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# Flap effectiveness appraisal for winged re-entry vehicles

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ABSTRACT

The interactions between shock waves and boundary layer are commonplace in hypersonic aerodynamics. They represent a very challenging design issue for hypersonic vehicle. A typical example of shock wave boundary layer interaction is the flowfield past aerodynamic surfaces during control. As a consequence, such flow interaction phenomena influence both vehicle aerodynamics and aerothermodynamics. In this framework, the present research effort describes the numerical activity performed to simulate the flowfield past a deflected flap in hypersonic flowfield conditions for a winged re-entry vehicle.


## 1. Introduction

In the last years Computational Fluid Dynamics (CFD) has played an important role in hypersonics, not only for research activities, but also to address several practical design issues [1]. The present paper focused attention on an important CFD issue for the design of aerospace vehicles, namely shock wave-boundary layer interaction (SWBLI) [2–4]. This particular flowfield feature, with the related recirculation zone, overheating and pressure overshoot, occurs whenever an aerodynamic control surface is deflected at very high speed conditions. It is also typical in the vicinity of fuselage–wing junctures, corner flows and in inlets and many other critical locations of the vehicle surface [2–7].

In this research effort we report and discuss the results of the computational analysis of the flowfield past a wing in flap-on configuration in hypersonic flow conditions. The study of such configuration presents some peculiar aspects. When a flap is deflected at hypersonic speed, overshoots of heat flux and pressure take place just after a recirculation bubble, located at the hinge line region, thus worsening the flap effectiveness as well [5]. Stabilization, trim, and control devices effectiveness, prerequisites for flyability and controllability of aerospace flight vehicles [6], can be then compromised. This is especially true considering that aerothermoelasticity of the airframe and, in particular, of stabilization, trim and control surfaces with large mechanical and thermal loads can be a special problem.

In addition, shock interactions can cause boundary layer separation with concomitant high heat transfer at reattachment which has a significant impact on the design of vehicle thermal protection systems [7]. Therefore, accurate prediction of shock interactions is essential for optimal design of hypersonic vehicles.

### 1.1. Shock interaction phenomena

Practical examples of Shock interference situations in the flowfield past space vehicles are given in figure below. In the case (a) of Fig. 1 the fuselage bow shock meets a fin bow shock; the second example (b) is relative to interference between the bow shocks of the launcher and booster [7]. A similar situation is encountered when the vehicle bow shock meets the shock forming ahead of the

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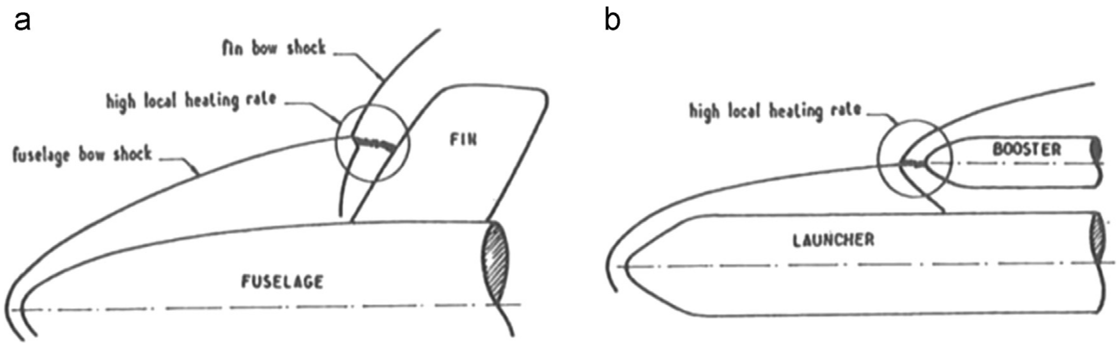


**Fig. 1.** Example of shock interference heating.

canopy. Such shock interactions result in more or less complex shock patterns including shear-layers or jets which can impinge on the vehicle causing high local heating rates, well in excess of those occurring at a nose stagnation point.

On the other hand, SWBLI occurs at impingement of a bow shock on the vehicle leading edges (wing and tail). Other typical occurrences are when a bow shock interacts with the shock ahead a deflected flap, or along axial corners in wing–body and fin–wing junctions, as shown in Fig. 2 [7].

SWBLI also occurs in air-intakes of air-breathing propulsive systems. Such interactions can induce separation of the boundary-layer which causes loss in control effectiveness or flow degradation in an engine inlet. Also the subsequent reattachment of the separated shear-layer gives rise to heat transfer rates that can far exceed the one of an attached boundary-layer.

An example of SWBLI interactions that take place at the boosters fairing of an expendable launcher is reported in Fig. 3 [8,9]. In this figure the pressure contours field on the surface and the symmetry plane of an expendable launch vehicle is reported for $M_\infty = 2.5$ and $\alpha = 5^\circ$.

Flow separation bubble at launcher boat-tail and effect of fuselage–booster shock–shock interaction are provided in Fig. 4. In this region pressure overshoot (and then overheating) must be carefully assessed for a reliable launcher design. Aerodynamic performances of launcher are influenced as well [2,8,9].

An example of shock impingement on wing leading edge of a re-entry vehicle can be found in Fig. 5 [2]. Here the pressure contours (left side) on the vehicle aeroshape and the Mach number field in the wing plane (i.e. the trace of bow shock on the wing plane) are provided for $M_\infty = 6$ (left side) and $M_\infty = 7$ at $\alpha = 5^\circ$. As shown, the interaction between wing shock and vehicle bow shock results in an overshoot of pressure (and then heat flux) localized at the wing leading edge. In particular, the point of wing leading edge where this interaction impinges depends on the freestream conditions and vehicle attitude.

Above examples point out that the assessment of SWBLI phenomenon demands for accurate flowfield investigations between numerical and experimental test campaigns, as discussed in the next chapter.

## 2. Flowfield past a double wedge

In this framework, the theoretical model was assessed and validated by performing a numerical simulation of a well-known test bench, namely the double wedge, and comparing numerical results with experimental data [5]. Successively, the CFD tool was used to compute the flowfield past a gliding high speed vehicle and results concerning the SWBLI phenomena are discussed. Some examples of simulations performed by the authors in previous projects are also shown [10,11].

The principal flowfield phenomena that take place in the flow field structure past a double wedge are summarized in Fig. 6.

The forward wedge ($\alpha = 30^\circ$) generates an oblique shock wave with inviscid shock angle 39.8°. The rearward wedge angle ($\alpha + \theta_w = 55^\circ$) exceeds the maximum inviscid flow deflection angle (43.3°) and hence generates a detached shock wave. The shock wave boundary layer interaction causes separation of the boundary layer on the forward wedge leading to a recirculation region and separation shock wave. The forward wedge oblique shock and separation shock interact to form a triple point above the separation region. A second triple point is formed by the intersection of the separation shock and rearward wedge detached shock resulting in a strong shear layer with subsonic flow above and a shock-expansion train beneath. The reattachment of the boundary layer results in a peak in surface heat transfer. Accurate prediction of shock interactions is therefore essential for optimal design of hypersonic vehicles.

With this in mind a numerical rebuilding of the experiment test campaign carried out with a double wedge test bed is exploited. Indeed, the flowfield that takes place past a double wedge is of practical interest. This configuration is a benchmark as it presents unique flow patterns typical of two-dimensional inlets and deflected control surfaces for re-entry vehicles.

The investigation of this test bench, by means of CFD, allows assessing the flow separation phenomenon that can significantly affect the flap efficiency of a hypersonic vehicle.