, (2) wheelspace interactions with the HGP, and (3) rotational internal cooling. Each of these has received, and continues to receive, significant research resources, but that research is by and large focused on determining the highest magnitude surface averaged cooling effectiveness. Recent research has increased into the more detailed and distributed quantification of boundary conditions, but scant efforts are directed to the reduction of unknowns.

Taking film cooling as perhaps the most impactful topic for overall turbine cooling, Figure 37 shows only some of the many proposed and/or tested film hole geometries seeking the label of best or optimal [91]. While invention is necessary and applauded, it more often than not fails to lead to a solution. The current premise is that this failure is due to the many unknowns, which for film cooling includes at least bridging the divide between tests and engines, unsteady flows, and idealized geometries. The summary of [91] noted critical to success factors including the adiabatic film effectiveness distribution, the manufacturing capability and its tolerances, the ability to repair distressed regions, cost, the sensitivity of film effectiveness to geometry and flow variations or ranges, the allowable operational tolerance, and component strength and durability. Also noted in [91] is that the more successful geometries are those that are robust to both manufacturing variations and flow fields. This is another way of saying that the geometries were less sensitive to the unknowns.

To put this example in a future development and validation/verification perspective, Figure 38 shows a possible roadmap for the entitlement of film cooling. The use of diffuser shaped film holes is the beginning, and the SOA line reflects a status quo. Several examples of proposed film cooling geometry types will probably obtain reasonable evolutionary gains of up to 20% greater surface averaged effectiveness. But the revolutionary advances lie more in the areas of highly distributed and controlled film cooling. Engineered transpiration may be one of these, but this is not assured at this time. The nebulous cloud labeled “Gap” is that portion of entitlement due to those unknowns that may not be overcome (e.g. full knowledge of turbulence), and may therefore remain as conservatism and margin in designs. Reducing unknowns to any significant degree is the most difficult challenge. It is also true that as progress is made, more unknowns are likely to be uncovered.

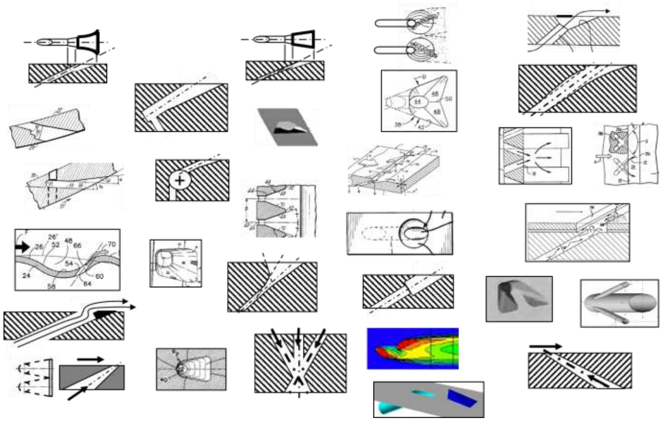


Figure 37. Some of the many proposed film injection geometries

One of the current trends in research is the acquisition of full surface boundary condition data and sometimes time accurate data also, or the predictive equivalent (but not data), for most aspects of turbine cooling, albeit in laboratory models and rigs. This is the direction that design must go in the future. Rather than for example taking full surface maps of film effectiveness, and reducing them to laterally averaged film curves to serve as boundary conditions, the full surface maps must be applied as the boundary conditions. The same is true for all HTCs, internal and external. Eventually, even the temporal aspects must be applied in design, though more likely only in cases where the flow unsteadiness is a significant portion of the total.

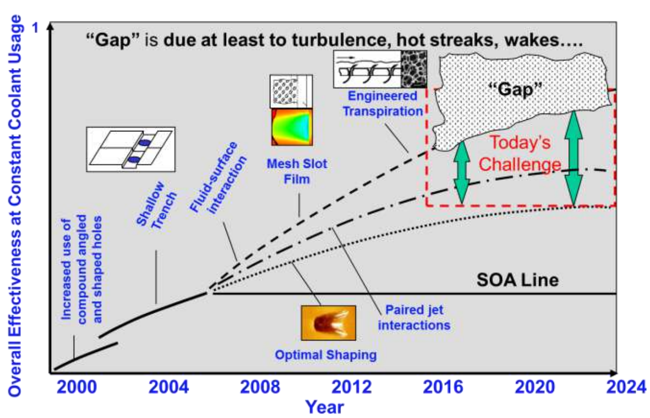


Figure 38. Possible roadmap for film cooling entitlement

CONCLUSION

This summary has attempted to provide a brief history of turbine cooling over the last approximately 50 years, not in a step by step chronological order, but instead reviewing the context and advances that have led to an overall increase in HGP component cooling effectiveness from 0.1 to 0.7. The "evolution" of turbine cooling methods has progressed in analogous though highly constrained fashion to the evolution of animal life, taking twists and turns as variants either succeeded or failed, and coming under the influence of external environmental factors to adapt. Innovation and invention have substituted for random variation and natural branching of species, but the performance of the engine has always dictated which technologies advance. Cooling technologies have generally advanced from the very simple forms to more complex, or enhanced forms, aided by complimentary or required advanced in materials and manufacturing.

The current SOA in turbine cooling has been presented in a summary format, leaving the details of actual correlations and design relevant data to the references. The SOA now contains dozens of internal component cooling methods, most centered on a few bulk forms of cooling such as serpentine channels and impingement, with their many variations. Film cooling in contrast, while seeking innovative geometries and deployments, still relies mainly on only a handful of basic forms that have been known for 40 years. This seeming disparity between advancement of internal and film cooling is appropriate given the heavy reliance on film for overall cooling effectiveness, the need for high reliability and stability, and close dependencies upon manufacturing. Even so, large decreases in coolant usage relative to the total compressor core flow, up to 50%, have been realized over time in the face of increasing TIT. The SOA is now up against cooling design margin limits (e.g. worst case stack-ups) and unknowns to make further advances. Some of the key challenges moving forward from SOA include:

1. Obtaining enough thermal boundary conditions data for full validation of designs;
2. Obtaining and interpreting engine verification while also avoiding false positives;
3. Assessment of changing boundary conditions due to material or engine degradation in operation; and
4. Consistently building statistical data on manufacturing effects (and installation), as well as how these affect cooling.

A future snapshot of turbine cooling is shown in Figure 39. The main areas of greatest impact have been discussed, these being new engine operating environments, component and systems integration effects, revolutionary turbine cooling, revolutionary manufacturing, and the quantification of unknowns. Just as in the last 50 years, the process within any particular technology will be evolutionary. But there are some crossroads present at this time that may provide the next "jump" in cooling technology curves (Fig. 3) with both lower coolant usage and higher cooling effectiveness. One key will be the marriage of design and manufacturing to bring about the concurrent use of engineered micro cooling or transpiration, with the ability of additive manufacturing (top and bottom of Fig. 39). If successful, this combination could see a further 50% reduction in coolant usage for turbines. The other key element concerns the quantification of unknowns, both known and unknown, which directly impacts validation and verification of current SOA and future turbine cooling (left and right of Fig. 39). In this author's opinion, the many aspects of the unknowns is the greatest challenge faced. Without progress in this area, the implementation and realization of future turbine cooling will be limited. Addressing the entire scope of the challenges will require future turbine cooling to be of robust simplicity and stability, with freeform design, much as nature would intend in its own designs.

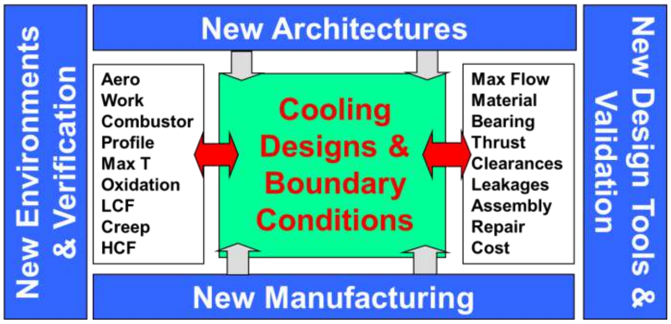


Figure 39. The future of turbine cooling

NOMENCLATURE

A	area
C	degrees Centigrade
Cf	coefficient of friction
Cp	specific heat at constant pressure
d	depth
D	diameter or hydraulic diameter
e	turbulator height
F	degrees Fahrenheit
H, h	heat transfer coefficient, or height
I	momentum ratio
k	thermal conductivity
L	length
M	mass velocity ratio, or jet blowing ratio
Nu	Nusselt number
P	pitch
Q	heat flux
Re	Reynolds number
s	equivalent slot width for a film row
S	spacing
T	temperature
T_3	coolant temperature
T_{41}	turbine firing temperature (also TIT)
U	freestream velocity
U	overall heat transfer coefficient
V	freestream velocity
W	flow rate (%W_c represents the turbine coolant flow rate, or any identified portion, relative to the total compressor core flow rate)
x,y,z	Cartesian coordinates

Subscripts

aw	adiabatic wall
c	coolant
g	gas
m	metal
o	normalizing value
rec	recovery value

Greek

η	adiabatic film effectiveness
λ	boundary layer thickness

ACRONYMS

AM	additive manufacturing
CFD	computational fluid dynamics
CMAS	calcium-magnesium-alumina-silicate
CMC	ceramic matrix composite
CSA	contract service agreement
DMLM	direct metal laser melting
DMLS	direct metal laser sintering
EBM	electron beam melting
HCF	high cycle fatigue
HGP	hot gas path
HP	high pressure
HPT	high pressure turbine
HTC	heat transfer coefficient
LCF	low cycle fatigue
LE	leading edge
MMC	metal matrix composite
SFC	specific fuel consumption
SLM	selective laser melting
SLS	selective laser sintering
SOA	state of the art
TBC	thermal barrier coating
TE	trailing edge
TIT	turbine inlet temperature (also firing temperature T41)

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