Propulsion and Power Research 2018;7(1):1–18

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Propulsion and Power Research

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ORIGINAL ARTICLE

Review on film cooling of liquid rocket engines

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Received 13 October 2015; accepted 21 March 2016
Available online 13 March 2018

KEYWORDS

Heat transfer;
Liquid rocket thrust chamber;
Film cooling;
Cooling effectiveness

Abstract Film cooling in combination with regenerative cooling is presently considered as an efficient method to guarantee safe operation of liquid rocket engines having higher heat flux densities for long duration. This paper aims to bring all the research carried out in the field of liquid rocket engine film cooling since 1950. The analytical and numerical procedure followed, experimental facilities and measurements made and major inferences drawn are reviewed in detail, and compared where ever possible. Review has been made through a discussion of the analyses methodologies and the factors that influence film cooling performance. An effort has also been made to determine the status of the research, pointing out critical gaps, which are still to be explained and addressed by future generations.

1. Introduction

Liquid rocket engines developed for space missions encompass a wide spectrum of performance and structural requirements. Thrust levels may vary from a few Newtons to many thousands of Newtons, with burning time from fraction of a second to hours. In all these engines, the energy released by the propellants must be contained inside the thrust chamber and accelerated through the nozzle to extract the thrust. Extremely high heat flux levels and temperature gradients are present not only in the immediate vicinity of the injector head, but also in the nozzle throat region. It is seen that the maximum heat flux occurs in the close proximity to nozzle throat, and an effective cooling of the throat area is crucial for enhanced reliability and reusability. Regenerative cooling is the standard cooling system for almost all modern main stage, booster, and upper stage engines [1]. Different cooling techniques such as film cooling, transpiration cooling, ablative cooling, radiation cooling, heat sink cooling and dump cooling have been developed in the past to reduce regenerative cooling load and propellant requirements. Film cooling can be employed either at the combustion chamber or at the nozzle of a

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Peer review under responsibility of National Laboratory for Aeronautics and Astronautics, China.

https://doi.org/10.1016/j.jppr.2018.01.004

Nomenclature

c	specific heat (unit: J/(kg·K))
CM	turbulent mixing coefficient
D	film cooling test section inside diameter (unit: m)
Da	Damkohler number
h	convective heat transfer coefficient (unit: W/(m^2·K))
$H*$	heat release potential, $(T_{ad}-T_{g,t})/T_{g,t}$
I	engine thrust (unit: N)
I_{sp}	specific impulse (unit: 1/s)
L	width of adiabatic wall (unit: m)
M	blowing ratio, $\rho_c v_c/\rho_g v_g$
$\dot{m}$	mass flow rate (unit: kg/s)
P	pressure (unit: bar)
Pr	Prandtl number
q	heat flux (unit: W/m^2)
Q_s	scaled heat flux ratio, $(q_{hot}-q_{cold})/(q_{max}-q_{cold})$
R	chamber radius (unit: m)
S	coolant slot height (unit: m)
T	local temperature (unit: K)
Tu	turbulence intensity (unit: %)
N	average velocity (unit: m/s)
v	velocity (unit: m·s^{-1})
V	vaporization rate of the liquid surface (unit: lbm·in^{-1}·s^{-1})
VR	velocity ratio, v_c/v_g
We	Weber number
x	axial distance measured from the coolant injection point (unit: m)
X	non dimensional distance from the coolant injection point, x/D
α	thermal diffusivity (unit: kg/(m·s))
β	tangential angle
δ	factor distinguishing the initial mixing and developed region of the jet
χ	entrainment factor for plane, unaccelerated flow with continuous slot injection
θ	shape factor for the mixing layer profile
γ	azimuthal angle
ρ	density (unit: kg/m^3)
ε	adiabatic film cooling effectiveness, $(T_{ad}-T_g)/(T_c-T_g)$
η	film cooling effectiveness, $(T-T_g)/(T_c-T_g)$
ω	injection parameter

Subscripts

ad	adiabatic
c	coolant
cc	combustion chamber
$cold$	non reactive heat flux
g	main stream flow
hot	reactive heat flux
max	maximum heat flux
e	entrainment
t	total

rocket engine. Liquid film cooling with fuel or oxidizer as the coolant can be employed in the combustion chambers of gas generator/expander/staged combustion cycle engines. In case of gas generator cycle, the turbine exhaust gas can be used as a gaseous film coolant in the combustion chamber or nozzle sections. It is found that all these methods lead to reduced wall temperatures.

The mechanism by which film cooling maintains a lower combustor wall temperature is considerably different from that of convective cooling. Film cooling is accomplished by interposing a layer of coolant fluid between the surface to be protected and the hot gas stream. The fluid is introduced directly into the combustion chamber through slots or holes and is directed along the walls (Figure 1). A typical temperature distribution from the hot combustion gases to the exterior of the chamber wall in a film cooled combustion chamber is shown in Figure 2. It can be observed that the coolant film produces a thermal insulation effect and reduces the chamber wall temperature. Coolant film may be generated by injecting liquid fuel or oxidizer through wall slots or holes in the combustion chamber, or through the propellant injector. The cooling effect will persist up to the throat region in the case of a shorter combustion chamber. In a fully film-cooled design, injection points are located at incremental distances along the wall length. In liquid film cooling, the vaporized film coolant does not diffuse rapidly into the main gas stream but persists as a protective layer of vapor adjacent to the wall for an appreciable distance downstream from the terminus of the liquid film. The film coolant also forms a protective film which restricts the transport of the combustion products to the wall, thus reducing the rate of oxidation of the walls.

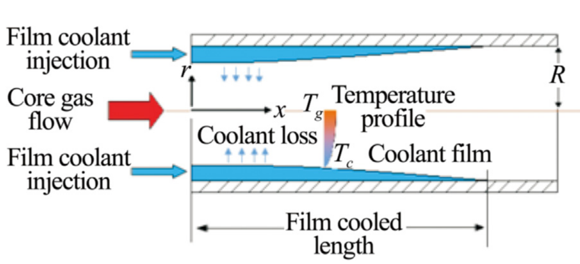

Figure 1 Schematic of the physical system.

Engines like SSME, F-1, J-2, RS-27, Vulcain 2, RD-171 and RD-180 use film cooling technique for combustion chamber cooling. Several open-cycle rocket engines have turbine exhaust gas (TEG) delivered to the nozzle for film cooling, including the F1 engine [2] and J2 engine [3] of the United States, the upgraded LE5 engine [4] of Japan and Vulcain 2 of the EADS Astrium. In Vulcain-2 engine, the combustion chamber wall film cooling has been employed from the injector face plate down to the nozzle throat Section [5]. Engines like Vulcain 2 uses this cooling

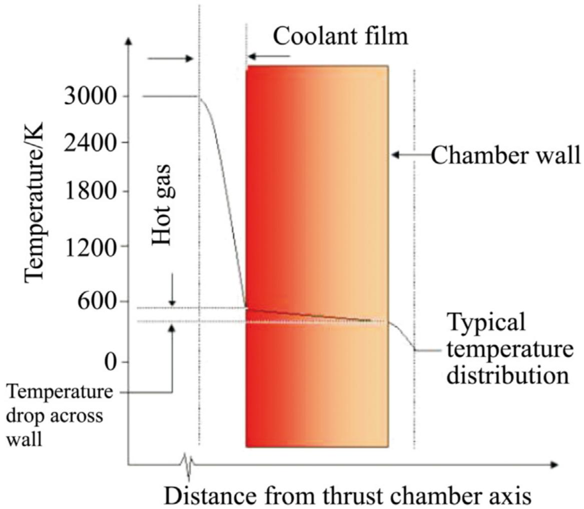

Figure 2 Typical temperature distribution of combustion chamber across wall.

method at both locations. The overall dump cooled nozzle extension used on Vulcain-1 has been replaced by a completely new nozzle design that provides turbine gas film cooling injection at the nozzle extension [5]. This has produced a cooling effect on the nozzle structure. Similarly turbine exhaust gas (TEG) is injected in J-2X engine as a coolant and pure hydrogen gas is used to cool LE-7A [6].

Film cooling of rocket engines will play a major role in the development of reusable and booster rocket engines as this technique can increase the chamber life by reducing the thermal stresses. Efficient use of the coolant films is critical because of the associated specific impulse reduction [7]. As film cooling is such an important cooling technology for the durability of liquid rocket engines, this paper presents a summary of past research in the area of film cooling. This review looks at the features of gaseous and liquid film cooling processes, the methodologies and results in theoretical, computational and experimental approaches and the major factors that influence film cooling performance. Every effort has been made to cover all the advancements published in this field.

2. Studies on film cooling

Origin of film cooling studies extend way back into history of fluid mechanics starting from the days of Reynolds [8] and Lamb [9] of the late 19th century. Reynolds [8] studied the behavior of vortex rings, a topic closely related to the modeling of film cooling jets. The application of a fluid film for the protection of surfaces in aerospace related field is often attributed to Wieghardt [10]. He applied this technique to de-icing of aircraft wings by blowing warm air over them. Research on film cooling for turbine components and rocket combustion chambers started in 1950s. Since then, there have been numerous experimental studies and several models have been developed for the prediction of film cooling effectiveness. Film cooling investigations over the years have concentrated mainly on gaseous film cooling applied to gas turbine airfoils followed by work on film cooling of rocket combustion chambers. Survey of current literature on airfoil film cooling shows extensive studies for coolant holes with various geometries and at various stream-wise injections. Film cooling is applied to nearly all of the external surfaces associated with the airfoils that are exposed to the hot combustion gases such as the leading edges, main bodies, blade tips, and end walls. In gas turbines, high pressure bleed air from the compressor is exhausted from the internal convective passages of the turbine blades, through holes drilled into the blade surfaces. Though the processes are similar, the key challenges involved in rocket engine film cooling are different from that of airfoil film cooling. The main differences between the two processes includes: (i) extreme heat flux, temperature and pressure conditions present in modern rocket engines, (ii) the presence of highly accelerated flows, (iii) the initial turbulent state of the coolant and core streams, (iv) the use of gaseous and liquid coolants, (v) two phase flow conditions present with liquid film cooling, (vi) surface curvature effects, (vii) the requirement of flow uniformity and wall adherence, (viii) density gradients and compressibility effects (Coolant to core stream density ratios are in the order of two or higher are typical in many engines), (ix) the effect of reactive coolant, (x) radiation effects, and (xi) unsteady flows (Unsteadiness in the core flow of a rocket engine can arise from various sources including turbulent flow in the feed lines, fluttering of pump wheel blades, vibrations of control valves, and unsteady motions in the combustion chamber and gas generator). Moreover, the film cooled length is an important parameter in the design of rocket combustion chambers. It is also to be noted that in rocket engines, film cooling is always applied in combination with other cooling methods (usually regenerative cooling).

Proprietary restrictions in space fairing countries and companies have resulted in very limited availability of detailed data about film cooling of liquid rocket engines in the open literature. Most of the studies available in the open literature are at least 30 years old [11,12]. However, there has been renewed interest in the subject due to the development of new engines. The following discussions are organized to provide an outline of two major divisions associated with rocket thrust chamber film cooling: gaseous and liquid film cooling. Research conducted in these areas highlighting the main contributions is provided.

3. Gaseous film cooling

Gaseous film cooling is considered to have potential use in nuclear rockets and high-energy liquid chemical rockets. In general, there are limited number of film cooling investigations dealing with gaseous-film cooling, applicable directly to a rocket engine. Among the earliest studies, Lucas and Golladay [13] investigated the applicability of gaseous-film cooling to a rocket combustion chamber. The effects of cooling a cylindrical portion of the combustion chamber with both tangential and inward-angled slot coolant injection and cooling the nozzle with tangential injection were studied. In each configuration, nitrogen was used as the coolant and in addition, limited data was also obtained with the cooled nozzle using propane as the coolant. It was concluded that correlations for adiabatic wall film cooling effectiveness can be employed to predict the performance associated with non-reactive, non-condensable gases. The requirement of the reactive coolant was higher than the non-reactive coolant with equivalent transport properties. Multi-slot gaseous film cooling experiments for standard nozzle using hydrogen, methane and carbon monoxide as coolants were carried out by Totten [14]. It was concluded that multi-slot cooling is a most promising means of cooling with high energy propellants. No throat erosion was noticed with methane and hydrogen whereas carbon monoxide induced a slight amount.

Back and Cuffel [15] had conducted an experimental investigation to delineate the structure of the flow field and temperature distributions in a shock wave/turbulent boundary-layer interaction with film cooling. The study revealed the influence of wall cooling on the flow field, wave structure, and size of the flow separation region. The size of the separation region became smaller compared to the case without cooling, and the separation and reflected shock waves merged together near the edge of the velocity boundary layer, extending in to the free stream as one wave. The ratio of heat transfer coefficient and wall shear stress with and without cooling was found to be 2.3 and 1.6–1.7. Stoll and Straub [16] used a parabolic finite difference boundary layer code with the k-ε model to investigate film cooling and heat transfer in nozzles with rounded, smooth throats. No influence of the throat on the cooling film stability was noticed. It was also pointed out that a coolant to mainstream mass flow ratio of 2.1% yielded a significant decrease in heat transfer, both in subsonic and supersonic regions. Arrington et al. [17] compared film cooling of a standard conical nozzle with a bell nozzle. Gaseous hydrogen and gaseous oxygen were used as propellants. Film cooling was successfully employed in both nozzle configurations. The effect of gaseous film injection in dual bell nozzle was numerically studied by Martelli et al. [18]. The injection was made in the first bell through an axis-symmetric slot located in the divergent section, and it was found that the expansion fan originating from inflection helps the film to protect the wall better. It was also observed that expansion generated by the inflection point lowered the wall recovery temperature and reduced the mixing. This allowed the film to protect the wall for a longer distance. Matesanz et al. [19] have carried out numerical simulation of slot film cooling in convergent-divergent nozzles using direct simulation of turbulence and large eddy simulation (LES) algorithms. Predicted adiabatic wall temperatures and film cooling effectiveness were within 20% of the corresponding experimental data.

The above results established the feasibility of gaseous film cooling for thermal load reduction in conventional and advanced rocket nozzles. However, the effect of film cooling on nozzle behavior in terms of flow separation and associated side loads can create limitations for this technique as evidenced by the reported breakdown of the nozzle wall during the early sea-level testing of the LE-7A engine [20]. Influence of film cooling on separation characteristics during start up and shut down transients can be more substantial depending upon the phasing between main flow and the secondary coolant flow dynamics. Transient flow in a subscale LE-7A nozzle with film cooling was numerically investigated by Takahashi et al. [21]. The nozzle had a joint section with backward facing step for film cooling between its upper section and extension sections. Rapid movement of the separation point was noticed when it passed through the joint section. Occurrence of large scale asymmetry of the flow and large side road was observed when the separation point passes through the joint section. The influence of film cooling on nozzle flow separation has been experimentally and numerically studied by Reijasse and Boccaletto [22]. The study revealed a dynamic phenomenon of the separation shock front region during rapid film injection. Wang and Guidos [6] have conducted transient numerical side load analysis of a film cooled nozzle extension. Simulations were carried out for engine start up and shut down transients. Simulations showed generation of peak side load due to Mach disk flow and the subsequent jump of the separation line.

Gaseous film cooling is found to be one of the most promising candidates to achieve lower heat load to the nozzle wall. For gas-generator cycle engines, the turbine exhaust gas (TEG) is readily available as film coolant. The characteristics during a steady operation are predictable, but film cooling at transient conditions is not well understood. The side loads generated during the transient operation is to be studied in detail, as side loads are one of the decisive factors for the use of film cooling during the transient operation of rocket nozzles.

3.1. *Effect of various parameters*

It is seen that many factors affect film cooling performance. The various parameters studied so far are listed in Table 1. Each of these factors is not necessarily independent of the others, and so a combination of any of these can potentially change the film cooling performance. Therefore, a large number of operating conditions may be possible and hence there is inherent difficulty in predicting film cooling performance.

Marek and Tacina [23] employed tangential slots to inject coolant air inside a rectangular test section to study the effect of free-stream turbulence levels on film cooling effectiveness. The film cooling effectiveness decreased as much as 50% when the free-stream turbulence intensity was increased from 7% to 35%. Gau et al. [24] conducted

Table 1 Factors affecting film-cooling performance.

Coolant/mainstream flow conditions	Geometric parameters	Other factors
Turbulence in main stream, *Tu* [23]	Injector geometry [33,39,41]	External shock wave [34,35]
Coolant Mach No [31]	Injector orientation [43]	Swirl in the mainstream [24]
Mainstream Mach No [29,30,44]		Heat capacity, c_c [35]
Blowing ratio, *M* [37,42]		

experiments in a film cooled circular pipe with an abrupt expansion of 2.4:1. These experiments demonstrated that the swirl in the mainstream had a significant effect on the film-cooling performance.

Many studies have discussed film cooling under compressible conditions. Volchkov et al. [25] theoretically analysed the effect of velocity and temperature compressibility on film cooling effectiveness. He concluded that compressibility effects are of the first order and can be neglected in practical combustion chambers. Repukhov [26] accounted the effects of compressibility on the turbulent boundary layer equations and obtained similar results. However, Hansmann et al. [27] and Pedersen et al. [28] suggested that compressibility can significantly influence film cooling effectiveness. Their results showed higher effectiveness at higher density and velocity ratios (coolant to mainstream). Pedersen et al. [28] further showed that this effect is more predominant at higher blowing ratios. The role of mainstream flow velocity in a film cooling duct was studied by O'Connor and Haji Sheikh [29] and Kuo et al. [30]. Kuo et al.'s experimental observations reported film cooling to be effective in subsonic flow, but not in supersonic flow. They observed bending of the injected flow stream towards the upstream direction against the incoming mainstream in the case of supersonic flow. This created a stagnation region and temperature rise in the flow field leading to negligible film cooling effect on downstream wall regions where the supersonic flow was dominant. However, there exists a vast amount of experimental data on supersonic film cooling which cover a wide range of Mach numbers and show that film cooling is more efficient in supersonic flows than subsonic flow. Experimental investigation of Juhany et al. [31] showed improvement in film cooling effectiveness with the increase of coolant Mach number. Comparison between Helium and air experiments indicated that the effectiveness increases with heat capacity of the gas. Keener et al. [32] investigations in a two-dimensional (2-D) Mach 2 nozzle showed that the boundary-layer thickness has a major effect on the wall heat fluxes. High heat fluxes were also detected during the flow establishment, starting with the nozzle shock from the shock tunnel, due to the absence of the boundary layer. Supersonic film injection experiments for two types of injection geometries and pressure ratios were conducted by Aupoix et al. [33]. The coolant film was generated by vaporizing liquid nitrogen into air in a mixing chamber. The experiments showed that film cooling is much more efficient in supersonic flows than subsonic flows. This is mainly attributed to the following: supersonic flow in cooling channels will provide thin boundary layers, smaller turbulent length scales and reduced mixing. The spreading rate of the mixing layer can be controlled via velocity, density ratios and convective Mach number. This may further reduce mixing. Dellimore discussed the disagreements in the literature over the compressibility effects in film cooling problems. He showed that compressibility influences film cooling performance when the flow Mach numbers are high and the total temperature ratio is low. As the flow Mach number increases, the growth rate of the shear layer decreases and moves shear layer impingement point farther downstream. This leads to improvement in film cooling performance. He also noticed that Volchkov et al. [25] and Repukhov [26] used data corresponding to low coolant Mach numbers and hence predicted negligible effect of compressibility on film cooling performance.

The effect of the external shock wave on the film cooling was investigated in the Mach 2.35 wind tunnel by Kanda et al. [34]. At higher pressure ratios, decrease in film cooling effectiveness was noted which was attributed to the decrease in the local Mach number. An increase in heat transfer coefficients and adiabatic wall temperatures were noted in the interacting region. Numerical investigations by Peng and Jiang [35] confirmed the above findings. Results by Peng and Jiang [35] showed reduction in adiabatic effectiveness resulting from impinging oblique shock wave. In addition, the oblique shock wave enhanced the mixing of the cooling stream with the free stream, especially for lighter-gas coolants. Helium was more easily affected by the shock wave as it diffused faster than methane or nitrogen.

An investigation of film cooling in the laminar supersonic regime was conducted by Yang et al. [36] and reported increased cooling with an increasing mass flow of the coolant, with the slot height being unimportant. Investigations on film cooling on a flat plate with a slot injection in laminar supersonic flows were done by Heufer and Olivier [37]. Experimental and numerical results showed no influence of the blowing parameters on the cooling effects. The cooling effect was found to be influenced by the core gas flow conditions and was reported that the film cooling technique is highly effective under laminar flow conditions. Experimental data by Hombsch and Olivier. [38] also confirmed that film cooling is much more efficient in laminar flow than in turbulent flow. It was also shown that tangential injection supersedes angled slot injection. Experiments by Arnold et al. [39] showed significant variations of wall temperatures due to injector design and more pronounced circumferential variations in wall temperature at higher combustion chamber pressures. Experiments were conducted in a subscale rocket combustion chamber with tangentially injected film of hydrogen. Tangential slot injection was investigated for various film-cooling parameters in the same experimental set up [40]. Experimental investigations by Shine et al. [41] showed that injector configuration had a major role in film cooling performance especially at higher blowing ratio. Shine et al. [42] had presented gaseous film cooling performance of straight cylindrical cooling holes inside a cylindrical test section. Experimental and computational results showed that film cooling performance change with blowing ratio due to differences in the magnitude of vortices created downstream of the coolant injection and the jet exit momentum. Study showed that an optimum blowing ratio exists for a given geometric configuration. The flow field associated with cylindrical coolant holes inclined in tangential and azimuthal direction, employed inside a circular pipe had been examined by Shine et al. [43] The results indicated that through the use of optimal coolant injector configurations, maximum cooling performance could be achieved. Significant increase in the heat transfer coefficients were noticed at the downstream of injection for all the injector cases.

Review shows that there are strong effects of free stream turbulence, injector orientation and shock waves on the performance of film cooling. Effect of parameters such as unsteady mainstream flow, shape of the coolant holes/slots, and length of the coolant holes/slots etc. needs further investigation. A slot introduces coolant uniformly around the circumference with less mixing with the overflowing mainstream than occurs with discrete coolant jets originating from a row of holes. Consequently, the slot provides an ideal performance and may be used as a basis of comparison. It can be noticed that some studies are carried out with slot film cooling while others are performed with discrete coolant holes.

3.2. Recent numerical efforts

Researchers have used Direct Numerical Simulation (DNS), Large Eddy Simulation (LES) and Reynolds Averaged Navier-Stokes (RANS) to explore film cooling flows. The selection of method usually depends on the complexity of the problem and the fidelity of the results. Initial studies featured RANS based computations along with k-ε turbulence model. Stoll and Staub [16] used a finite difference boundary layer code to predict wall heat flux in a converging-diverging nozzle. Jansson [45] used both k-ε model and an algebraic stress model to simulate steady and unsteady simulations. Mean velocity and temperature profiles were predicted fairly well for the steady state, while the temperature predictions deviated significantly for the unsteady case. Cruz and Marshall, Dellimore, Betti and Martelli [44,46,47,51] etc. have used Spalart-Allmaras (SA) turbulence model along with RANS based solver. It is seen that SA model with a constant turbulent Prandtl number predicted mean temperature and velocity profiles very close to experimental values. Dellimore [44] later used a blended k-ϱ, k-ω model and captured effectiveness curve moderately well. Voegele et al. [48] used LOCI-CHEM RANS solver to model a film cooled wall along with turbulence models such as two equation shear stress transport (SST) model and one equation SA model. The SA model predicted the initial decay region fairly well, SST model approached experimental values for the wall wake, but failed to predict the film decay rate in the far field. Na et al. [49] used finite volume code with second- order upwind differencing and eddy-diffusivity turbulence model to study film cooling. It was observed that for a flat plate configuration, SST turbulence model provided better predictions, whereas the realizable k-ε model was suited for a semi-cylindrical leading edge with a flat after body configuration. The importance of radiative heat transfer in film cooled nozzle was computationally analyzed by Badinand and Fransson [50]. Finite volume method with the spectral line-based weighted sum of gray gases had been implemented along with Navier-Stokes flow solver for calculating radiative heat transfer. It was observed that for the shocked nozzle, Mach disk emission is as intensive as the combustion chamber emission and the radiative heat flux is also intensive.

Raijasse and Boccaletto [22] experimentally and numerically studied film cooling influence on the nozzle flow separation phenomenon. Numerical study was carried out with the RANS code named CPS developed by ONERA, Bertin Technologies and CNES. Numerical results predicted the experimental features and revealed separation shock foot motion during film coolant injection. A 3DRANS solver, modified to simulate multi-component mixtures of thermally perfect gases was developed by Betti et al. [52] for the analysis of liquid rocket engines. Spalart-Allmaras one equation model was modified to take into account compressibility effects had been used in the model. Betti et al. [52] further analyzed thrust chamber film cooling efficiency using a multi-component RANS solver and reproduced the test case results pertaining to oxygen/methane film-cooled thrust chamber. Matteo et al. [53] used EcosimPro, an object oriented tool and developed a quasi 2-D integral formulation to study a film wall jet in combustion chambers. A DLR H_2/O_2 combustion chamber test results have been used to validate this model.

Maqbool et al. [54] conducted experiments with the objective of validating the film cooling performance prediction of LOCI-CHEM, NASA's hot flow design tool. Schlieren images and heat flux measurements showed qualitative consistency with the numerical results on most part of the film cooled wall with the exception on the upper non film cooled side. These models are capable of supporting the preliminary design of liquid rocket engine systems, analysing the performance of film cooling and predicting the loss in performance.

The current computational efforts are mainly based on RANS with eddy-diffusivity or Reynolds stress models. It is noted that LES and DNS approaches uses turbulent scales of the flow directly and may be suited to film cooling flows in contrast to RANS approach where turbulence is fully modeled. Matesanz et al. [19] conducted LES and DNS studies using a finite element Navier Stokes CFD code to study slot film cooling in converging-diverging nozzle. The predictions were very good, but limited to fewer comparisons. Tyagi and Acharya, and Muldoon and Acharya [55,56] also used LES and DNS approaches and were successful in predicting important kinematic and thermal properties. However, these studies were applicable to turbine film cooling cases.

3.3. Side load analysis

Transient nozzle side loads have been noticed for many rocket engines during their initial development and has the potential to cause structural damages [57–59]. When TEG is used to cool nozzle extension part, this cause additional side load physics due to interaction of the film coolant flow with the Mach disk flow. Extensive experimentation has been carried out by Boccaletto et al. [60] to characterize the influence of coolant jet on flow separation. It was observed that positioning of the main jet separation line is clearly influenced by the film cooling flow. In both steady and transient conditions, forward -backward movement of the shock system was noticed under various coolant flow rates and the amplitude of the shock excursion was smaller in transient conditions [22]. Wang and Guidos [6] has conducted transient 3-D numerical simulations to study the side-load physics of a rocket engine consisting of a film cooled nozzle extension. During start up transients, a slight imbalance in the circumferential wall pressure between the film coolant exit and the Mach disk flow separation line caused the Mach disk flow to become asymmetric and resulted in side loads. Wang solved transport equations using a time-marching sub-iteration scheme to study the side load physics of a film cooled nozzle. Turbulence was modeled based on an extended k-ε model [61] with wall function approach. The characteristics and magnitude of the side load calculations compared reasonably well with those of a LE-7A test [59]. It can be concluded that the side load observed during the transient operation of film cooled rocket engines is an important issue and needs to be investigated further.

3.4. Correlations for adiabatic effectiveness

Earlier theories on the prediction of film cooling effectiveness are based on the assumptions proposed by Librizzi and Cresci [62]. They assume that the coolant fluid is fully mixed with the mainstream in the boundary layer and therefore exhibits a uniform temperature. The discontinuity of temperature appears only at the outer edge of the boundary layer. The growth of the boundary layer is similar to usual laws and do not have any influence of injection. The equation for film cooling effectiveness is proposed in terms of coolant Reynolds number (Re_c), ratio of coolant velocity to mainstream velocity (VR) and the non dimensional distance from the coolant injection point (X). The following equations are proposed by various researchers.

Stollery and Ehwany [63] proposed

$$\varepsilon = 3.09\left(X/VR\right)^{-0.8} Re_c^{0.2} \tag{1}$$

Kutateladze and Leont'ev [64]

$$\varepsilon = 3.1\left\{4.16 + \left(X/VR\right)^{-0.8} Re_{c-}^{0.25}\right\}^{-0.8} \tag{2}$$

Librizzi and Cresci [62]

$$\varepsilon = 3.0\left\{3 + \left(X/VR\right)^{-0.8} Re_{c-}^{0.2}\right\}^{-1} \tag{3}$$

Hartnett et al. [65]

$$\varepsilon = 3.39\left(X/VR\right)^{-0.8} Re_c^{0.2} \tag{4}$$

Tribus and Klein [66]

$$\varepsilon = 4.62\left(X/VR\right)^{-0.8} Re_c^{0.2} \tag{5}$$

It can be noted that the major differences in these equations are mainly in the coefficients. This is mainly due to the differences in the assumptions about temperature profile, i.e., the amount of mixing of the coolant with the mainstream. The more the mixing, the lower the coefficient. Spalding [67] proposed a correlation based on the assumption that the flow near the wall behaved more like a jet than a boundary layer. It was noticed that this assumption is valid when the coolant injection velocity is appreciably larger than that of the mainstream. The proposed correlation is

$$\varepsilon = 3.4\left\{X\left|1 - \frac{1}{VR}\right|\right\}^{0.5} \tag{6}$$

Spalding [67] compared the theories and experimental data from various sources and proposed a correlation for effectiveness that would fit all experimental data. Spalding had shown that it could be used over a broad range of geometrical slot arrangements and velocity ratio (v_c/v_g). Their result for uniform properties is as follows.

For $X < 7 : \varepsilon = 1$
For $X \geq 7 : \varepsilon = 7/X$
Where,

$$X = 0.91\left(X/VR\right)^{0.8} Re_c^{-0.2} + 1.41\left(X\left|1 - \frac{1}{VR}\right|\right)^{0.5}$$

However, it was limited to tangential injection of the coolant parallel to an adiabatic wall. Subsequently, Papell [68] presented further semi-empirical corrections to the equation to correct data obtained with coolant injection through angled slots and normal holes. Hatch and Papell

[69] developed film cooling correlations for relatively high gas stream temperatures. In contrast to above models, Hatch and Papell assumed that the coolant film existed as a discrete layer and no mixing. Empirical modifications are then made to take care of this mixing phenomenon. Hatch and Papell's equation for gaseous film cooling with angled slot is

$$\begin{aligned} \ln \varepsilon &= \left(\frac{h_g L_x}{(mc)_c} - 0.04\right)\left(\frac{Sv_g}{a_c}\right)^{1/8} f(VR) + \ln\cos(0.8\beta_{eff}) \\ f(VR) &= 1 + 0.4\tan^{-1}\left(\frac{1}{VR} - 1\right) \quad \text{for } \frac{1}{VR} \geq 1 \\ &= 1.5 \times VR \times (VR - 1) \quad \text{for } \frac{1}{VR} < 1 \\ \beta_{eff} &= \tan^{-1}\frac{\sin\beta}{\cos\beta + \frac{1}{M}} \end{aligned} \tag{7}$$

Sellers [70] had shown that this equation could be used for multiple-slot correlation as well as single-slot. Sellers used $K=0$ and indicated that for rocket engine film cooling experiments, a value of $K=0.04$ might be too large. He argued that multiple-slot case could be analyzed by considering the concept of progressively decreasing adiabatic wall temperature. NASA SP 8124 [71] is based on the fact that the effects of turbulence between cooling film and mainstream cause an entrainment and is applicable to gaseous film cooling.

$$\varepsilon' = \varepsilon \times \frac{\dot{m}_e}{\dot{m}_c} \tag{8}$$

$$\frac{\dot{m}_e}{\dot{m}_c} = \frac{\dot{m}_t - \dot{m}_e}{\dot{m}_c} f(\Psi_r, \varsigma, x, r) \tag{9}$$

Where Ψ_r and ς are the reference entrainment fraction and mixing length height respectively. A correlation to predict slot film-cooling efficiency of a wall jet, under conditions of variable turbulence intensity, flow and temperature was developed by Simon [72]. Simon proposed the following expressions as a result of analytical and experimental considerations:

$$\varepsilon = \frac{1}{1 + CM'\left(\frac{x}{MS}\right)} \tag{10}$$

Where $CM' = f(CM, \delta, x, M, S)$.

Simon [72] considered that at the coolant exit, a semi-contained turbulent jet is formed near the wall. This zone is present till the inner edge of the growing main jet reaches the wall. This is followed by the fully developed region where complete mixing between the coolant and the mainstream happens. Dellimore et al. [73] extended Simon's model to include the effect of variation in the mainstream pressure gradient. It was found that the response of the coolant film to the pressure gradient depends on the velocity ratio VR. A modified film-cooling model for application in a combined convective and film-cooled combustion chamber with an accelerated hot gas was developed by Arnold et al. [74]. The model was used to predict film cooling effectiveness at different combustion-chamber pressures and film blowing rates at sub-, trans-, and supersonic conditions. He used tangential injection with an angle of 15° between the hot gas and coolant. The prediction of film cooling effectiveness is given as

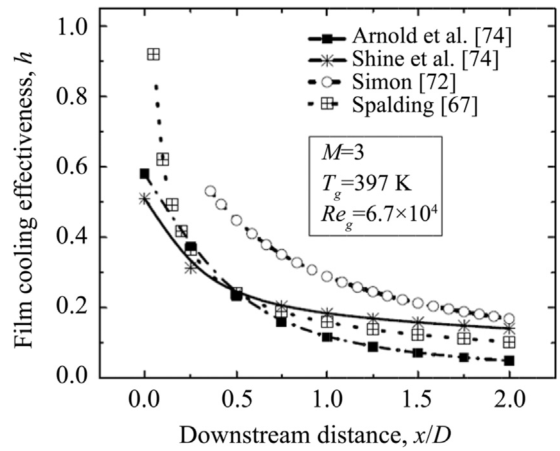

Figure 3 Comparison of adiabatic effectiveness prediction between different studies.

$$\varepsilon = 1.9Pr^{2/3}/1 + 0.329\left(\frac{c_g}{c_c}\right)\left[\left(\frac{x}{s}\right)\left(\frac{1}{M}\right)\right]^{0.8} {Re_g}^{-0.2}\omega \tag{11}$$

The non-dimensional parameter ω depends on the angle of injection between the hot-gas fluid and the film coolant. A summary of different analytical and semi-empirical correlations for determining film cooling efficiency available in open literature and their application in a GOX/kerosene combustion chamber is also described by Kirchberger et al. [75].

The film cooling effectiveness values predicted by Arnold et al. [74], Simon [72] and Spalding [67] are compared with the experimental values of Shine et al. [42] in Figure 3. Simon's model [72] predicts considerably higher values compared to other models. This is due to the increased dispersion of the coolant when injected through discrete holes. Predictions from other models are very close in the near injection regimes.

The gaseous film cooling studies mentioned earlier and their main focus of study are summarized in Table 2. It shows that many aspects of gaseous film cooling process are now getting revealed. The feasibility of gaseous film cooling has been established and is found to be an effective cooling method both experimentally and numerically especially for the nozzle cooling. It is also found that there are strong influences of blowing ratios, coolant Mach number, external shock wave, free stream turbulence, and injector orientations on film cooling performance. Correlations are available to predict film cooling performance which helps in the design of cooling system of rocket nozzle. It is observed that introduction of a coolant film has resulted in an increase in heat transfer coefficients downstream of injection. This leads to poor net heat flux reduction. There are no apparent limitations on cooling capability, time, or chamber pressure with film cooling. However, if one of the propellants (usually fuel) or an inert fluid is used as a coolant at the nozzle throat, there is a performance penalty

Table 2 Gaseous film cooling studies applicable to rocket thrust chambers.

	Main focus of study
Lucas and Golladay [13]	Applicability of gaseous-film cooling
Back and Cuffel [15]	Influence of wall cooling on the mainstream flow field
Stoll and Straub [16]	Assessment of heat transfer to nozzle wall with film cooling
Arrington et al. [17]	Performance in conical and bell nozzles
Martelli et al. [18]	Applicability in duel bell nozzles
Matesanz et al. [19]	Applicability of LES algorithms
Marek and Tacina [23]	Effect of free-stream turbulence
Gau et al. [24]	Effect of swirl in the mainstream
O Connor and Sheikh [29]	Effect of mainstream flow velocity
Kuo et al. [30]	Effect of mainstream flow velocity
Juhany et al. [31]	Effect of coolant Mach number
Aupoix et al. [33]	Supersonic film injection
Kanda and Jiang [34]	Effect of the external shock wave
Peng et al. [34]	Effect of the oblique shock wave
Heufer and Olivier [37]	Film cooling in laminar supersonic flows
Arnold et al. [39]	Effect of injector design
Shine et al. [41–43]	Effect of injector configuration
Spalding [67]	Correlation for effectiveness
Hatch and Papell [69]	Correlation for high gas stream temperatures
Papell [68]	Correlation for angled slots and normal holes
Sellers [70]	Correlation for multiple-slot
Simon [72]	Correlation for variable turbulence intensity, flow and temperature
Dellimore et al. [73]	Correlation for an accelerated hot gas flow
Arnold et al. [74]	Effect of Tangential slot injection
Hombsch and Olivier [38]	Different injection angles, coolant mass flow rates and free stream conditions

(specific impulse loss). Film cooling with low molecular weight gases will produce less specific impulse loss and may be more attractive. Further improvement into these kinds of investigations will have to focus on measurement accuracy and numerical effort through LES/DNS modeling.

4. Liquid film cooling

Liquid film cooling process is different from that of gaseous film cooling because of the presence of phase change during the cooling process which vastly increases the cooling capacity. However, literature on liquid film cooling is quite limited compared to that of gaseous film cooling. Table 3 summarizes the major experimental, analytical and numerical studies conducted for liquid film cooling applicable to rocket combustion chambers.

4.1. Feasibility studies

One of the first experimental studies in film cooling is done at JPL by Boden [76]. The experimental engine was operated at 4448 N thrust with aniline alcohol fuel and nitric acid oxidizer at 21.8 bar chamber pressure. Film coolants such as water, aniline-alcohol fuel, 60-octane gasoline, methyl alcohol, anhydrous ammonia, and JP-3 jet fuel with a maximum flow rate of 7% of the engine propellant flow was used in the experiments. A reduction of heat flux to the engine walls from 10% to 97% was noted. Film cooling investigations in a liquid ammonia-liquid oxygen rocket engine was conducted by Morrell [77]. Various coolants studied were water, ethyl alcohol and liquid ammonia. These experiments helped in estimating the film coolant requirements of cylindrical rocket chambers having similar propellant injectors. Knuth [80] determined sufficient conditions for the stability of thin liquid film flowing under the influence of high velocity turbulent gas streams. The applicability of film cooling to rocket engines using earth-storable, space-storable and cryogenic propellant combinations were investigated by Stechman et al. [82]. In all experiments, fuel was used as film coolant. The studies showed that N_2O_4/MMH, ClF_5/MMH and other similar propellant combinations are readily adaptable to small film-cooled spacecraft engines. Totten [14] made a study on the behavior of a liquid film in a typical nozzle throat. It was observed that injecting the liquid in a straight converging section and at the throat provided the best liquid film cooling performance. Cook et al. [84] had noticed that carbon deposition was the limiting factor of hydrocarbon fuels. Volkmann et al. [85] studied the effects of film cooling on reducing the heat flux experienced at the throat of a rocket. Tests were conducted in a subscale LOX/RP-1 high pressure (138 bar) combustor. Peak heat flux reduction of 70% was observed with film cooling in his studies. Kirchberger et al. [86] conducted film cooling experiments on a sub-scale heat-sink test article running on GOX and kerosene. His results showed that kerosene was much more effective film coolant than nitrogen.

The above studies established the feasibility of using liquid film cooling technique in rocket engines operating with different propellant-oxidizer combinations. It may be noted that a wide variety of propellant combinations are readily adaptable to film cooling. Engine parameters and the coolants used in these studies are described in Table 4.

4.2. Injector types

Film cooling performance is affected by a number of geometric parameters. There are a few studies which deal with the effect of angle of injection on film cooling performance. In most studies, coolant holes are tangentially angled to the hot gas flow. Coolant holes that are inclined in tangential and azimuthal direction toward the chamber wall are generally referred as compound angle holes. However other factors such as shape of the hole, spacing between the holes, and length of the hole etc. have not been discussed extensively as in the case of gas turbine film cooling [93]. Figure 4 shows straight (holes parallel to the axial direction of the core gas flow), tangential and compound angle arrangements.

Table 3 Major liquid film cooling studies applicable to rocket thrust chambers.

	Nature of study	Focus of study, Injector types
Boden [76]	Experimental	Feasibility of films, Radial injectors
Morrell [77]	Experimental	Feasibility of different coolants, Vertical and Tangential slots
Kinney et al. [78]	Experimental	Performance study, Visualization of films, Porous and jet type
Abramson [79]	Experimental	Cooling of nozzles, Tangential slots
Knuth [80]	Experimental	Stability of liquid films, Radial injector
Warner and Emmons [81]	Experimental	Feasibility of H_2 as a coolant, Dual slot radial injector
Stechman et al. [82]	Experimental	Propellants as coolant
Kesselring et al. [83]	Experimental	Development of analytical model, Tangential injector
Cook and Quentmeyer [84]	Experimental	Hydrocarbons as coolant
Volkman et al. [85]	Experimental	Cooling of nozzle throat
Arrington et al. [17]	Experimental	Cooling of bell nozzle
Kirchberger et al. [86]	Experimental	Kerosene as coolant
Crooco [87]	Analytical	Analysis of evaporation of liquid film
Knuth [80]	Analytical	Method for calculating evaporation rate of liquid film
Emmons [88]	Analytical	Determination of the heat transfer coefficient
Gater et al. [89]	Analytical	Analytical model including film instability and transpiration effects
Stechman et al. [82]	Analytical	Introduced 'flow instability efficiency correction factor'
Grisson [90]	Analytical	Incorporated transpiration, radiative heat transfer and free-stream turbulence
Yu et al. [11]	Analytical	Swirling of the liquid film
Shembharkar and Pai [91]	Numerical	Couette flow model
Wang and Luong [92]	Numerical	Regeneratively cooled engine
Zhang et al. [12]	Numerical	Coolant loss is approximated by diffusion of vapor

Table 4 Comparison of various experiments.

	Engine parameters	Coolants used
Boden [76]	$I = 4448$ N, $P_{cc} = 21.8$ bar	water, aniline-alcohol, gasoline, $P_{cc} = 21.8$ bar ethyl alcohol, anhydrous ammonia, JP-3
Morrell [77]	$I = 4448$ N, $P_{cc} = 15.2$–18.6 bar	water, ethyl alcohol, ammonia
Knuth [80]	Test section with combustion products, $P_{cc} \approx 1$ bar	water
Stechman et al. [82]	$I = 44$-4448 N, $P_{cc} = 0.7$–34.5 bar	MMH, H_2, B_2H_6 CH_4

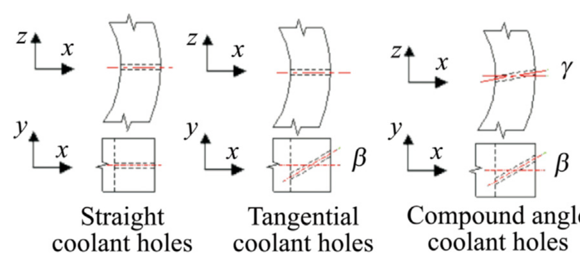

Figure 4 Schematics of different cooling hole shapes is the tangential angle and is the azimuthal angle, (from Ref. [43]).

Boden [76] used nine injector configurations out of which seven were oriented in the radial direction and two were drilled at an angle of 70° off the radial direction. The coolant was injected through multiple drilled holes distributed around the inner wall at one or more axial locations in the engine. An inner ring downstream of coolant injector was used to force the coolant axially before entering the engine. The data obtained was reviewed and correlated by Welsh [94]. It was observed that injection from a single axial location in the combustion chamber was the least efficient method and injection of coolant in a swirling pattern had a negligible effect on cooling performance, although the addition of a deflector plate increased the effectiveness of the coolant. Morrell [77] had successfully conducted liquid film cooling experiments with a vertical rectangular slot injector and tangential type with slots cut at 45° to the axis. Tangential slot injection failed to improve the effectiveness compared to the vertical slot injector. Kinney [78] used porous and jet type injectors with holes cut at an angle of 25° to the axis. No significant difference was noticed in the results with the two different coolant injectors. Abramson [79] used annular slots inclined at 30° to the centre line of the nozzle in his internal film cooling experiments of the exhaust nozzle of a liquid ammonia-liquid oxygen rocket engine. Film cooling of the entire nozzle was achieved with film coolants such as water and anhydrous ammonia. Knuth [80] conducted experiments with radial injector holes and determined sufficient conditions for the stability of thin liquid film flowing under the influence of high velocity turbulent gas streams. Gator et al. [89] conducted experiments with a flat film and measured the amount of liquid that remained attached to a wall with a knife-edge capture slot. Warner and Emmons [81] injected coolant radially through circumferential slots inside the combustion chamber of a gaseous hydrogen-air rocket engine. He found that dual slot injection effectively reduced the quantity of coolant required to film-cool a given length of surface compared to that required when a single slot is employed. Kesselring et al. [83] performed tests with tangential coolant injectors in a nickel calorimetric chamber. The authors assumed that the liquid film immediately evaporated, based upon calculations of the normally expected heat flux without transpiration. Experimental and numerical investigation of tangential and compound angle cylindrical coolant holes were carried out by Shine et al. [41]. It was noted that adding a compound angle produced no improvement in effectiveness compared to tangential injection. Longer liquid film length for tangential injector was seen compared to the compound angle injector.

It can be concluded that tangential coolant injection is a good choice for liquid film cooling process wherein higher effectiveness and film uniformity is of concern. Although several experimental investigations are available in the area of liquid film cooling, they do not specifically deal with liquid film injector orientations. Therefore more investigations need to be initiated to determine the relative performance of different film cooling configurations.

4.3. Analytical and numerical models

The analytical and numerical models available in literature for the liquid film cooling are examined in this section. One of the early works reported is that of Knuth [80] wherein the analysis was based on an extension of the Reynolds analogy to heat, mass and momentum transfer in the turbulent core of a two component fully developed turbulent pipe flow with unidirectional radial diffusion. He had considered only the unidirectional radial diffusion and neglected all other heat transfer mechanisms including the evaporation of liquid film due to convective and radiative heat transfer. The method involves determination of the evaporation rate and the surface temperature of the liquid wall film. Crocco [87] utilized the same interfacial energy balance as Knuth, and hypothesized that liquid film coolant evaporates and diffuses from the boundary towards the hot combustion products. The vapor is then confined in a laminar sub layer that behaves as a thermal barrier. Emmons's [88] analysis considered incompressible flow of the hot gas stream, flowing over a stable liquid film having a uniform temperature equal to the boiling point of the liquid. The velocity profile within the sub layer was based upon a diffusivity variation relationship, originally suggested by Rannie [95] and modified by Turcotte [96]. Turcotte's analysis of the sub layer considered the effect of vapor injection upon turbulence. Using the Reynolds analogy, Emmons obtained an expression relating the heat transfer coefficient between the liquid film surface and the hot gas stream. None of the above analyses take into account the heat transfer by radiation which is significant at high temperatures prevailing in the combustion chamber. It also does not account for the disturbances at the surface of the liquid film and the free-stream turbulence effects.

Kinney [78], Graham [97], Sellers [98] and Emmons [88] equated the convective energy transfer on the surface of the liquid film from the hot gas stream to the energy utilized for the phase change of the liquid coolant. It was assumed that the radiant energy transfer was negligible. Furthermore, in the experimental investigations conducted by Graham, Kinney and Sellers and in a portion of the experimental investigation conducted by Emmons, the wetted surface was essentially adiabatic. Some of the unknown quantities needed for completing the solution were approximated by means of empirical formulae, and the remaining were determined experimentally. Sellers determined two unknowns and Graham and Emmons each determined one unknown from the experimental data. The disadvantage was that the final analytical expression could not be applied with confidence to situations that differ significantly from that for which the unknowns were determined. Subsequently, heat and mass transfer analyses for the wall region wetted by the liquid film was carried out by Gator et al. [89]. A correlation procedure was suggested, which require experimental data points, such as the measurement of the structure of the boundary layer region above the liquid film. Film instability at the liquid-gas interface was considered as important as transpiration effects, but the same was considered only above a critical value of liquid flow rate based on observations by Taylor et al. [99]. No assessment of the validity of the correlation procedures was carried out since accurate experimental data were not available.

Analytical methods suggested by Stechman et al. [82] considered only the convective heat transfer from the main core gas and loss through the chamber walls. He used a modified Bartz [100] equation to calculate turbulent heat transfer coefficient from the combustion gases to the film coolant. The convective heat transfer coefficient from the liquid film coolant to the chamber walls was calculated assuming turbulent liquid flow on a flat plate and using a 'flow instability efficiency correction factor'. He assumed (i) the coolant film exhibits no mixing or chemical reaction with the main core gas, (ii) the coolant film temperature profile does not change rapidly as the coolant moves downstream, (iii) the gradients across the coolant film are small, and (iv) no heat transfer occurs in the chamber walls. The predictions obtained from the model had errors ranging from -20% to 13% depending on the thrust chamber configuration. Kesselring et al. [83] performed tests in a nickel calorimeter chamber using a propellant combination of OF_2/B_2H_6 (oxygen difluoride/diborane). They had developed an integral method to determine the film temperature and film heat transfer coefficient. The model was based on an assumed cubic temperature profile through the wall.

Liquid film cooling model by Huzel and Huang [101] gave semi-empirical correlations for calculating the liquid film cooled length (FCL). This is defined as the distance beyond which the flow ceases to exist. The entrainment of the liquid film was accounted for the first time and is given by

$$FCL = \frac{1}{A}\ln\left(1 + \frac{A\dot{m}_c}{V}\right) \tag{12}$$

where $A = A(Xe)$ and is the liquid entrainment parameter. Xe is the function of ratio between the core flow temperature and the interface temperature. After calculating the film cooled length, NASA model gives an expression for the calculation of the film cooling effectiveness and is

$$\eta = f(\theta, \dot{m}_c, \dot{m}_e, \chi) \tag{13}$$

Trotti [102] compared the predictions from the above model to that of experimental results from a high pressure combustion chamber which used gaseous oxygen as oxidizer and Kerosene JetA-1 as fuel. He observed that this model generally over predicted in most cases. It was also noted that Stechman's [82] film cooling model was not better than NASA model as it was developed for rockets characterized by low combustion chamber pressure.

A numerical model, assuming a turbulent boundary layer flow for the hot gas stream and a Couette flow model for the liquid coolant film was proposed by Shembharkarand and Pai [91]. The model predicted an exponential drop in evaporation rate and did not account coolant transpiration effects. Liquid film length predictions were significantly higher than experimental results. An attempt was made by Grisson [90] to develop a general analysis of liquid film cooling. In Grisson's comprehensive model, transpiration effects, radiative heat transfer and free-stream turbulence were included. Flat plate correlations were used with a modified leading edge distance for convective heat transfer calculations. The radiation calculations were based on Hottel's chart which over-predicted the radiative heat transfer at high temperatures. The entrainment effects were not considered in the analysis and the model was valid only at low coolant flow rates. Wang and Luong [92] developed a computational methodology to predict the hot-gas-side and coolant-side heat transfer in film cooling assisted, regeneratively cooled liquid rocket engine combustors. It was found that film cooling produced the low heat flux near the injector face plate. Yu et al. [11] had performed a literature review on film cooling models along with the assumptions employed in the analyses. The paper described a model experiment and the important processes in film cooling were identified through an order of magnitude analyses. Pertinent benefits of swirling of the liquid film to reduce entrainment were also discussed in this paper. Zhang et al. [12] numerically solved the governing equations for the liquid film and the gas stream coupled through the interfacial matching conditions. The gas-liquid interface was assumed at the state of thermodynamic equilibrium. The radial component of velocity at the interface was calculated based on the diffusion of coolant vapour from the interface to the core gas flow. The method involved the assumption of arbitrary values for pressure gradient and local liquid film thickness. Liquid entrainment and free-stream turbulence effects were not considered. It is unclear whether the effects of transpiring vapour were properly included as a boundary condition in their model. Zhang et al. [103] have conducted numerical study of film and regenerative cooling in a thrust chamber at high pressure. It was observed that the liquid film reaches supercritical regime at the introduction into the thrust chamber. However, it formed a thin layer with low temperature and protected the wall. A one dimensional analytical model of liquid film cooling in rocket combustion chambers operating at subcritical conditions is proposed by Shine et al. [104]. Simplifying assumptions such as steady one-dimensional flow, adiabatic film-wall interface, non-reactive coolant and constant core gas temperature were made to develop this model. The model incorporated mass transfer via entrainment by adapting suitable correlations from literature pertaining to annular flow conditions. The results showed that the effects of radiation and coolant entrainment are significant. It was confirmed that the liquid film length decreases with increase in gas Reynolds number, coolant inlet temperature and free-stream turbulence. The effect of combustion chamber pressure was also investigated and found to be insignificant at higher pressures.

The models described above are focused on the heat and mass transfer at the interface. Majority of the models do not account for the interfacial instability and the annular entrainment of liquid film. Flat plate correlations were generally used neglecting the effects of cylindrical combustion chamber. Enough attention has not been given to the radiation heat transfer from the high temperature gases. Recent research in annular two phase flow indicates the presence of interfacial instability phenomenon affecting annular entrainment. The transpiration of vapour from the liquid film decreases the normally expected convective heat flux that makes the radiation significant. Consequently the results obtained through existing models differ significantly in the prediction of the liquid film length in practical combustion chambers. The cooling mechanism of liquid film at high pressure would be different from that at low pressure. In the supercritical regime, the flow is similar to single-phase flow and all the thermal energy transferred from the hot gases is devoted to heating up the film. Many rocket engines are operating at supercritical conditions and therefore, research on film cooling under supercritical conditions would be beneficial.

4.4. *Liquid film entrainment studies*

Limited studies are available in literature which characterizes the mechanism of entrainment and film stability in liquid film cooling flows. Kinney et al. [78] had made visual observations of liquid film flows on the inner surface of the tubes containing flowing air. Water, water-detergent solutions, and aqueous ethylene glycol solutions were used as film coolants with air stream momentum flux varying from 40,000 to 200,000 Pa. Kinney et al. [78] observed disturbances at the liquid-gas interface causing loss of coolant, when the coolant flow rate was above a certain value. This value was found to be varying with liquid viscosity and surface tension, and did not show any change with air stream Reynolds number. Knuth [80] from his liquid-film stability experiments confirmed that longer wavelength disturbances appeared only after some critical flow rates of coolant. Small disturbances with wavelength of the order of ten times the film thickness were observed at all flow rates. However, liquid droplets were entrained by the gas stream from the crests of long wavelength disturbances. Contrary to the observations by Kinney and Knuth, Gater et al. [89] observed that the disturbances at the liquid-gas interface were dependent only on the momentum flux of the gas stream. He proposed that the quantity of liquid entrained was a function of momentum flux of the gas and surface tension of the liquid film. However, it may be noted that Gator's experiments were at lower gas mass flux conditions compared to earlier experiments. Coy et al. [105] conducted entrainment studies using slot injectors with a Mach number of the test section of about 0.6 and gas momentum fluxes varying from 30,000 Pa to 99,000 Pa. He concluded that there exists a critical flow rate of the coolant film beyond which any additional liquid injected would become entrained into the gas phase. Miller and Coy [106] had measured the thickness of liquid film driven by high momentum core gas flow. It was observed that at higher gas momentum fluxes, the entrainment depended more on viscosity than surface tension of the liquid film. However, a thorough understanding of the coolant entrainment mechanism is incomplete and uncertain at present and a detailed study in this aspect is essential to unravel the physical phenomenon behind the process. This can be due to the limitations of current instrumentation methods and the complexity of the entrainment process. The liquid-gas interface characteristics of liquid film cooling were investigated by Shine et al. [107]. It was observed that the disturbance waves started appearing at the liquid-gas interface at coolant flows above a critical value. The liquid-gas interface remained undisturbed for a fairly short distance of the order of diameter of the tube at low coolant flow rates. This undisturbed distance increased with the increase in momentum flux ratio (I). It was also observed that the disturbance wave frequency mainly depended on the core gas velocity, whereas the film thickness was a strong function of the momentum flux ratio.

Table 5 We_g values for film cooling experiments.

References	We_g
Morrell [77]	95,000
Kinney et al. [78]	5000–65000
Knuth [80]	80,000–2,00,000

In liquid film cooling flows, the liquid film flow is shear-driven and the liquid film-gas core aerodynamic interaction causes the liquid disintegration process. Therefore, a core flow Weber number, We_c representing the ratio of the disrupting aerodynamic force to the surface tension retaining force is usually used to characterize the flow conditions.

The weber number, We_g is defined as:

$$We_g = \frac{\rho_g {V_g}^2 D}{\sigma_c} \tag{14}$$

The We_g values calculated for the film cooling experiments are shown in Table 5. It is observed that liquid film cooling experiments are characterized by high aerodynamic forces which results in higher We_g values.

4.5. Reduction in specific impulse

A film of liquid or gas flow through rocket nozzle throat with a temperature different from core gases can result in performance losses [7]. Analysis by Coulbert [108] revealed that a typical performance loss due to film cooling is proportional to the quantity of coolant flow. The following studies have reported reduction in specific impulse with film cooling. Boden's [76] experiments showed reduction in specific impulse for all the film coolants. Except for low coolant flows, all film coolants caused lesser impulse reduction in comparison to an inert coolant. Abramson [79] had reported that specific impulse reduction of the film cooled engine was approximately 78% of the uncooled engine. The use of ammonia as a coolant resulted in a slightly higher specific impulse than was obtained with water as coolant. Morrell's [77] experiments showed a 4% reduction in specific impulse (I_{sp}) for water coolant flow of 5% of total flow, 4% reduction in I_{sp} for alcohol coolant flow of 15%, no reduction in I_{sp} for ammonia coolant up to 11%, and only 2% reduction in I_{sp} for an ammonia coolant flow of 15%. Stechman et al. [82] also noted performance loss with film cooling. He assumed no mixing of the coolant and mainstream and predicted the performance of the film cooled engine. The performance obtained from his experiments was within the accuracy of the values predicted. Arrington et al. [17] experiments with a standard conical nozzle and a bell nozzle showed higher performance of the nozzles with lower fuel film cooling. Film coolant injection will create a boundary layer film at a temperature different from the core gas flow. The net thrust developed by the engine will be less than that would result if the fluid had been thoroughly mixed. This gas stratification is the primary reason for the loss of specific impulse associated with film cooling. An additional specific impulse loss may be incurred due to the operation at propellant mixture ratios other than optimum in order to insure sufficient propellant as film coolant.

Film cooling with hydrogen, helium and ammonia is advantageous because low molecular weight will contribute to a high ratio of temperature to molecular weight without coolant combustion, and, hence, no appreciable performance degradation will occur. With film cooling, there are no heat flux limitations as observed with regenerative cooling or time constraints with ablative cooling. Therefore the complete cooling requirement of a rocket combustion chamber can be met from the film cooling process if one is willing to pay the penalty resulting from temperature stratification and the associated specific impulse loss. Other attractive features of this process are: (i) the applicability to a wide range of engine size, (ii) no limit to the duration of operation for film cooling, and (iii) no storage problems involved when the fuel is used as the coolant. However, transient operation of the film cooling involves key challenges. If the film is established before ignition, a fuel rich mixture in the combustion chamber will result. Starting the engine before the flow of coolant will result in a deposit of condensable combustion products on the wall, which will plug the coolant injector passages [94]. The stability characteristics associated with each coolant will be different and further research needs to be conducted to establish the guidelines during the starting and pulsating operation of the engine.

4.6. *Use of endothermic fuels*

Coolant fuels such as liquid hydrogen, liquid ammonia and liquid methane contribute cooling effects through the absorption of sensible and latent heat. A second category of fuels provide endothermic reactions and offer an additional heat sink for cooling. The endothermic reactions will provide gaseous, lighter components with high heating values, short ignition delay time and rapid burning rates. A large number of such reactions have been reported by Lander and Nixon [109]. Sobel and Spadaccini [110] had demonstrated the use of JP-7, JP-8 and JP-10 as endothermic fuel for hypersonic scramjet cooling. The use of endothermic fuels for rocket film cooling application is another area of potential research. Many liquid rocket engines use RP-1 as propellant which has narrow density and volatility range, lower sulphur, olefin and aromatic content than aviation fuels [111]. However, RP-1 film would crack into small molecules forcing solid carbon and might be harmful to combustion chamber walls [112]. It is reported that adding H_2O into hydrocarbon fuel can alleviate its coking behaviour especially at high temperature [113,114]. RP-1 mixed with 5%–10% hydrogen or water is proposed by Yang and Sun [115] for liquid rocket applications. Numerical simulations using RANS and k-ε model was conducted and proposed that this mixture had the potential to improve film cooling performance. An EDC model with 10 components and 17 step reaction is used for simulating the combustion. It is also reported that adding 5%–10% hydrogen to RP-1 coolant film reduced the specific impulse loss.

4.7. *Effect of thermal decomposition of coolant*

Crocco [87] was one of the first researchers investigated film cooling of a reactive coolant and showed theoretically that exothermic coolants reduce the film cooling effectiveness. Abramson's [79] experiments on film cooling of rocket nozzles showed that more coolant flow was required with a reactive coolant to cool the entire nozzle compared to an inert fluid. Welsh [94] noted that reactive film coolants caused less specific impulse reduction than would an inert coolant. Similar observations were obtained by Morrell [77] while performing film cooling experiments inside a combustion chamber of liquid ammonia-liquid oxygen rocket engine. He conducted film cooling experiments with water, ethyl alcohol and liquid ammonia as film coolants. Stechman et al. [82] observed that combustion chamber has to be operated fuel rich with high percentage of film cooling to reduce the specific impulse reduction. Takita [116] numerically studied cooling efficiency of hydrogen for a cylindrical body in supersonic air flow compared to cooling with nonreactive gases. He observed that cooling efficiency change is considerable according to whether a coolant is combustible or not. Kirk [117] studied surface heat flux augmentation due to near-wall reactions over a film cooled plate. He observed that the impact of near wall, secondary reactions on a film cooled surface is mainly a function of 5 non-dimensional groupings: Damkohler number (Da), heat release potential (H^*), scaled heat flux (Q_s), and mass and momentum blowing ratios. The near wall reactions will augment the surface heat flux which can be modelled as a convective heat transfer coefficient times the difference in driving (film) and the wall temperature. The driving temperature may approach the order of adiabatic flame temperature at high Damkohler number, however the changes in convective heat transfer coefficients are of lower magnitude. Jang et al. [118] investigated film cooling of combustion chamber and nozzle experimentally and analytically for a hydrogen peroxide/kerosene bipropellant thruster. He predicted the film cooled length by using the model proposed by Grisson [90] with a modification to account for the effect of the reactive coolant. The thermal decomposition of the hydrogen peroxide coolant was incorporated through an experimentally determined empirical constant. At higher coolant flow rates, higher rate of the thermal decomposition along with reduction in film cooled length was observed. These studies show that film cooling with a reactive fluid is significantly important and more knowledge of the process is vital to develop models which provide adequate levels of predictability of the film cooled length.

4.8. *Oxidizer as the coolant fluid*

Few studies have been conducted to investigate the possibility of the use of liquid oxygen for the cooling of thrust chambers. Dederra and Kirner [119] had conducted tests to study the feasibility of using liquid oxygen as a regenerative coolant. Price [120] observed that thrust chamber cracks observed with cyclic testing with LOX as the coolant have similar characteristics to those with liquid hydrogen as the coolant. He noted that in the event of small leak of coolant oxygen into the combustion chamber, the following might happen. (i) oxidation of carbon film at the chamber wall leading to higher chamber wall temperature. This could cause oxidation of the metal and a catastrophic failure. (ii) Liquid oxygen entering through cracks could film cool the carbon layer with no oxidation of carbon film or metal wall. Therefore further experimentations were suggested by Price [120] to confirm the two possibilities. The use of liquid oxygen as a film coolant may be an interesting scenario and needs to be explored in future.

5. Conclusions

Presently film cooling has become an important technology to tolerate high heat flux and high combustion temperature. This review has examined the origins of gaseous and liquid film cooling of rocket combustion chambers and summarized the research carried out so far, including experimental, analytical and numerical methods. The existing literature supports the idea of employing gaseous film cooling into the throat and divergent portions of the nozzle of a rocket engine. Correlations are now available to predict the film cooling effectiveness under varying injection conditions. The effect of variables such as blowing ratios, injection Mach number, injector configurations, accelerated hot gas, compressibility, pressure gradient etc. have been quantified. However in some cases, increase in heat transfer coefficient is noted which corresponds to lesser net heat flux reduction. Current level in understanding the nozzle side loads during transient operation is limited. The mechanisms that contribute to nozzle side loads due to film cooling during engine transition operations is to be identified and a more predictive ability has to be developed. Many studies have used subscale models to isolate the effects of different variables in their studies. Consequently many of the results have to be revaluated considering the actual operating environment of rocket engines.

Liquid film cooling is usually employed in the combustion chamber walls with fuel as coolant. The task of developing a complete understanding of liquid film cooling process starting from first principles is still far to be reached, however, considerable progress have been made. Models are available to predict the liquid film length with reasonable accuracy for engines working under subcritical conditions. Many present engines are operating at supercritical conditions and therefore further research is to be carried out in this regard. The shape of the coolant hole and the angle of injection on film cooling performance is another area that needs further understanding. Tangential injectors are used in most cases and the effect of other geometric factors has not been properly characterized. The current understanding of the liquid film entrainment effects and physics is relatively sparse; however, the acceptance of liquid film cooling is based on the available experimental data. It is also observed that there are only very few numerical investigations available pertaining to liquid film cooling. The current analytical models are very complex and contain many empirical correlations. Studies showed that film coolant entrainment results in significant loss of the coolant and a reduction in liquid film cooled length. It is clear that knowledge of this process is crucial, and systematic visualization tests needs to be carried out with sophisticated diagnostic techniques that are nowadays available.

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