Chapter 9
Aerospace Flight Modeling and Experimental Testing

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Abstract Validation processes for aerospace flight modeling require to articulate uncertainty quantification methods with the experimental approach. On this note, the specific strategies for the reproduction of re-entry flow conditions in ground-based facilities are reviewed. It shows how it combines high-speed flow physics with the hypersonic wind tunnel capabilities.

9.1 Introduction

Space missions are built upon massive technology knowledge and on the latest progress in engineering. They fascinate as they represent for the public the most advanced knowledge as well as a typical dive into the unknown. For scientists and engineers, such missions are an occasion to push the scientific knowledge and to establish better what we know to offer solid basis for further discoveries.

In practice, space exploration leads to extreme challenges as it aims to investigate planets, or asteroids, in the solar system and return samples for analysis across very severe flight conditions. Such missions need to be designed using physical models and robust numerical methods. However, those tools used by aerospace engineers remain, for most of them, on the need for more research development for their validation and consolidation to be able to plan successful and fruitful missions.

Aerothermodynamic testing is one of the crucial points for the design of aerospace vehicles. At first place, it aims at establishing as much as knowledge possible on critical flight phenomena. Ground-based facilities are operated to reproduce flight-relevant environment for the testing of the vehicle configuration and its Thermal Protection System (TPS) to allow for an accurate evaluation of their performances. Two types of facilities are classically used for the ground testing to support the pre-flight analysis. First, the required flow-field is reproduced for its analysis in

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L. Aslett et al. (eds.), *Uncertainty in Engineering*, SpringerBriefs in Statistics,
https://doi.org/10.1007/978-3-030-83640-5_9

high-enthalpy facilities such as shock tunnels or expansion tubes. Then in a second step, the interaction between the dissociated gas and the vehicle' surface is studied to determine the thickness of Thermal Protection Material (TPM) from databases built in plasma wind tunnels. Accurate flight duplication is thus necessary in order to properly address those stringent requirements without over-sizing the TPS.

All the data produced are processed to define at best the modeling for all the required physical phenomena. Very performant frameworks for such processing are the Uncertainty Quantification (UQ) methods. Bayesian approach, in particular, are extremely powerful as they allow to determine the required information from a limited experimental knowledge with a probabilistic treatment. However, such key results, for hypersonic flight, are not only due to powerful mathematical treatment but also thanks to consistent experimental methodologies. This combination between mathematical and experimental approach is essential to generate useful knowledge on the physical modeling to be determined. Then it is primordial to have a good understanding on how the ground testing are linked to the fundamental mathematical models used for simulating the high-speed flow physics. It serves to setup the best experimental environment to allow a fruitful processing of the acquired data.

Therefore, this note intends to review the rationale of the ground testing methodology for aerospace vehicle. It offers a synthesis on the high-speed ground testing underlying the links to their scientific basis. It presents how similitude laws could be applied or need to be adapted as more and more physical phenomena have to be considered for aerospace flights. It would help engineers and applied mathematicians for working together facing the challenges of high-speed flight investigations for aerospace development.

9.2 Aerospace Flights and Planetary Re-entry

Aerospace flight can be considered for very different trajectories. It could follow a planet orbit up to the limit of its escape velocity, but it could also correspond to an interplanetary transfer with super-orbital velocities. Those situations involve a large variety of kinetic energy and trajectories. Figure 9.1 illustrates those typical orbit trajectories, around Earth, with their corresponding velocities.

Super-orbital atmospheric re-entry, also known as hyperbolic re-entry, is characterized by high velocities and is encountered when a probe is entering an atmosphere from a hyperbolic orbit rather than from an elliptical orbit. It is the case for some interplanetary probes or sample return missions. Typical velocities for probes entering the Earth's atmosphere from hyperbolic orbits scale from 11 to 14 km/s, which correspond to specific enthalpies between 60 and 100 MJ/kg. This is considerably higher than the usual re-entry velocities from circular or elliptic orbits, e.g., 8.2 km/s for the Space Shuttle. Up to now, the Stardust probe was the fastest artificial object to perform a controlled re-entry in the Earth's atmosphere, at 12.8 km/s.

The environment of a high-speed re-entry flight is much more severe compare to a Low Earth Orbit (LEO) entry with particularly high heat flux and important stagnation

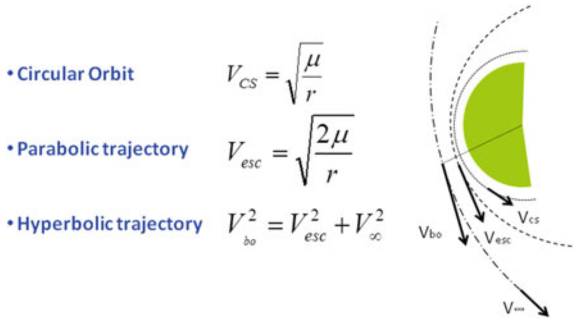

Fig. 9.1 Orbit trajectories and velocities. μ is the Earth gravitational parameter and r is the radius of the orbit

pressure. In addition to the classical features of re-entry flows like non-equilibrium thermo-chemistry and complex gas–surface interactions some specific phenomena become important as shock layer radiative heating and ablation phenomena. Those physical process were not considered much at smaller velocity but they cannot be anymore neglected for super-orbital re-entry. In top of this, more complex physical reality coupling phenomena appear in the flow between radiation and ablation that lead to a very intricate situation. It is therefore not possible to extrapolate what has been studied and learned for orbital re-entry to super-orbital conditions and specific ground testing strategies are required.

9.3 Similitude Approach for Hypersonic Flows

Similitude in classical fluid dynamics establishes a correspondence between different flows, based on the mathematical model representing these flows, without having to solve the set of equations. This correspondence then can be used to relate two real physical flow situations or to relate a family of solutions for the model.

With two correlated applications:

- The flow fields are similar (i.e., solution of the same set of equations) even if the dimensions and the temporal evolution of the phenomena are on different scales.
- When applicable, the use of the similarity laws allows to replicate in wind tunnels the flow field occurring in flight around a re-entry vehicle.

This second aspect of the similitude approach is mostly useful since it allows to study on ground typical re-entry situations. Hypersonic flows are particularly interesting for the application of similitude approach because it exists many different situations in hypersonic regime which can be describes with a variety of models leading to different similitude laws. On one hand, it gives opportunities to develop simplify solutions, but on the other hand, the physical nature of the high-speed flows, mostly due to the high-temperature effects, severely restrict exact similitude and impose to study approximate similitude. This family of flows is essentially determined by the mathematical model chosen to describe the flows and of which the flows are themselves solutions. It appears that the similitude strategy evolves as the model integrates more and more physical aspects for the description of the flow. In hypersonics going from inviscid regime to high-temperature effects, the similitude approach will be adapted to retain the most relevant flow parameters corresponding to each case. Similitude in hypersonic has been studied and discussed by different authors: in earlier time, by Hayes and Probstein for inviscid and viscous hypersonics [11, 12], by Freeman for dissociated gases [9] and in a more general form by Viviand for CFD development [6], but it provides also very useful material for experimental studies. The most common approaches are briefly presented below, as well as their limitations.

9.3.1 Inviscid Hypersonics

At high Mach number when considering a slender body (small thickness ratio $\tau << 1$) at small angle of attack ($\alpha << 1$) in a perfect gas, the governing Euler equations can be further simplified using the small disturbances theory. The similarity parameters are then $M\tau$, $\alpha\tau$, and γ. The slenderness ratio τ is defined as $\tau = d/l$, where d is the body's radius and l its length. The parameter $K = M\tau$ is called the hypersonic similarity parameter [11]. On these conditions, with a constant isentropic exponent γ, the results of the similitude may be expressed in dimensionless form. For family of affinely related bodies of thickness ratio, τ in two-dimensional flows the pressure coefficient C_p could be written:

$$\frac{C_p}{\tau^2} = 2\left[\frac{\gamma+1}{4} + \sqrt{\left(\frac{\gamma+1}{4}\right)^2 + \frac{1}{K^2}}\right] = f(K,\gamma) \tag{9.1}$$

The typical testing strategy in this inviscid framework could be represented as in the figure (Fig. 9.2).

That approach holds for inviscid hypersonic, over a wide range of K for slender bodies as long as the Mach number in the flow is large enough and τ small. Nevertheless, it does not correlate well as the body thickness is increased, curved shock and boundary layers start to be predominant in the flow. As most hypersonic vehicles are blunt rather than slender for thermal considerations this hypersonic similarity has limited applications.

9.3.2 Viscous Hypersonics

Hypersonic flows present thick boundary layers that are also growing fast as Mach number is increased. These boundary layers cannot be ignored generally in hypersonic problems as they determine the major feature of the flow physics. They also

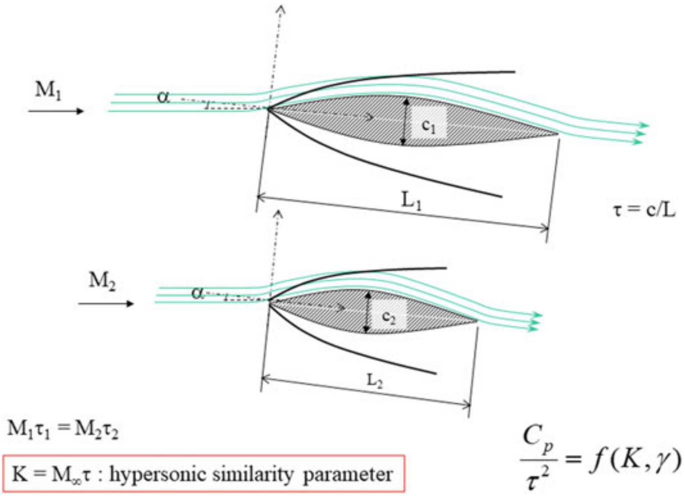

Fig. 9.2 Inviscid hypersonic similitude

need to be taking into account for their interaction with the inviscid flow; ground-based facilities need to be designed toward those considerations to provide a way for studying these effects in relevant hypersonic flight conditions. To this end, similitude approach is a precious guide to identify which combination of flow parameters need to be taken into account in experimental simulation. Viscous hypersonic similitude has been presented first by Hayes and Probstein in a paper [12] where they review the general features of viscous similitude at high speed. The inviscid flow need to be represented, then the hypersonic similarity parameter $K = M\tau$ and γ need to be invariant. The interaction of the viscous part of the flow with the inviscid field will be determined by the viscous-inviscid interaction parameter χ expressed as

$$\chi = \frac{M_{\infty}^{3}\sqrt{C_{\infty}}}{\sqrt{Re_{x},\infty}} \tag{9.2}$$

with $C_{\infty} = \frac{\mu_r T_{\infty}}{\mu_{\infty} T_r}$ and $Re_{x,\infty} = \frac{\rho_{\infty} U x}{\mu_{\infty}}$.

With these conditions the perfect gas model has to be assumed and the continuum hypothesis valid, i.e., the mean free path is at least one order of magnitude smaller than the characteristic length of the flow. The viscous hypersonic similarity requires reproducing the free-stream Reynolds (Re) and Mach number (M) and the temperature ratio $\frac{T_w}{T_{\infty}}$, where the subscript w indicates the wall temperature and the ∞ symbol in the subscript refers to the free-stream. If the gas mixture is not the same, the heat capacity ratio γ also needs to be reproduced [13]. This eases facility development, as lowering the gas temperature lowers the speed of sound, and thereby increases the achievable free-stream Mach number. The condensation temperature of the test

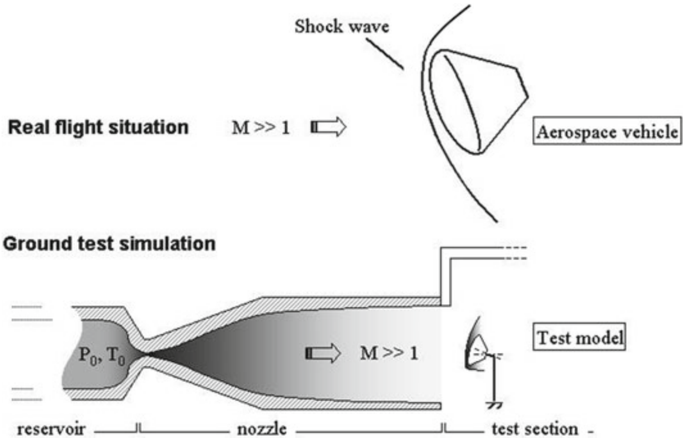

Fig. 9.3 Schematic representation of cold hypersonic testing in ground facility

gas imposes the upper limit of what is achievable in a given facility. In a general view it resorts a Mach–Reynold simulation. Most of the ground testing capabilities are designed following this principle working with scaled models (Fig. 9.3). Their operation envelope are commonly presented in a Ma–Re graph that indicates which flight domain can be simulated (Fig. 9.4).[1] One has to remark that the temperature ratio parameter $\frac{T_w}{T_\infty}$ is not often taking into account but it could be an important aspect to consider in ground testing as it could appear as a limitation in the study.

9.3.3 High-Temperature Hypersonics

When a real gas, as air, experiences a strong shock at high speed, it will increase tremendously its thermal energy. In such conditions, the molecular collisions are energetic enough to cause dissociation and ionization and the gas to depart from the perfect gas model. For a blunt body at a velocity of 7 km/s, the temperature immediately after the shock is around 14,000 K, and around 8,000 K downstream the shock, where the flow may return to equilibrium. At such high temperatures, chemical effects have to be taken into account. For air at a pressure of 1 atm, vibrational excitation begins at 800 K, O2 begins to dissociate at 2,500 K and is fully dissociated for 4,000 K, point for which N_2 begins to dissociate. At 9,000 K, N_2 is fully dissociated and ionization begins. One can easily understand that the flow downstream the shock becomes a plasma: molecules are dissociated and atoms are partially ionized. The parameters to be respected for a flow involving chemistry are $V^2/2D$, where V is the free-stream velocity and D the typical dissociation energy of the gas molecule considered, the Damköhler number Da, defined as $Da = L/l_D$,

[1] Graphic extracted from AGARD AR 319.

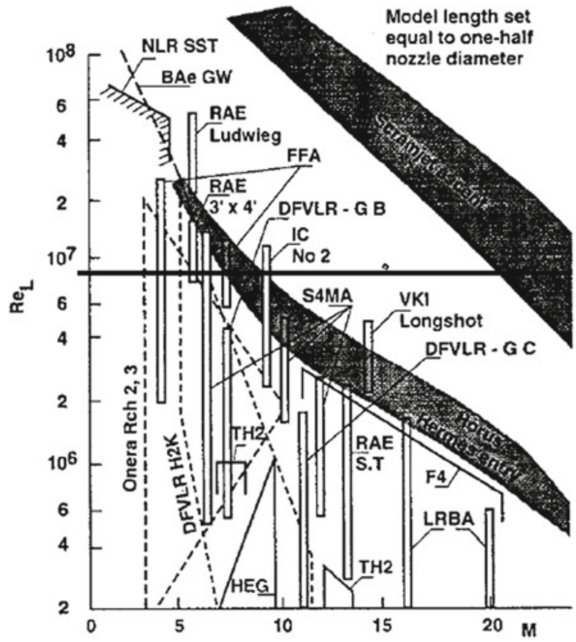

Fig. 9.4 Mach–Reynolds map for hypersonic facilities [20]

where L is the characteristic length of the flow and l_D the characteristic length associated to the dissociation reaction, and the temperature ratio T_w/T [13]. It should be brought to the reader's attention that the gas behind the shock is a chemically reacting mixture of perfect gases, and not a real gas as it is sometime incorrectly referred to in literature.

The Damköhler number for the gas appears considering the mass conservation equation in a non-dimensional form as it is expressed in the equation below:

$$\frac{\partial \rho_i}{\partial t} + \nabla.(\rho_i V + J_i) = \sum_{r=1}^{N_r}(\nu_{ir}^{''} - \nu_{ir}^{'})\left\{\text{Da}_{fr}\prod_{j=1}^{N_s}\rho_j^{\nu_{ir}^{'}} - \text{Da}_{bw}\prod_{j=1}^{N_s}\rho_j^{\nu_{ir}^{''}}\right\} \tag{9.3}$$

$$\text{Da}_{fr} = \frac{k_{fr}\rho_\infty L}{U_\infty} \qquad \text{Da}_{bw} = \frac{k_{bw}\rho_\infty^2 L}{U_\infty} \tag{9.4}$$

Chemical reactions take a very short but finite time to happen. Assuming that the flow is composed of a single species, the dissociation rate is proportional to the density, while the recombination rate is proportional to the square of density. Hence, the characteristic lengths associated to the dissociation and recombination reactions, respectively, scale as $l_D \propto 1/\rho$ and $l_R \propto 1/\rho^2$, where ρ is the flow density [13]. The recombination length is usually larger than the dissociation length. Under certain conditions, at very high altitude, one can assume that $l_D \sim L$, and

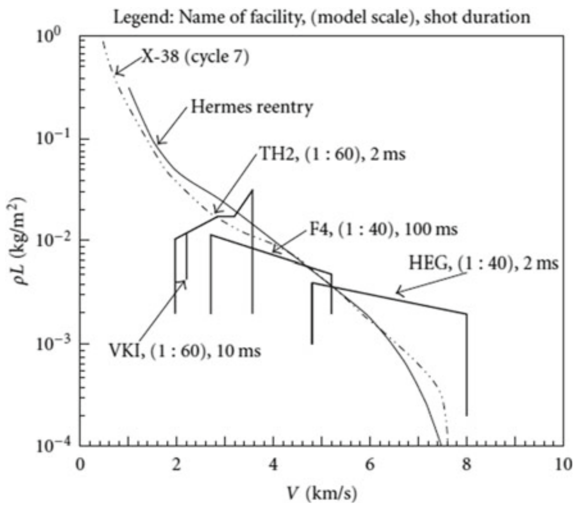

Fig. 9.5 High-enthalpy facility map [15]

therefore $Da = o(1)$, while l_R is much larger. The flow is frozen; it is too fast for recombination reactions to take place. In that case, the binary scaling parameter ρL must be reproduced in order to obtain the correct Da [16]. One could note that the diffusion phenomena is also considered in this analysis, as the diffusion term in the equation scale with ρL [5]. With this approach, the high-enthalpy facilities are designed to reproduce the real flight velocities (V_∞) and to allow for an operation considering the ρL parameter as they can be represented in the graph of Fig. 9.5.

This leads to some complications. Firstly, the same air mixture as in real flight is commonly used, as the chemistry processes are too complicated to reproduce to use another one. The typical dissociation energy D is therefore conserved, and the actual free-stream velocity must be reproduced to duplicate the group $V^2/2D$. Secondly, the required density to achieve in the wind tunnel becomes large in order to maintain the proper value of the binary scaling parameter for duplication of flight at lower altitude. Thirdly, the binary scaling approach, strictly speaking, is built upon the hypothesis of a single species mixture. That approach has therefore limited application for more complex mixtures such as air [7]. Finally, as altitude decreases, density increases and both l_D and l_R become smaller. The binary scaling parameter does not hold anymore, as both ρL and $\rho^2 L$ should be reproduced at the same time. The same holds when flow velocity increases, and therefore also temperature. This prevents from using the similarity laws, and flow duplication can thus only be performed on a full-scale model, with the actual flow velocity.

9.4 Duplication of Dissociated Boundary Layer with Surface Reaction

Following the development presented in the previous paragraph, it is observed that in spite of all the possibilities offer by the ground testing, considering the high-temperatures effects in hypersonic, only the dissociation in the shock layer could be simulated to some extend on scaled models. If one wanted to further take into account, the recombination in the gas he would have to consider full-scale models. Looking into more details, one could underline that all the important phenomena, concerning the heat transfer typically, are laying in the boundary layer. Figure 9.6 gives an illustration of real flight situation in front of an Aerospace vehicle.

The study could then be reduced to this confined layer in particular for the heat-transfer problems in hypersonic. In this situation, the full-scale environment will be reproduced by duplicating all the characteristics of the boundary layer in the ground testing facilities. This statement was already made in earlier publications from researchers working on dissociated boundary layers using shock tube facilities [19].

The stagnation point is of particular interest for this duplication because the flow conditions, returning to zero velocity, are more easily reproduced in a laboratory. Before identifying the parameters that need to be retained for the testing conditions it will be useful to give a physical description of reacting boundary layers, as they manifest themselves along surfaces in hypersonic flows. Dissociated non-equilibrium boundary layers with surface reaction have already been presented extensively by major authors [2, 19] and the reader is encouraged to consult these references. The

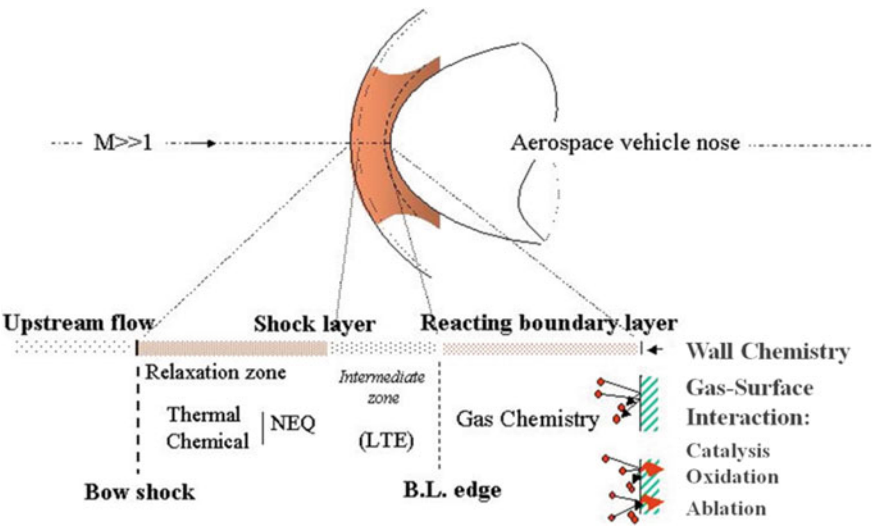

Fig. 9.6 Re-entry environment along stagnation line

purpose of this section is not to expose in details the theory specific to such boundary layers but to recall their main characteristics and better understand how they could be simulated in ground-based facilities. Boundary layers are the location where the diffusion phenomena are dominant, their relative importance are characterized by typical non-dimensional numbers: Prandtl (Pr), Lewis (Le), and Schmidt (Sc). Those will not be commented here to rather focus on the chemical non-equilibrium feature of the flow. The chemical reactions taking place in the gas phase are called homogeneous reactions while those happening between the gas and the solid surface are called heterogeneous reactions. The chemical non-equilibrium in the gas phase is characterized by the gas Damköhler number (Da_g). It corresponds to the ratio between the typical time of the flow, to cross the boundary layer, to a typical reaction time for the gas chemistry: $Da_g = \tau_f/\tau_c$. When $Da_g \rightarrow \infty$ the boundary layer reaches a local thermodynamic equilibrium (LTE). On the contrary when $Da_g \rightarrow 0$ it leads to a frozen boundary layer, where no chemistry happen. These different conditions have significant consequences on the wall heat flux as it has been shown in reference [8]. The reactions at the wall are usually considered as first-order reactions, they are represented by a reaction rate (k_w) for each dissociated specie. k_w could be related to a recombination coefficient γ_i (or catalycity parameter), interpreted as a probability of recombination at the wall, by the Hertz–Knudsen formula:

$$k_w i = \gamma_i \sqrt{\frac{k \cdot T_w}{2\pi \cdot M_i}} \tag{9.5}$$

The wall reaction rates are also characterized by a surface Damkohler number (Da_w). It compares the time of diffusion for the species across the boundary layer to the time of reaction at the wall: $Da_w = \tau_{Diff}/\tau_{React}$. When $Da_w \rightarrow \infty$, it does not necessarily imply that the reaction rate at the wall tend to infinity ($kw \rightarrow \infty$), but simply that the surface reaction are much more faster than the diffusion process. It is said that the GSI phenomena are "limited by diffusion". In the other extreme case, when $Da_w \rightarrow 0$ the diffusion is much faster than the reaction and the GSI phenomena are "limited by reaction" or "reaction controlled". From this description, it appears that the diffusion and reaction processes should be accurately reproduced. To this extend, it could be understood that the dimension and the environment of the reaction boundary layer must be duplicated (Fig. 9.7). Two situations could be distinguished for the boundary layer to be duplicated in the laboratory: stagnation point region and off-stagnation point when the boundary is developing along the surface.

If one considers only the stagnation point region, it has been shown that a complete duplication of real flight condition is possible in ground facility, if the total enthalpy (He), the total pressure (Pe), and the velocity gradient ($\beta = du/dx$), of the flight conditions, can be matched locally on the test sample [4, 14]. In this case, the testing is realized in plasma wind tunnels (Arcjet or Plasmatron facilities) that are able to produce dissociated flows for a long time base which is suitable for tests involving aerothermochemistry. The theoretical frame for the testing with ICP wind tunnels has been adapted by Russian scientists, in a methodology called Local Heat Transfer

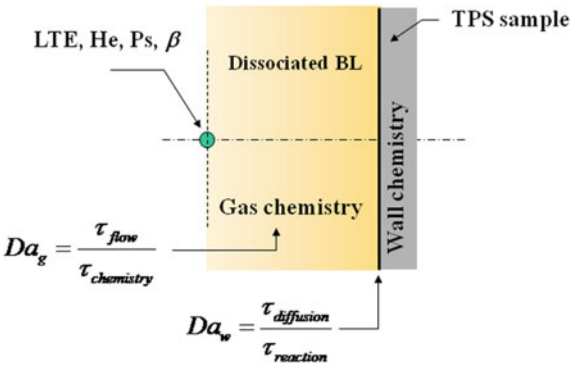

Fig. 9.7 Typical features of the dissociated stagnation point boundary layer

Simulation (LHTS) [14] and an assessment of this methodology has been conducted at the VKI Plasmatron facility [1, 4]. The duplication of the flight condition at stagnation point is strictly reduced to the boundary layer with its appropriate treatment [1]. In the case of a subsonic plasma facility, like the VKI Plasmatron, the testing configuration could be presented as in Fig. 9.8.

The experimental assessment of the LHTS methodology has been conducted at VKI by Chazot et al. [4] and Barbante and Chazot [1]. The results are presented on the Figs. 9.9 and 9.10. It could be seen that the boundary layers profiles from the hypersonic flow is very well duplicated with the subsonic plasma flow, when matching the characteristics parameters at the edge of the boundary layer.

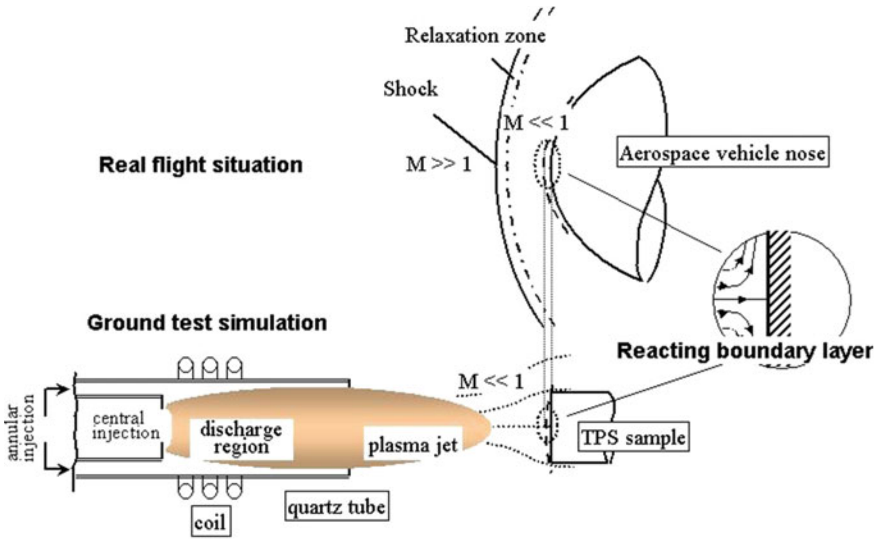

Fig. 9.8 TPS testing in plasma wind tunnel in LHTS conditions

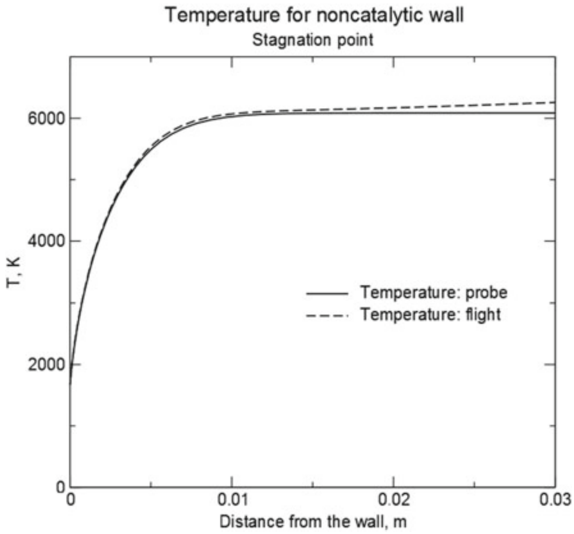

Fig. 9.9 Temperature profiles comparison between flight and ground testing conditions [1]

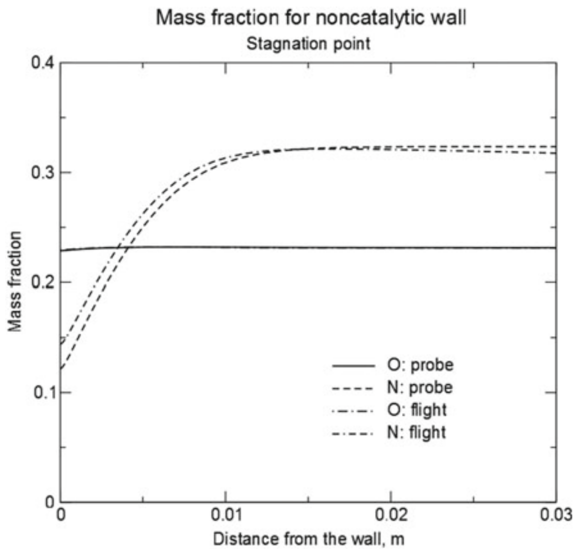

Fig. 9.10 Mass fraction profiles comparison between flight and ground testing conditions [1]

9.5 Considering Flow Radiation

Shock layer radiative heating appears around 9 km/s in Earth's atmosphere and 7 km/s in Mars' atmosphere [17]. It reaches 10 % of the total heat flux for probes having a diameter smaller than 1 m and entry velocities approaching 13 km/s in the Earth's atmosphere [21]. It is then a physical phenomena to be considered for super-orbital re-entry.

However on ground testing for scaled models flow radiation is not often addressed in the literature. It is usually mentioned to explain that this feature of the flow cannot be properly reproduced considering similarity rules for a model of reduced dimension [18]. But even if this conclusion is fully valid some more details could be added to better understand the issues with radiative heat-transfer in hypersonic facilities.

The radiative flux at a certain point P is the integral of the radiative intensity in all directions and over the entire frequency spectrum. Considering a point S on a surface, the radiative flux reaching S from a point P of the surrounding is the difference between the energy emitted and that absorbed integrated along the optical path from P to S. In the case of a scaled model in a high-enthalpy facility, respecting the same condition for the gas and the flow velocity, it could be shown that the optical thickness in the radiating shock layer scales with the product ρL, considering the absorption coefficient remaining the same on the two situations.

One is then brought back to the same approach as the binary scaling exposed before. In these conditions, the radiative heat-flux on the surface of a scaled model could be reproduced in a ground-based facility.

However, if one considers the flow passing through the radiating environment around the model, it appears that the scaling does not hold any longer. The amount of energy E radiated by a control volume is proportional to the mass contained in that volume: $E \propto m = \rho.L^3$. The amount of flow $\dot{m}$ ingested in the shock layer could be expressed as $\dot{m} \propto \rho.U_\infty.L^2$. Therefore, when the flow-radiation coupling need to be taken into account the heat radiated per unit mass passing in a control volume scales as: $E/\dot{m} \propto L$. In conclusion, even if the radiative flux on the surface of a scaled model could be reproduced, the radiative heating of the flow around the same model is not respected.

This problem arises if the radiative heating is important enough to have an influence on the rest of the flowfield. This coupling is quantified by the radiative cooling parameter Γ, also referred to as Goulard number, defined as

$$\Gamma = \frac{2 \cdot Q^r_{ad}}{1/2 \cdot \rho_\infty \cdot U^3_\infty} \tag{9.6}$$

where Q^r_{ad} is the radiative heating for an adiabatic flow, that is without radiative cooling, ρ_∞ the free-stream density, and v the shock velocity. This parameter serves as an approximate measure of the coupling between radiation and the flow [10]. It is the ratio of the amount of radiation generated by the shock, assumed to be twice the radiative heating of the surface, by the kinetic energy heat flux entering the shock layer. If $\Gamma > 0.01$, radiation is coupled to the rest of the flowfields and the effect of improper radiation scaling extends to the rest of the flowfield.

Furthermore, there is a coupling between radiation and gas–surface interaction processes. Indeed, the use of ablative material is compulsory for high-speed re-entry. The ablation processes are very efficient to prevent the hot shock layer gas from reaching the wall and absorb part of the shock layer radiative heat flux. An accurate estimation of the absorbed radiation is complex since the thickness and thermochemical state of the ablation gas layer are difficult to predict up to now [17].

Other surface phenomena such as catalycity and oxidation should be also taken into account, it makes the problem even more intricate when one is aware that such a models are not yet fully established, especially in high-enthalpy flows [3].

9.6 Ground Testing Strategy for High-Speed Re-entry

Ground testing facilities commonly used to study re-entry flows and that are mainly concerned with high-enthalpy flows can be divided in two categories:

- Impulse facilities, such as shock tubes and ballistic ranges, are only able to produce flows that last typically a fraction of a second. It is usually assumed long enough to let the steady flow establish itself, but too short compared to the thermal inertia of the material surface. They are thus mainly used to investigate the aerothermodynamic effects, gas kinetic, and radiation processes. In this category, expansion tubes are able to reach free-stream enthalpies characteristic of high-speed re-entry.
- Plasma wind tunnels, such as inductively coupled plasma facilities or arc-jets, are able to operate for long test durations, in the order of minutes. However, they have not been designed to reproduce the flow radiation.

Similar flights conditions or direct flight duplication are possible to reproduce for sub-orbital re-entry velocity in those facilities for a limited time (impulse facilities) or in a limited region (stagnation point in plasma wind tunnels). Each category of facility is addressing a specific aspect of the flowfield as it is sketched in Fig. 9.11.

The velocity of the flow is much higher in the case of high-speed re-entry which concern super-orbital conditions. This results in considerably higher free-stream

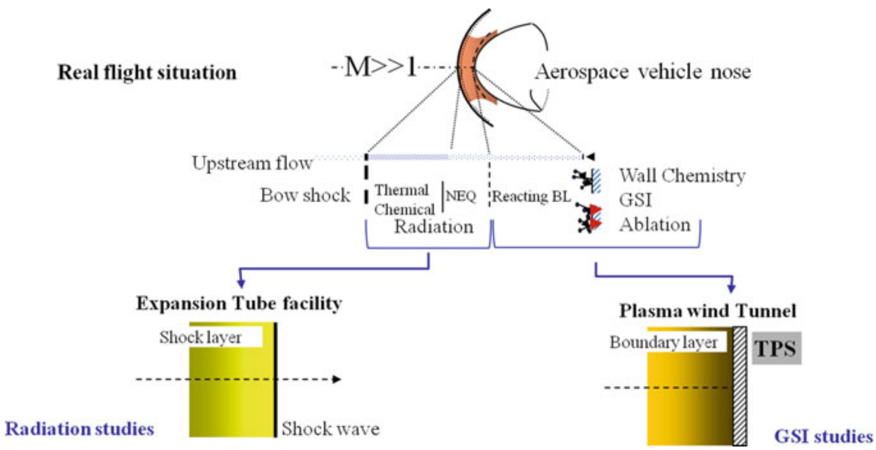

Fig. 9.11 Different types of facilities for flow radiation or gas–surface interaction studies

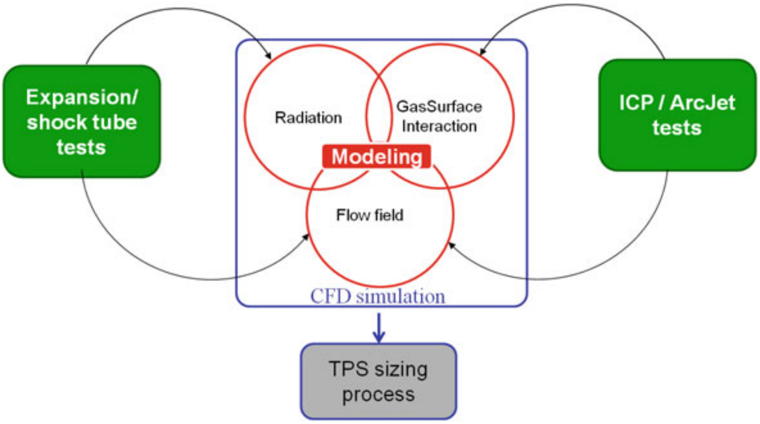

Fig. 9.12 Testing methodology for high-speed re-entry TPS sizing

enthalpy and pressure in the flow, and leads to coupling phenomena that cannot be reproduced on scaled down models.

High-speed re-entry requires therefore a new approach of ground testing. Ground testing facilities should be used to investigate separately different phenomenon playing a role in the aerothermodynamic of the flow, rather than to duplicate flight, as it is the case for lower velocity re-entry.

Those investigations should be performed on a panel of different facilities in order to develop models and databases. Models of shock layer radiation can be developed based on measurements performed in impulse facilities, under conditions similar to those encountered in high-speed re-entry. However, material processes such as ablation and radiative heating cannot be performed in the same facility due to the short test duration involved. In particular, gas–surface interaction have to be studied in plasma wind tunnels. As it is known, those facilities could produce the required heat flux level, but have not been designed to take into account the correct coupling phenomena including the radiation processes present below a shock layer. Models of gas–surface interaction have therefore to be developed under different conditions than that of high-speed re-entry.

Those models, developed separately, should then be implemented in Computational Fluid Dynamics (CFD) codes and allow extrapolatation to the actual flight conditions. Since they cannot be validated within the flight conditions envelope unless flight-testing is performed, they need to capture the main physical phenomena and their accuracy is of prime importance. This, in turn, allows sizing and designing re-entry probes as well as assessing their performance in flight, with computational methods rather than direct ground testing facilities. The general framework of the testing methodology for high-speed re-entry is summarized in Fig. 9.12.

9.7 Conclusion

The reproduction of high-speed re-entry conditions in ground-based facilities is a real challenge. Because of the coupling phenomena that characterized this type of flows their accurate experimental simulation on scaled-down models is impossible in ground-based facilities. It appears as well that only little research has been conducted on dedicated strategies for ground testing of high-speed re-entry flows. Most of the time, testing is limited to qualification tests that reproduce the heat flux level without taking into account all the physical phenomena involved.

The two main limitations concern radiation and gas–surface interactions: both cannot be correctly reproduced at the same time in a single facility. Indeed, the time-scale achieved in impulse facilities is shorter than those relevant for gas–surface interactions, while the correct radiation phenomena are difficult to reproduce in plasma wind tunnels.

A solution is to develop models for radiation and gas–surface interaction in the relevant facilities, under controlled environments. Since the conditions in which those models can be validated in ground facilities are different from the one encountered in high-speed flows, they specifically require to be physic based in view of their extrapolation to flight conditions. In such a context, model validation and their incorporation in CFD codes are crucial for the development of aerospace applications. UQ methodologies are expected to play a major role for this approach as they represent the unique way to give solid basis to the validation process. In order to manifest all their benefit and correctly address the problem, UQ methods imperatively need to be articulated with the experimental procedure. To this end, this brief review exposes the rationale of experimental testing and how it is linked to the physics of aerospace flights. It would serve as a basis for the development of Uncertainty Quantification apply to the validation of high-speed flow modeling.

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