Luft- und Raumfahrttechnik

Ghassen Yahiaoui

Film Cooling Investigations for Rocket Nozzle Flows with a New Test Facility Technique

Untersuchungen zur Filmkühlung in Raketendüsen mit einer neuen Versuchsanlagentechnik

SHAKER VERLAG

Film Cooling Investigations for Rocket Nozzle Flows with a New Test Facility Technique

Untersuchungen zur Filmkühlung in Raketendüsen mit einer neuen Versuchsanlagentechnik

Von der Fakultät für Maschinenwesen
der Rheinisch-Westfälischen Technischen Hochschule Aachen
zur Erlangung des akademischen Grades eines
Doktors der Ingenieurwissenschaften
genehmigte Dissertation

vorgelegt von

Ghassen Yahiaoui

Berichter:
Univ.-Prof. Dr.-Ing. H. Olivier
Univ.-Prof. Dr.-Ing. W. Koschel

Tag der mündlichen Prüfung: 14. März 2018.

Berichte aus der Luft- und Raumfahrttechnik

Ghassen Yahiaoui

Film Cooling Investigations for Rocket Nozzle Flows with a New Test Facility Technique

Untersuchungen zur Filmkühlung in Raketendüsen
mit einer neuen Versuchsanlagentechnik

Shaker Verlag
Aachen 2018

Bibliographic information published by the Deutsche Nationalbibliothek
The Deutsche Nationalbibliothek lists this publication in the Deutsche Nationalbibliografie; detailed bibliographic data are available in the Internet at http://dnb.d-nb.de.

Zugl.: D 82 (Diss. RWTH Aachen University, 2018)

Printed in Germany.

ISBN 978-3-8440-6033-1
ISSN 0945-2214

Shaker Verlag GmbH • P.O. BOX 101818 • D-52018 Aachen
Phone: 0049/2407/9596-0 • Telefax: 0049/2407/9596-9
Internet: www.shaker.de • e-mail: info@shaker.de

Acknowledgments

The present document has been written just after finishing my work as a scientific assistant and doctoral candidate at the Shock Wave Laboratory of the RWTH Aachen University, in December 2015. This work is an activity in the framework of the Sonderforschungsbereich Transregio 40 (SFB-TRR40), "Fundamental Technologies for Development of Future Space-Transport-System Components under High Thermal and Mechanical Loads". The financial support provided by the German Research Foundation is highly acknowledged. During my stay at the laboratory, I had the opportunity to work on various interesting topics and techniques related to supersonic flow problems such as the design and testing of reentry bodies, shock wave-body distances in high-enthalpy flows etc.

For my person, the working environment at the laboratory was in general harsh and hostile. Besides a short first period, I was mainly my self instructor and supervisor. Leading this project toward its end despite all encountered difficulties and obstacles required a huge amount of mental strength, sacrifices and endurance. Several attempts to earlier publish the findings of the second part of this work about supersonic film cooling experiments in a scientific journal were internally prevented. I am very proud having succeeded this project while preserving principle, personality as well as confidence. Moreover, I did not resign but I became determined, mentally stronger and forced the outcome despite the negative impacts which I had to accept while taking this path.

I owe thanks and acknowledgments to different people. For obvious reasons, I cannot acknowledge my indebtedness to all of them, and so I must necessarily limit my thanks to those who helped me directly during my time at the Shock Wave Laboratory. Therefore, I would like to acknowledge the help of my predecessor Maximilan Hombsch who introduced me to the main laboratory platforms such as the TH2 Shock Tunnel. With my colleagues, Maximilian, Yaşar Uygun and Christian Henkes I had a great time, inside as well as outside the institute.

Special thanks go to the entire workshop team with which I had a very close contact. The advices during the design and manufacturing phase of the facility helped me to early recognize possible difficulties, thus, to eliminate potential problems in an early stage. Therefore, I am very thankful to the members of this excellent team; Heinz Schobben, Aaron Heibges, Hans-Peter Michel, Yilmaz Şeker as well as Markus Eichler.

I would like to express my gratitude to Professor Wolfgang Koschel for reviewing this work as well as for his kind support and suggestions. The provided materials, such as his excellent lecture documents about space propulsion helped me to significantly shape and broaden my knowledge.

Finally, I would like to thank my family and friends for their unconditional support and encouragement to pursue my interests. My special thanks go to my mother and my father, Soraya and Fredj Yahiaoui for their encouragement, forbearance and support in this endeavor.

Abstract

To survive and prevail in a world of high economic competition, it is imperative for industrial manufacturers to enhance their products and push the optimizing capabilities to their limit. To achieve this goal, high technology companies which are operating in the aerospace sector are considerably investing in research and development. For example, rocket engines manufacturer are concerned with designing long-life reusable motors with increased thermodynamic cycle performances and less weight. This allows launcher operators to increase their payload offer and/or accomplish longer missions, making them more competitive. However, increasing the performance of rocket motors is a very challenging task due to the excessive thermal loads imposed to the flow path materials. These materials are generally exposed to gas temperatures exceeding by far the limit of their structural integrity. Thus, to guarantee its survivability, innovative thermal management techniques are necessary. Among these techniques, film cooling is a promising cooling method which has been implemented in several engines such as the Vulcain 2 and J-2X. Film cooling has been studied since early years, however, its mechanics is not yet well understood especially in complex flow regimes. Therefore, it remains an actual research topic where considerable effort in research is invested.

Consequently, the rising demand for reliable supersonic film cooling (SSFC) data for rocket nozzle design applications led to the development of a new type of facility. Most of the former experimental data on SSFC in the literature were gathered in conditions, far from that encountered in real engines and combustion gas generators. Thus, they are not appropriate for the validation of computational tools and the resulting effectiveness correlations are inaccurate in the prediction of film cooling performances. The new experimental platform developed within this thesis work is intended to solve this problem by its capability to nearly match most of these conditions. Furthermore, it allows for the investigation of several engine relevant aerothermodynamic problems such as nozzle flow separation and nozzle exhaust plume interactions etc.

In the new facility, high nozzle reservoir pressures and temperatures are achieved by means of the detonative combustion of a premixed gas in a confined tube. A detonation wave produces a high-enthalpy flow which expands through a nozzle to the desired experimental flow conditions. The facility simulates engine-like flow conditions in a combustion environment for various oxidizer-fuel mixture ratios. It is therefore capable to reproduce realistic gas-to-wall temperature ratios similar to that encountered in combustion chambers, turbine blades, power plants, rocket engines etc. if the wall temperatures are matched. This way, the new short duration flow device affords a simple and cost-effective research tool for harsh flow environment testing in laboratories. Its intermittent nature allows for a relative large number of tests within a short period of time in chamber conditions which are not feasible with conventional wind tunnels. Hence, it will signifi-

cantly contribute to the understanding of complex rocket nozzle flow phenomena, the validation of numerical tools as well as correlations.

Kurzfassung

Um in einer Welt mit hohem wirtschaftlichen Wettbewerb bestehen zu können, ist es für industrielle Hersteller unerlässlich, ihre Produkte kontinuierlich zu verbessern und die Optimierungsmöglichkeiten an ihre Grenzen zu bringen. Um dieses Ziel zu erreichen, investieren Hightechunternehmen, die in der Luft- und Raumfahrtbranche tätig sind, erheblich in Forschung und Entwicklung. Zum Beispiel befassen sich Raketentriebwerkshersteller mit der Konstruktion langlebiger wiederverwendbarer Motoren mit erhöhten thermodynamischen Zyklusleistungen und weniger Gewicht. Dies ermöglicht Betreibern von Trägerraketen ihr Nutzlastangebot zu erweitern und/oder längere Missionen durchzuführen, wodurch sie dadurch wettbewerbsfähiger werden. Die Erhöhung der Leistung von Raketenmotoren ist jedoch eine sehr schwierige Aufgabe aufgrund der übermäßigen thermischen Belastungen, denen die Materialien des Strömungswegs ausgesetzt sind. Im Allgemeinen sind diese Materialien Gastemperaturen ausgesetzt, die die Grenze ihrer strukturellen Integrität überschreiten. Um eine zuverlässige Betriebsfähigkeit zu gewährleisten, sind innovative Wärmemanagementmethoden erforderlich. Unter diesen Methoden ist die Filmkühlung eine vielversprechende Kühlstrategie, die in mehreren Motoren wie dem Vulcain 2 und J-2X umgesetzt wurde. Die Filmkühlung wurde seit langer Zeit untersucht, ihre Mechanik ist jedoch insbesondere in komplexen Strömungsumgebungen noch nicht ausreichend verstanden. Daher bleibt es ein aktuelles Forschungsthema, in das erhebliche Forschungsaufwendungen investiert werden.

Folglich führte die steigende Nachfrage nach zuverlässigen Daten der Überschallfilmkühlung für Raketendüsendesigns zur Entwicklung eines neuen Anlagentyps. Denn die meisten der früheren experimentellen Daten über Überschallfilmkühlung in der Literatur wurden unter Strömungsbedingungen erzeugt, die denen in realen Motoren und Gasgeneratoren nicht entsprechen. Daher sind die für die Validierung von Strömungslöser nicht geeignet und die resultierenden Korrelationen für die Kühleffektivität sind ungenau bei der Vorhersage der Filmkühlungsgüte. Die neue Anlage, die im Rahmen dieser Arbeit entwickelt wurde, soll dieses Problem lösen, indem es die meisten dieser Strömungsbedingungen nahezu emuliert. Darüber hinaus ermöglicht es die Untersuchung von verschiedenen motorrelevanten aerothermodynamischen Problemen, wie z. B. die Strömungsablösung in der Düse als auch Wechselwirkungen im Düsenstrahl usw.

In der neuen Anlage werden hohe Totaldrücke und Temperaturen durch die detonative Verbrennung eines vorgemischten Gases in einem geschlossenen Rohr erreicht. Eine Detonationswelle erzeugt eine Hochenthalpieströmung, die sich durch eine Düse auf die gewünschten experimentellen Strömungsbedingungen einstellt. Die Einrichtung erzeugt motorähnliche Strömungsbedingungen in einer Verbrennungsumgebung für verschiedene Mischungsverhältnisse zwischen Brennstoff und Oxidator. Es ist daher in der Lage, realistische Gas-zu-Wand-Temperaturverhältnisse ähnlich denen in Brennkammern, Turbinenschaufeln, Kraftwerken, Raketentriebwerken usw. zu reproduzieren, wenn die Wandtemperaturen übereinstimmen. Auf diese Weise bietet die neue Kurzzeitströ-

mungsanlage eine einfache und kosteneffektive Forschungsplattform für Versuche unter extreme Strömungsbedingungen in Laboratorien. Die intermittierende Natur der Anlage erlaubt eine relativ große Anzahl von Tests innerhalb eines kurzen Zeitraums unter Reservoirbedingungen, die mit herkömmlichen Windkanälen nicht möglich sind. Damit wird es wesentlich zum Verständnis komplexer Raketendüsenströmungsphänomene, zur Validierung numerischer Strömungslöser sowie Korrelationsmodelle beitragen.

Contents

List of Figures

List of Tables

Nomenclature

Latin symbols

A	[m^2]	Cross-sectional area
a	[m/s]	Speed of sound
b_n	[m]	Nozzle width
c^*	[m/s]	Characteristic velocity
c_D	[−]	Nozzle discharge coefficient, $c_D = \dot{m}_r/\dot{m}_{is}$
c	[−]	A constant
C_f	[−]	Low-speed adiabatic skin-friction coefficient, $2\tau_w/\rho_e u_e^2$
c_p	[J/ (kg K)]	Specific heat at constant pressure, $\gamma R/\left(\gamma - 1\right)$
d	[m]	Hydraulic diameter of circular duct
F	[−]	Blowing ratio, $F = \rho_c u_c/\rho_e u_e$
g_0	[m^2/s]	Gravitational acceleration at earth's surface, $g_0 \approx 9.81\,\mathrm{m^2/s}$
h	[J/kg]	Specific enthalpy
h^0	[J/kg]	Heat of reaction extrapolated to zero temperature
h	[m]	Nozzle height
h_g	[W/ (m^2 K)]	Convective heat-transfer coefficient of the bulk-flow
I_{sp}	[s]	Specific impulse
K_p	[−]	Acceleration parameter, Eq. (5.3)
k	[W/ (m K)]	Thermal conductivity
L	[m]	Length
L_n	[m]	Length of divergent nozzle section
Ma	[−]	Mach number
Ma_C	[−]	Convective Mach number, $Ma_C = \left(u_e - u_c\right)/\left(a_e + a_c\right)$
$\dot{m}$	[kg/s]	Mass flow rate
Nu	[−]	Nusselt number, Eq. (1.7)
Pr	[−]	Molecular Prandtl number, $Pr = \nu/\alpha$
p	[Pa]	Pressure
Q	[−]	Non-dimensional heat of reaction
$\mathbf{q}$	[J/kg]	Heat of combustion
$\dot{q}$	[W/m^2]	Heat flux density
R	[mm]	Radius from nozzle axis to wall
R	[J/ (Kg K)]	Mixture specific gas constant
$\Re$	[J/ (mol K)]	Universal gas constant, $\Re = 8.314\,\mathrm{J/mol\,K}$
Re	[1/m]	Unit Reynolds number
Re_θ	[−]	Reynolds number based on momentum thickness, $u_e\theta/\nu_e$

Re_c	[−]	Coolant film Reynolds number based on slot height, $u_c s/\nu_c$
Re_Δ	[−]	Reynolds number based on energy thickness, $u_e \Delta/\nu_e$
Re_d	[−]	Reynolds number based on hydraulic diameter, $u_e d/\nu_e$
Re_x	[−]	Reynolds number based on distance along wall, $u_e x/\nu_e$
r	[−]	Adiabatic recovery factor
r_c	[mm]	Radius of axial curvature at the throat
S	[K]	Sutherland constant
St	[−]	Stanton number, Eq. (1.8)
s	[J/ (kg K)]	Specific entropy
s	[mm]	Blowing slot height
T	[K]	Temperature
T_{aw}	[K]	Adiabatic wall temperature
T_r	[K]	Bulk-flow recovery temperature
T_{ref}	[K]	Reference temperature
Tu	[−]	Turbulence intensity
t	[s]	Time
t_s	[mm]	Splitter thickness
u, v, w	[m/s]	Mean velocity components in x, y and z directions
V_v	[m^3]	Volume of vacuum tank
$\overline{W}$	[kg/kmol]	Mixture molecular weight
W	[kg/kmol]	Molecular weight
w	[mm]	Cooled width (slot circumference in cylindrical duct)
X_s	[mm]	Virtual streamwise length of the boundary layer, Eq. (2.20)
X_c	[mm]	Virtual streamwise length of the boundary layer, Eq. (2.25)
X	[−]	Mass fraction of species
x	[m]	Axial distance to throat
x_e	[m]	Effective distance, Eq. (1.22)
x_s	[m]	Distance along wall originating at the blowing slot
y	[m]	Normal wall distance
z	[−]	Dimensionless enthalpy, Eq. (1.24)

Greek symbols

α	[μV/K]	Seebeck coefficient
α	[deg]	Nozzle expansion half-angle
α_d	[m^2/s]	Thermal diffusivity, $\alpha_d = k/(\rho c_p)$
β	[$W\sqrt{s}/(K\,m^2)$]	Thermal effusivity, $\beta = \sqrt{\rho c k}$
Γ	[−]	A gas constant defined by Eq. (1.2)
$\mathcal{E}$	[−]	Expansion area ratio, $\mathcal{E} = A/A_t$
ϵ	[−]	Emission coefficient
ω	[−]	Exponent of temperature dependence of viscosity
σ	[$W/(m^2\,K^4)$]	Stefan-Boltzmann constant, $\sigma = 5.670367(13) \cdot 10^{-8}\,W/(m^2\,K^4)$
γ	[−]	Isentropic exponent, $\gamma = c_p/c_v$
τ	[s]	Nominal test time
Δ	[mm]	Boundary-layer energy thickness
δ	[mm]	Boundary-layer thickness
δ^*	[mm]	Displacement thickness
θ	[mm]	Momentum thickness

μ	[Pa s]	Dynamic viscosity
ν	[m^2/s]	Kinematic viscosity, μ/ρ
ϕ_i	[−]	Constant defined by Eq. (3.54)
φ_m	[−]	Turbulent mixing coefficient
ρ	[kg/m^3]	Density (mass per unit volume)
η	[−]	Cooling efficiency
ξ	[−]	Correlation parameter
π	[−]	Pythagorean constant, $\pi = 3.14159...$

Subscripts

a	Ambient condition
aw	Adiabatic wall condition
bl	Boundary layer
c	Coolant film at point of injection, with coolant injection
cc	Combustion chamber
CJ	Chapman-Jouguet state
d	Detonation wave
e	Quantity at the hot-flow boundary layer edge or nozzle exit plane
h	Hot-gas
hm	Hot-gas mixed with coolant film
i	Species index
iw	Isothermal wall condition
nc	Without cooling
r	Reflected shock wave
ref	Reference condition or value
s	Injection slot
T	Detonation tube
t	Nozzle throat
w	Wall value
0	Stagnation value
1	Initial, undisturbed region in the detonation tube
2	Region immediately behind detonation wave after reaction is complete
5	Region behind reflected shock wave

Superscripts

$< . >$	Averaged quantity
$\overline{(.)}$	Mass-weighted-averaged quantity
$(.)^*$	Reference value

Acronyms

BK7	Optical Borosilicate Glass
BNC	Bayonet Neill-Concelman
CEA	Chemical Equilibrium with Applications

CFD	Computational Fluid Dynamics
DAQ	Data Acquisition
DDT	Deflagration to Detonation Transition
DNS	Direct Numerical Simulation
FLIR	Forward-Looking Infrared
GOX	Gaseous Oxygen
JANAF	Joint Army, Navy, Air Force
JP-10	Jet Propellant-10, (exo-tetrahydrodicyclopentadiene, $C_{10}H_{16}$)
KASIMIR	Kanalsimulation im Rechner
LabVIEW	Laboratory Virtual Instrument Engineering Workbench
LES	Large Eddy Simulation
LOX	Liquid Oxygen
LOCI-CHEM	Density-Based Navier-Stokes Solver
LRE	Liquid Rocket Engine
MATLAB	Matrix Laboratory
MMH	Monomethylhydrazine, $CH_3N_2H_3$
NASA	National Aeronautics and Space Administration
NIST	National Institute of Standards and Technology
NTO	Nitrogen Tetroxide (Dinitrogen), N_2O_4
PCI	Peripheral Component Interconnect
PDE	Pulse Detonation Engine
PXI	PCI eXtensions for Instrumentation
RANS	Reynolds-Averaged Navier-Stokes
ROCFLAM	Rocket Flow Analysis Module
ROF	Mass Ratio of Oxidizer to Fuel
SSFC	Supersonic Film Cooling
SSME	Space Shuttle Main Engine
SWL	Shock Wave Laboratory
TDK	Two-Dimensional Kinetics
TTL	Transistor-Transistor Logic
UDMH	Unsymmetrical Dimethylhydrazine, $C_2H_8N_2$