

The Importance of Cooling Technology in Propulsion & Power Systems

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Abstract. Turbine cooling is the breakthrough technology for gas turbine engines and although the turbine engine and cooling are considered mature technologies, to date they have only achieved about 60 to 70% of the cycle advantage available. Moreover, great improvements have been made in hot section materials and protective coatings, yet without hot section cooling, the high thrust to weight war fighter engine and the high by-pass ratio turbofan engine would be greatly compromised. Specifically, the level of combustor exit temperature of today's engines which is key for improved cycles would be limited to less than 2300°F using today's best materials and alloys. Air cooled turbine cooling technology utilizes convection in conjunction with film cooling to achieve combustor exit temperatures at least 1000°F higher than the highest combustor exit temperature that could be achieved with just materials alone. Moreover, a stoichiometric turbine would be yet another 1000°F higher. Thus, to continue the trend towards stoichiometric capability, hot section cooling technology will be a key ingredient. The open system air cooling technology is comprised of convection and film cooling. Although convection is near its maximum potential, film cooling still has about 50 to 60% of its potential remaining to be utilized. Thus, improved air-cooled technology will most likely concentrate on improved film cooling capability. In addition, closed loop cooling systems will also probably be revisited.

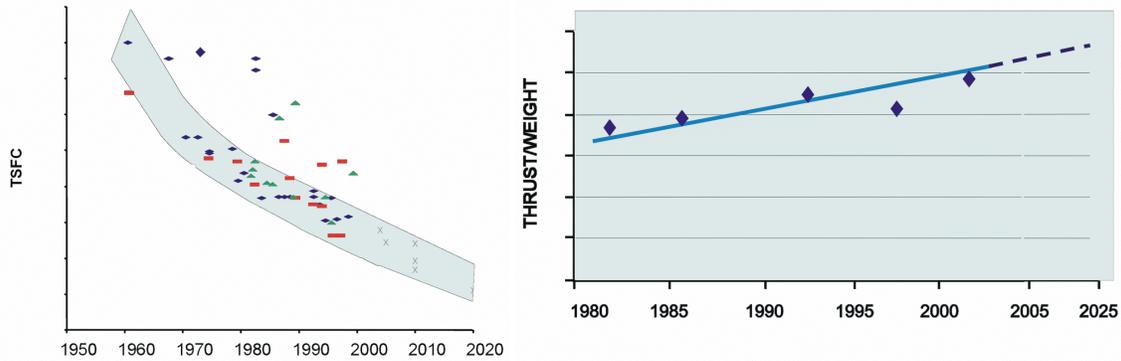
Trends. Turbine cooling technology is the breakthrough technology that is not only used in turbines, but other engine components such as combustors and exhaust nozzles and is the primary technology that has allowed increased engine by-pass ratio (BPR) in commercial engines for reduced fuel burn by 30% since 1970. In addition, turbine-cooling technology has allowed the increase of thrust to weight (T/W) in military war fighters by 2X, again, since 1970. Turbine cooling technology will be the key technology to produce commercial engines with yet another 25% fuel burn reduction and military war fighter engines with another 2X in thrust-to-weight capability in the coming years.

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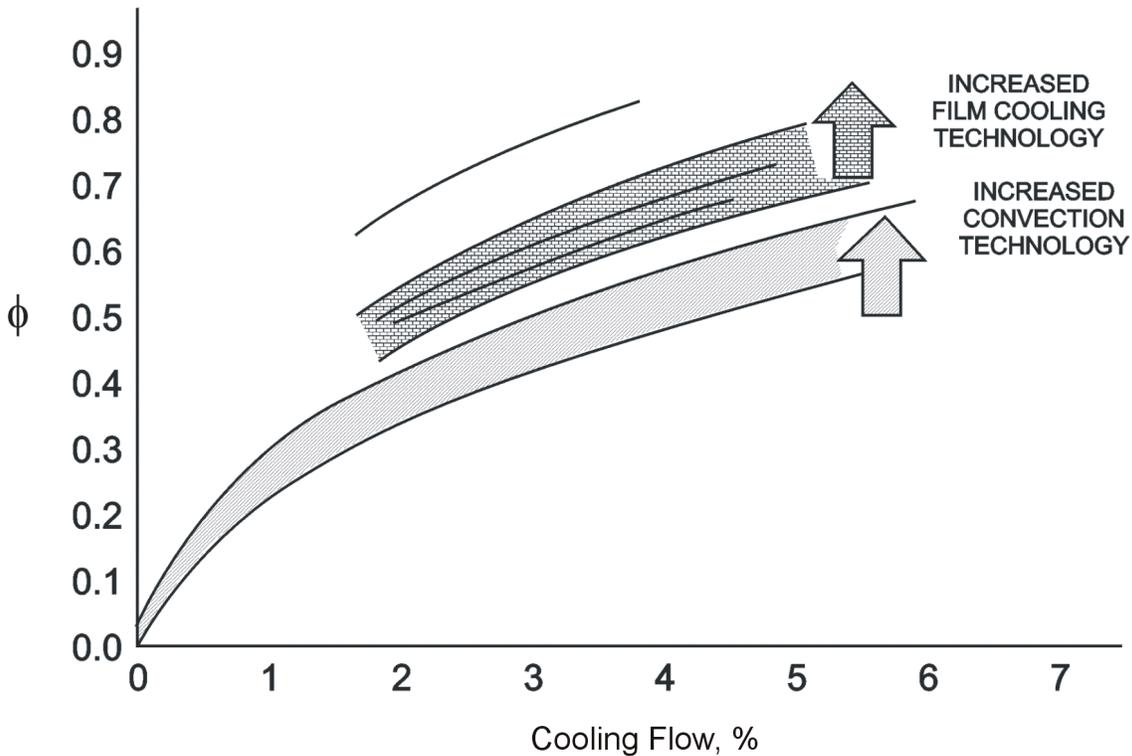
The Commercial Subsonic Engine Cycle. Commercial subsonic engine fuel burn trends indicate that fuel burn has improved approximately 1% per year for the past 30 years. If this trend continues, fuel burn will drop another 25% by 2025. The question is: What does this engine cycle look like and does the turbofan engine have the capability to attain these goals? Propulsor efficiency gains from increased engine by-pass ratio (BPR) and reduced fan pressure ratio (FPR) along with thermal efficiency gains from increased cycle overall pressure ratio (OPR) and component efficiencies are in the direction to satisfy the aforementioned fuel burn forecasts. Similarly, FPR and BPR trends point to levels of 1.3 and 17/1, respectively in these future engines compared to today's engines of approximately 1.7 and 8/1. In addition OPR trends project to levels of 65/1 compared to today's engines of approximately 45/1 and component efficiencies likewise project increases of approximately 2%. The propulsion and thermal efficiency gains will require lightweight structural materials (probably composites) and compressor disk materials and high exit temperature (T_3) capability. Yet, the breakthrough technology will be in better turbine cooling since the rotor inlet temperature to drive these engines will be in the range of 3500°F to 3600°F which is 400 to 500°F higher than today's engines.

The Supersonic War Fighters. As previously noted, war fighter thrust to weight (T/W) trends indicated a 2X improvement over the next 25 years. The cycle is a turbojet type cycle and in order to minimize the engine weight and consequently maximize thrust-to-weight of the engine, the core horsepower per pound turbine gas flow is maximized through increased rotor inlet rotor inlet temperature (RIT). Specifically, these new engines will have RIT values similar to the advanced subsonic commercial engines. The significance is that the supersonic war fighters are tracking toward the same turbine gas temperature requirements and the same cooling technology.

Materials. The capabilities of turbine materials and coatings have also progressed significantly over the past 30 years. Typical single crystal (SC) materials can operate at surface temperatures of 2150°F, which is 200/300°F increased capability compared to early 1970's materials. Thermal barrier coatings have been developed that operate successfully at interface and surface temperatures of 2150° and 2600°F, respectively. However, even with these very significant advances, an un-cooled turbine with the best SC materials and coatings would be limited to rotor inlet temperatures of 2200°F or less. Such an engine built around this RIT capability would only be at 1960's fuel burn and core horsepower capability.

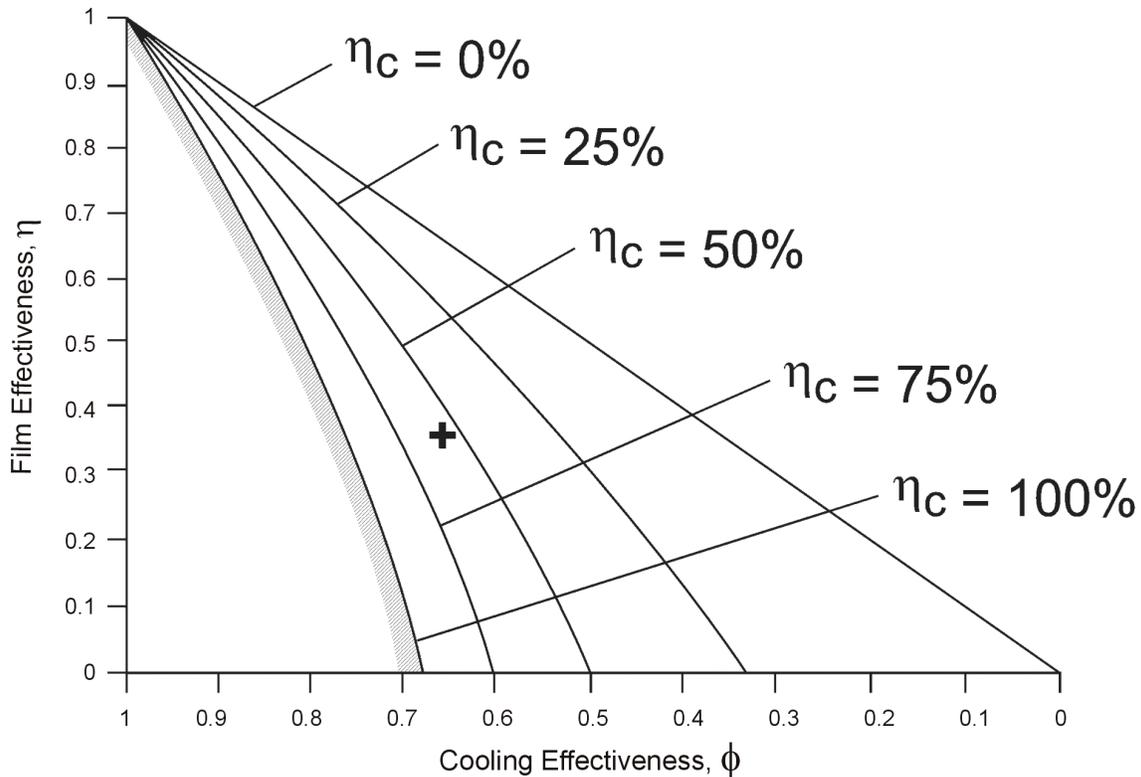
The Cooling System. Consequently, in the 1960's turbine cooling was recognized as game breaker technology and there were two prime approaches: a closed loop liquid cooled convective system and open loop air-cooled system. In the ensuing competition, the air-cooled system proved to be the system of choice. In particular, the source of air for cooling is the compressor core air flow and typically 20 to 30% of this flow by-passes the combustor to cool the high and low turbines. Even though the RIT of today's engines is increasing, levels of cooling flow are maintained or even reduced to minimize turbine efficiency losses. Unlimited cooling flow is not an option since typically at values of 35 to 40%, engine thrust plateaus even though the gas temperature of the engine is still increasing.

Typical Air Cooled Designs. Most modern high-pressure turbines (HPT) are single stage turbines with a high vane and high blade. The typical high vane cooling design is a convectional film cooled design that uses about 10 to 12% cooling flow. This design uses an internal impingement insert for internal impingement cooling along with leading edge, pressure side, and suction side film cooling. The vane trailing edge (TE) is usually cooled with arrays of TE slots that typically exit on the pressure side of the airfoil. These designs have capability to stoichiometric gas temperatures and overall cooling effectiveness, ϕ ($\phi = (T_{\text{gas}} - T_{\text{metal}}) / (T_{\text{gas}} - T_{\text{coolant}})$) levels of approximately 0.8. Most modern vane designs use SC materials with TBC's. The typical high blade cooling design is a single piece casting utilizing convection and film cooling. The leading edge (LE) and TE is impingement cooled with direct feed air while the mid-chord is cooled with a counter flow serpentine heat exchanger. The internal convection cooling is augmented with LE, pressure side (PS), and suction side (SS) film cooling. This design flows typically 4 to 5% cooling air and operates at maximum surface temperatures of approximately 2150°F. Most modern designs use SC materials with TBC's and the overall cooling effectiveness of the high blade design is typically 0.5 to 0.7.

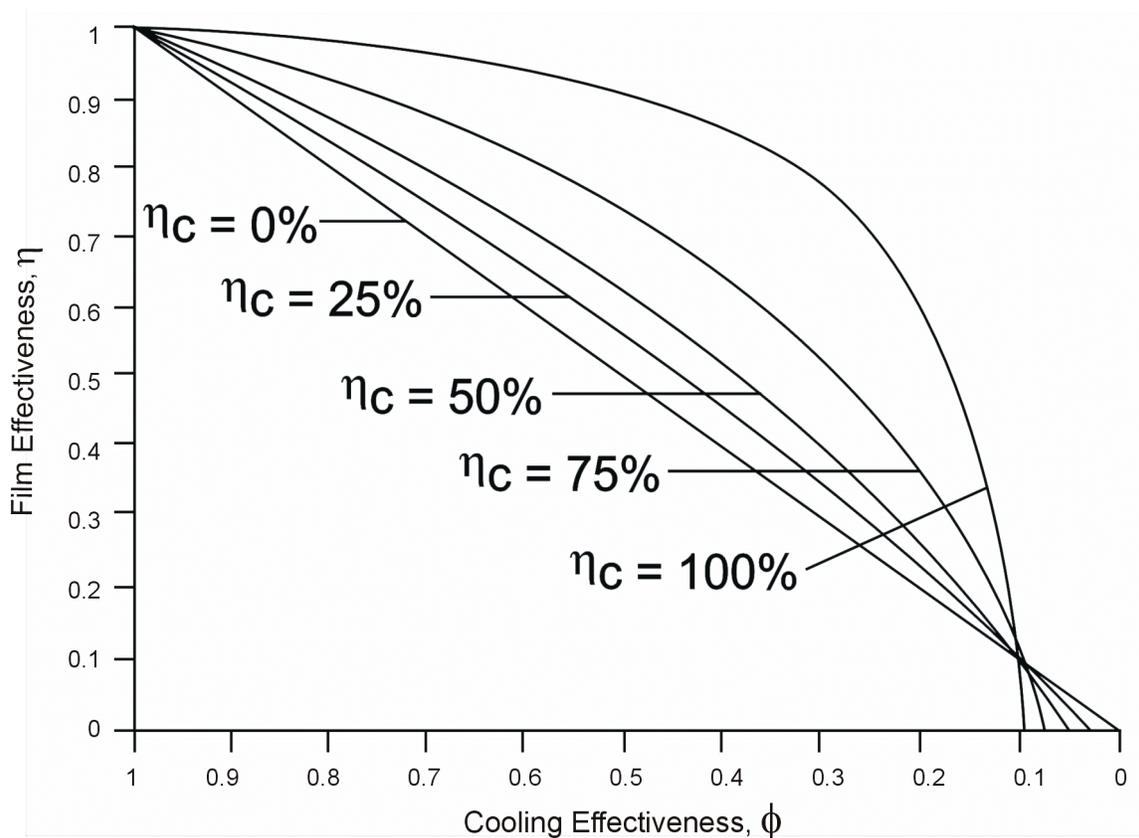


The Cooling Designs. Typical capabilities of turbine cooling designs are compared with cooling effectiveness as a function of cooling flow. Convection only vane designs are limited to cooling effectiveness values of approximately 0.6 at 10% cooling flow compared to 0.8 for a convection/film cooled vane design. The difference in maximum gas temperature capability is about 1000°F. Convection only versus convection/film cooled blade designs at 4% cooling have similar trends.

The Cooling Maps. A cooling map which is typical of a 4% blade is defined as blade surface average film cooling effectiveness (η_f) from 0 to 1 versus blade section average cooling effectiveness (ϕ) from 1 to 0 at levels of cooling efficiency (β) from 0 to 100%. For a typical cooling design with an overall cooling effectiveness of 0.65, β is in the range of 50-80% and film cooling effectiveness is approximately 0.35 to 0.4. It is of interest to note that the main potential for increasing blade overall cooling effectiveness is in film cooling advances as opposed to the convective component since only about 35 to 40% of the total film cooling potential is utilized by the current film cooling technology.



Another Cooling Map. A reduced cooling flow cooling map is defined using the same turbine blade as described by the previous cooling map. Specifically, the blade flow is reduced approximately 80%. The same film effectiveness/cooling effectiveness plot is generated. In a way this is “fun with numbers.” However, for the reduced flow blade, the cooling effectiveness of the blade is actually reduced for increasing convection above a film cooling effectiveness of 0.10. In a more traditional sense, below a film effectiveness of 0.1, cooling effectiveness is increased with increasing convection and at exactly 0.1 film cooling effectiveness, the cooling effectiveness is a constant regardless of the level of convection. The significance of this study is that if film cooling effectiveness could be increased by an order of magnitude over today’s technology levels, with a corresponding reduction in blade cooling flow, designs with this high level of film effectiveness would require insulated convective surfaces to give the highest levels of cooling effectiveness.



Some Concluding Thoughts. Turbine cooling today could be described as a mature technology. Yet the production turbine engine has probably attained only 50% of its ultimate potential. Specifically, there remains about another 1000°F in gas temperature capability to be utilized using today's fuels either by more efficient commercial turbo-fan engines or higher thrust-to-weight war fighters engines. In either case the turbine cooling job will get much more difficult. For instance, at a near stoichiometric rotor inlet temperature the first vane cooling flow must approach zero levels which probably means that if the vane is air cooled it must be cooled with air that is also used for combustion or it must be fuel cooled. In addition, in the future, cycle overall pressure ratio will probably double or triple with compressor exit temperatures in excess of 2000°F. This opens up a whole new cooling adventure were the cooling of turbine engine parts in rotating cavities will take on a similar posture as with the decision in the 1960's to cool turbine flow path parts.

Indeed, the industry has come a long way in turbine cooling technology but I would conclude by saying the hard part is yet to come.