I. INTRODUCTION

The progress in the design and calculation of turbomachinery blades in transonic and supersonic flow is intimately linked to a detailed knowledge of the phenomenon of shock boundary layer interaction which requires a priori of course the determination of the position and strength of the impinging shock and the state of the boundary layer on the surface on which the interaction takes place. Our understanding of the problem is still very incomplete, but it would be probably non-existent if the turbomachinists would not have at their disposal a huge amount of theoretical and experimental work accumulated in the past 1 or 2 decades in the fields of transonic and supersonic flow over isolated wings, the flow in propulsion nozzles, ejectors, turbo-jet inlets, etc. However, although the problems in the turbomachinery field are very similar to those encountered in the above mentioned areas, one must not fall into the trap of applying their solutions indiscriminately to our problems.

The flow problem with which the compressor and turbine designer is faced is illustrated in Fig. 1 which shows a compressor cascade [1] and a turbine cascade with their typical shock configurations at design point. The calculation process involves in both cases 2 steps: (1) determination of the shock configuration (leading edge shock configuration in the case of the compressor, trailing edge shock configuration for the turbine cascade), and (2) determination of the effects of the shock-boundary layer interaction.

In the case of the compressor cascade, the first step involves the computation of the inlet flow field and the suction surface Mach number before the impingement of the lower branch of the L.E. shock (passage shock) which is usually considered to be a normal shock. The second step is concerned with the evaluation of local or complete flow separation from the blade surface, the transition of the normal shock near the suction surface into a

[1] from Ref. 1

λ- shock and the drastical change of the effective flow channel downstream of the passage shock in the case of heavy flow separation from the suction surface.

In the case of the turbine cascade, the first step consists in finding a solution to the problem of the confluence of 2 supersonic flows at the trailing edge of a blade. If the initial shock position and shock strength is known, the complete shock configuration can be determined by the method of characteristics. The second step consists, as in the case of the compressor cascade, in the study of the variations of the boundary layer properties due to the shock impingement of the left trailing edge shock on the blade suction surface.