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Smooth leading edge transition in hypersonic flow

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Abstract The boundary layer transition along the attachment line of a smooth swept circular cylinder in hypersonic flow is investigated in a blowdown wind tunnel. A wide range of spanwise Mach numbers Me (3.28 to 6.78) is covered with the help of different models at several sweep angles ($60^\circ \leqslant \Lambda \leqslant 80^\circ$). The transition is indirectly detected by means of heat flux measurements. The influence of the wall to stagnation temperature ratio is investigated by cooling the model with liquid nitrogen.

List of symbols

D	cylinder diameter
M	Mach number
P	static pressure
Q_∞	free-stream flow velocity
r	recovery factor
$\bar{R}$	$=(V_e\eta)/v_e$ Reynolds number
$\bar{R}^*$	$=V_e\eta^*/v^*$ Reynolds number
R_θ	$=(V_e\theta)/v_e$ Reynolds number
$R_{D\infty}$	$=(Q_\infty D)/v_\infty$ Reynolds number
s	location of transition along the leading edge
St	$=\phi_w/(\rho_{e_0}V_{e_0}C_p(T_r-T_w))$ Stanton number
T	absolute temperature
T_r	$=T_e(1+r(\gamma-1)/2M_{e_0}^2)$ recovery temperature
T^*	$=T_{e_0}+0.1(T_w-T_{e_0})+0.6(T_r-T_{e_0})$ Poll's reference temperature
x, y, z	coordinate system (respectively chordwise, spanwise and normal to the cylinder surface)
U, V, W	velocity components outside the boundary layer
Y^+	$=y/D$ non-dimensional coordinate
η	boundary layer length scale $=(v_{e_0}/(\mathrm{d}U_e/\mathrm{d}x))^{1/2}$
ρ	density
v	kinematic viscosity
Λ	sweep angle
θ	spanwise momentum thickness
ϕ_w	heat flux

Superscripts

$*$	parameter computed at the reference temperature T^*

Subscripts

c	critical conditions
e_0	boundary layer edge conditions along the attachment line
max	last thermocouple location
t	stagnation conditions
tr	transition conditions
w	wall conditions
∞	free-stream conditions
1	onset of transition
2	end of transition

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1 Introduction

The first experimental studies related to attachment line transition on swept wings were performed in the 1950s, and concerned essentially low speeds. The purpose was to investigate the possibility of having a laminar flow over the whole aircraft wing. It was emphasized that the boundary layer state over the wing is driven directly by the laminar or turbulent character of the attachment line boundary layer, and depends on the leading-edge sweep and radius (Gray 1952).

An important mechanism giving a turbulent flow along the whole attachment line was identified very early (Pfenninger 1965) and named leading edge contamination: when the boundary layer which spreads over the attachment line is initially turbulent at the wing-fuselage junction it can remain turbulent along the attachment line if the Reynolds number is sufficiently high. A similar situation can be achieved by introduction of a source of large perturbation as, for instance, an isolated roughness element or a trip wire across the attachment line. These types of transition are usually considered as being dominated by essentially non linear phenomena.

When the upstream end of the wing is not fixed to a wall, a boundary layer develops from the tip. If the leading edge region is modelized as a swept cylinder and provided that the ratio of the spanwise length to the cylinder diameter is sufficiently large, the flow will achieve an asymptotic state independent of the spanwise coordinate. Even if the cylinder surface is perfectly smooth, the constant thickness boundary layer along the attachment line can also become turbulent above a critical Reynolds number. It is generally believed that, in such a situation, the transition occurs after amplification of initially very small perturbations which ultimately break into turbulence. The present paper is devoted essentially to this second type of transition which is usually called the smooth regime.

Another type of transition can occur in the vicinity of the attachment line. It is characterized by longitudinal streaks and named cross-flow instability. In principle cross-flow instability occurs only away of the symmetry plane containing the attachment line. However as shown by Poll (1985) it is not always easy to distinguish between leading edge contamination and transition by cross flow instability.

Numerous works were devoted to the study of attachment line transition in incompressible flow. By means of hot wire measurements Pfenninger and Bacon (1969) have shown that, in the smooth regime, leading edge transition occurs after an initial phase corresponding to amplification of small perturbations existing in the upstream flow. Amplification begins above a critical Reynolds number $\bar{R}_c$ and transition, characterized by the appearance of turbulent spots and a significant intermittency, occurs above a higher Reynolds number $\bar{R}_{tr}$. The disturbances, which were initially two-dimensional with wave fronts orthogonal to the attachment line, become later three-dimensional at transition. These results were confirmed by Poll (1979) who suggested that, far from the tip of the model ($s/\eta \simeq 8000$) transition begins at $\bar{R}_{tr} > 650$ and that a fully turbulent flow is achieved for $\bar{R} > 740$. Moreover the wall can be considered as smooth provided that the size k of a roughness element is such that $k/\eta < 0.8$ were η is a length scale characterizing the boundary layer thickness on the attachment line.

Studies of the flow over swept wings or swept cylinders in supersonic flow were performed initially in order to determine the influence of the sweep angle on the heat transfer rate. In 1959 Beckwith and Gallagher detected the transition during the study of the flow over a swept cylinder at $M_\infty = 4.15$ in a large range of sweep angles ($0 < \Lambda < 60^\circ$). Later Allen et al. (1965) conducted tests on a 80° swept delta wing at several angles of attack α, in a blowdown hypersonic tunnel at $M_\infty = 6.8$ ($M_e = 5.9$), with a cylindrical leading edge and a hemispherical nose of same radius. Later Bushnell (1967) and Bushnell and Huffmann (1967) after examination of existing heat flux measurements in the $2.5 < M_\infty < 8$ and $\Lambda > 40^\circ$ range, suggested that the boundary layer remains laminar until $R_{D\infty,tr} \simeq 8 \times 10^5$.

In 1978, Poll introduces the reference temperature concept in order to obtain an empirical formulation for the heat transfer coefficient in the case of turbulent flow along an attachment line. A formula giving the reference temperature T^* as a function of the wall temperature T_w, the temperature at the edge of the boundary layer T_{e_0} and the recovery temperature T_r was proposed by Poll (1981) and the corresponding semi-empirical formula was correlating fairly well a set of collected heat transfer data from the NACA/NASA teams. Moreover it was demonstrated that, provided one use a Reynolds number $\bar{R}^*$ based on this reference temperature, the onset of transition in the presence of a source of large perturbation (contamination or rough regime) was well described by a unique critical value $\bar{R}^* \simeq 245$. This value surprisingly coincides with low speed results (Poll 1979). Assuming that the reference temperature concept is also valid for the smooth regime Poll (1994) suggested the value $\bar{R}^*_{tr} \simeq 650$ for natural transition. However this conjecture is based on a rather small set of experimental data (Bushnell and Huffmann 1967; Creel et al. 1987).

A first attempt to check the validity of the $\bar{R}^*$ criterion for natural transition was performed by Yeoh (1980) at Cranfield using a swept cylinder equipped with a hot film located at $Y^+ = 2.13$ on the attachment line. For $M_e = 1.78$ he obtained $\bar{R}^*_{tr} \simeq 400$. However the length of the model was perhaps insufficient to reach an asymptotic flow situation independent of the spanwise coordinate. Moreover the upstream end of the model was close to the nozzle wall and there is a possibility of contamination of the attachment line flow by the turbulent boundary layer on the nozzle wall.

Some experimental investigations were performed at NASA Langley by Creel et al. (1986), (1987) and Creel (1991) for $\Lambda = 45^\circ$, 60° and 76° at $M_\infty = 3.5$ ($M_e = 1.66$, 2.39 and 3.17 respectively) with $T_w/T_r = 1$. The transition was detected by means of recovery temperature measurements along the attachment line. They found a critical value of the Reynolds number nearly independent of M_e namely $R_{D\infty,tr} \simeq 6.5$–$8 \times 10^5$ ($\bar{R}^*_{tr} \simeq 610$–715).

Skuratov and Fedorov (1991) carried out experiments for the following test conditions: $M_\infty = 6$, $\Lambda = 40^\circ$ and 60° ($M_e = 1.87$ and 3.1) in order to check the validity of Creel's results. The model was a swept circular cylinder with a hemispherical nose and a spanwise length $Y^+_{\max} = 10.2$; an asymptotic state was apparently reached for $Y^+ \geqslant 5$. The transition was detected by means of a thermosensitive coating. They found a critical value of the Reynolds number which seems to be a slightly increasing function of M_e: $R_{D\infty,tr} = 7 \times 10^5$ for $M_e = 1.87$ and $R_{D\infty,tr} = 9.2 \times 10^5$ for $M_e = 3.1$. One must notice that the hemispherical nose generates a strong entropy gradient which may perhaps affect the results.

Holden and Kolly (1995) conducted tests in a shock tunnel for $10.4 \leqslant M_\infty \leqslant 11.4$. They used a cylinder of length $Y^+_{\max} = 11$ equipped with thin films allowing to detect the transition from fluctuation measurements. Sweep angles in the range 60–80° were used. The interchangeable upstream ends of the model are cut-off nearly parallel to the free stream flow according to the sweep angle being used. For a smooth configuration they obtained $R_{D\infty,tr1} \simeq 8 \times 10^5$ ($R^*_{tr1} \simeq 550$) for $M_e = 4.54$ ($M_\infty = 10.5$ and $\Lambda = 66.5^\circ$).

Murakami et al. (1996) performed experiments in a shock tunnel using a liquid-crystals visualization technique in order to detect the transition for $T_w/T_r = 0.6 - 0.9$. The purpose of their experiments was to determine the critical value of Poll's parameter $\bar{R}^*$ in the smooth regime. The model had a sharp upstream end parallel to the free stream direction with a length $Y^+_{\max} = 7.5$. For $\Lambda = 45^\circ$ and 60° with $M_\infty = 5$ ($M_e = 1.89$ and 2.89), the boundary layer is laminar up to $R_{D\infty,tr} = 12 \times 10^5$ and 9×10^5 (respectively $\bar{R}^*_{tr} \simeq 730$ and 700). These values are higher than those of Creel et al. at $M_\infty = 3.5$, and, as the value of T_w/T_t is smaller, they could reveal a stabilizing effect of cooling on the transition onset. Another possible explanation of this discrepancy is the fact that, as in the Skuratov case, the value of $Y^+_{\max}$ is perhaps not sufficient to reach the asymptotic state: if

Table 1. Available results for supersonic flow over smooth models

Reference	M_∞	Λ°	$Y^+_{\max}$	T_w/T_t	M_e	$R_{D\infty,tr1} \times 10^{-5}$	$\bar{R}^\star_{tr1}$	$R_{\theta,tr1}$
Allen (1965)	6.8	80	6	$\simeq 0.5$	5.92	>4.2	>305	>444
Yeoh (1980)	2.4	58.5	2.13	$\simeq 0.9$	1.78	?	400	202
Creel (1986–87)	3.5	45	10	$\simeq 0.9$	1.66	7	625	310
	3.5	60	10	$\simeq 0.9$	2.39	8	715	435
	3.5	60	20	$\simeq 0.9$	2.39	7	670	405
Creel (1991)	3.5	76	7	$\simeq 0.9$	3.17	6.55	610	448
Skuratov (1991)	6	45	10.2	$\simeq 0.8$	1.87	7	515	272
	6	60	10.2	$\simeq 0.8$	3.1	9.2	650	454
Holden (1995)	10.6	60	6.8	0.28	3.57	5.7–7.75	406–471	310–360
	10.5	66.5	6.8	0.26	4.54	7.4–9.7	464–542	434–507
	10.6	75	6.8	0.26	6.47	<7.3	<383	<533
	10.6	80	6.8	0.26	8.06	<10.15	<374	<714
Murakami (1995)	5	45	7.5	$\simeq 0.75$	1.89	12	730	392
	5	60	7.5	$\simeq 0.75$	2.89	9	695	484
Coleman (1996)	1.6	76	7.3	$\simeq 0.9$	1.53		>670	>315

this is the case, a longer cylinder would have given slightly lower values of $R_{D\infty,tr}$. Table 1 gives a synthetic presentation of the previously mentioned results. The symbol $>$ (or $<$) denotes that the transition was not detected for the maximum (minimum) reached Reynolds number value, and that the critical value of this criterion is located above (below) this limit.

A first attempt to study theoretically the stability of the supersonic flow along an attachment line over a smooth wall was performed by Malik and Beckwith (1988). They used the linear stability theory with perturbation in the form of Tollmien–Schlichting (TS) waves. For one of the Creel's experiments with $M_\infty = 3.5$ and $\Lambda = 60^\circ$ they found a critical Reynolds number $\bar{R}_c = 640$ corresponding to $\bar{R}^\star_c = 391$. The most amplified TS waves make an angle about 55° with the attachment line and the theory predicts a strong stabilizing influence of wall cooling. More exactly Malik and Beckwith showed that the critical Reynolds number $\bar{R}_c$ increases from 640 up to 1280 when the wall to recovery temperature ratio decreases from $T_w/T_r = 1$ down to 0.8. These results were later confirmed numerically by Arnal et al. (1990).

However this kind of approach, based on TS waves, neglects the chordwise dependence of the basic flow. Several stability studies performed in the case of an incompressible flow have demonstrated that the Görtler–Hammerlin (GH) form of perturbations, which involves a linear dependence over x, is more appropriate for the study of attachment line instability. On the basis of the results obtained for incompressible flow one may expect that using a TS type perturbation instead of the GH type brings an overestimation of the critical Reynolds number by at least 15% to 20%. More recently Lin and Malik (1995) have studied the temporal three-dimensional stability problem for the same Creel's experiment: they found a much smaller value for the critical Reynolds number $\bar{R}_c = 350$. Although this result concerns the temporal instability, which cannot be directly compared to the spatial instability studied by Malik and Beckwith or Arnal, it perhaps suggests that the most unstable modes may differ significantly from TS waves.

As usual the linear stability studies predict a critical Reynolds number for instability which is significantly lower than the transition Reynolds number. However, because the flow along an infinite swept cylinder is independent of the spanwise coordinate, one could expect that, as soon as the flow is unstable the critical perturbation undergoes amplification and eventually degenerates into transition. Therefore if, in the limit of an infinitely long cylinder ($s/\eta \to \infty$), a difference exists between the critical Reynolds number values for instability and transition, it would mean that the flow along the attachment line can remain forever in an intermediate state between a laminar stable and a fully turbulent flow.

This survey of previous experiments reveals that the number of experimental results, specially for high values of the spanwise Mach number, is very small and that very little is known about the influence of the spanwise Mach number or the ratio of the wall to stagnation temperature. The present paper gives some recent results obtained in a small low enthalpy hypersonic wind tunnel with an attempt to cover a fairly large range of Mach numbers and to investigate the influence of the wall to stagnation temperature ratio.

2
Experimental set-up

The experiments were carried out in a low enthalpy hypersonic blowdown wind tunnel which gives free-stream Mach numbers of 7.14 ($\pm 0.5\%$) and 8.15 ($\pm 0.7\%$). Several swept circular cylinders, with different diameters and sweep angles, were used in order to investigate a wide range of values for M_e and $\bar{R}^\star$. The upstream end of the models is sealed and cut off parallel to the free stream direction. The wall roughness was checked carefully and the height of the defects was never larger than 10^{-3} mm. Transition is detected by means of Stanton number measurements along the attachment line. The heat flux measurements are performed with the thin-wall technique: all the models are equipped with thermocouples, spot-welded on the inner surface of the thin skin. As shown in Fig. 1, the model is injected into the test section by means of a pneumatic jack and the heat flux is deduced from the time evolution of the wall temperature. By cooling the model wall with liquid nitrogen before injection into the flow it was possible to test the influence of the wall to stagnation temperature ratio and to reach a value $T_w/T_t \simeq 0.23$ corresponding to a wall temperature about 180 K.

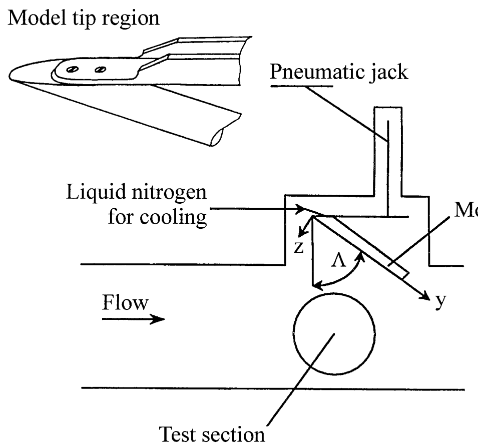

Fig. 1. Experimental setup and sketch of the model tip

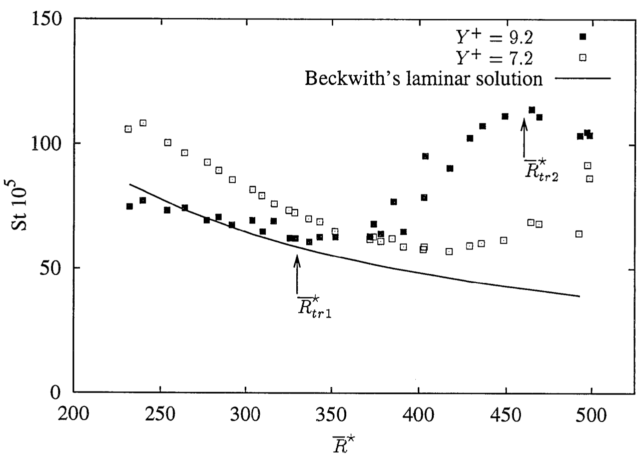

Fig. 2. Typical records of the Stanton number versus $\bar{R}^*$ ($M_\infty = 7.14$, $\Lambda = 79.5°$, $Me = 6.07$, $D = 33$ mm)

By performing several runs with different wind tunnel stagnation conditions it was possible to obtain, for each thermocouple location along the attachment line, the distribution of the Stanton number with respect to the Reynolds number. The chordwise velocity gradient, needed for the computation of $\bar{R}^*$ was deduced from the modified Newtonian theory. The validity of the Newtonian theory for low values of the normal Mach number is of course questionable. This problem has been investigated by Poll (1993) who has shown that, provided that the normal Mach number is larger than 1.5, the error on the velocity gradient is less than 5% (2.5% on $\bar{R}$). The Fig. 2 shows, for two thermocouples, some typical Stanton number distributions versus $\bar{R}^*$. As shown in Fig. 2, for each thermocouple location along the attachment line two transition Reynolds numbers can be defined: the value $\bar{R}^*_{tr1}$ corresponding to the beginning of transition is characterized by a minimum of the Stanton number with respect to $\bar{R}^*$ (below this value the flow is laminar). The value $\bar{R}^*_{tr2}$ corresponding to the end of transition is characterized by a local maximum of the Stanton number with respect to $\bar{R}^*$ (above this value the flow is turbulent).

Provided that the length of the cylinder is sufficient the flow should achieve an asymptotic state independent of the spanwise coordinate Y^+ and corresponding to the so-called infinite swept cylinder situation. Assuming that this asymptotic situation is reached, the exact laminar solution of Beckwith (1958) for the Stanton number value is also plotted in Fig. 2 for comparison purpose. Achievement of the asymptotic state, at least for the inviscid flow, can be judged by looking at the schlieren visualizations in order to see if the stand-off distance of the shock in front of the cylinder reach a constant value. A second test, more appropriate for the boundary layer, consists in a comparison of the Stanton number measurements with the theoretical value: in the asymptotic region the product $St \times \bar{R}$ should reach a constant value and a boundary layer which is still growing along the attachment line is characterized by higher values of the Stanton number. As shown in Fig. 2 a reduced length $Y^+ = 7.2$ was insufficient in the present case while the thermocouple located at $Y^+ = 9.2$ was in good agreement with the theoretical value in the laminar range.

3 Results and discussion

The results of about 350 runs are summarized in Table 2. Each model is identified by a four digit number: the first two digits give the cylinder diameter in millimeters and the last two digits the sweep angle.

3.1 Ambient wall temperature

The first 20 configurations of Table 2 correspond to runs with a model wall at ambient temperature giving a typical value about 0.4 for the wall to stagnation temperature ratio. Figure 3

Table 2. Present experimental results

Model	M_∞	Λ	Y^+_{max}	T_w/T_t	M_e	$R_{D\infty,tr1} \times 10^{-5}$	$\bar{R}^*_{tr1}$	$R_{\theta,tr1}$
C1480	8.15	80	21.4	0.39	6.78	2.74	220	352
C1980	8.15	80	15.2	0.39	6.78	3.53	250	400
C3381	7.14	81	8.9	0.39	6.31	3.9	290	425
C3379a	7.14	79.5	9.1	0.39	6.07	4.6	330	391
C3379b	7.14	79.5	9.3	0.42	6.07	5.7	350	506
C3379c	7.14	79.5	8.6	0.38	6.07	3.7	300	419
C1978	7.14	78	15	0.44	5.8	3.63	300	391
C3376a	7.14	76.5	9.3	0.38	5.57	5.6	405	492
C3376b	7.14	76.5	8.6	0.38	5.57	4.3	355	434
C3375	7.14	75	9.1	0.45	5.32	6.8	450	570
C3374a	7.14	74	9.1	0.46	5.15	8.55	512	581
C3374b	7.14	74	8.6	0.38	5.15	5.22	420	447
C3373a	7.14	73	9.1	0.39	4.99	6.8	490	545
C3373b	7.14	73	9.3	0.44	4.99	8.5	530	576
C3372	7.14	72	9.1	0.45	4.83	>11	>610	>630
C3370	7.14	70	9.1	0.46	4.53	10.4	615	603
C3369	7.14	68.5	9.1	0.45	4.32	>11.6	>670	>613
C3368	7.14	67.5	9.1	0.45	4.18	>12	>685	>595
C3365	7.14	65	9.1	0.48	3.85	>13.8	>735	>596
C2060	7.14	60	9.3	0.38	3.28	>4.9	>481	>339
C3379C	7.14	79.5	8.6	0.22	6.07	<2.48	<250	<360
C3376C	7.14	76.5	8.3	0.24	5.57	3.15	310	400
C3374C	7.14	74	8.6	0.24	5.15	5.75	450	517

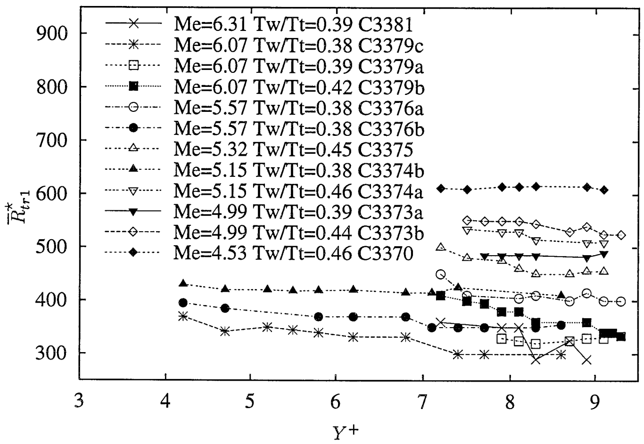

Fig. 3. Influence of the spanwise location on the beginning of transition ($D=33$ mm)

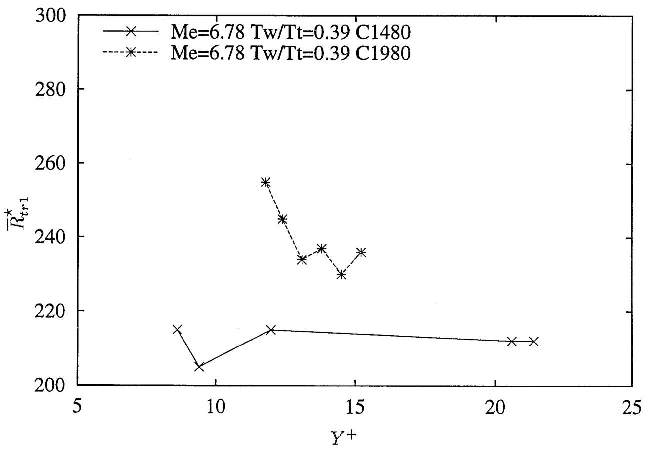

Fig. 4. Influence of the spanwise location on the beginning of transition ($D=14$ and 19 mm)

shows the evolution of the transition Reynolds number, with respect to the reduced coordinate Y^+, for the tests performed with a cylinder of diameter 33 mm. The value $\bar{R}^*_{tr1}$ corresponding to the beginning of transition is a slightly decreasing function of Y^+ which seems to reach a constant value for $Y^+ > 8.5$. This is confirmed by Fig. 4 which shows the results for the cylinders of diameter 14 and 19 mm. However one can notice that a perhaps slightly higher value of Y^+ is needed for the highest sweep angles ($\Lambda \simeq 80°$).

One some models the thermocouples have been spot-welded twice, the first time over the upstream part of the cylinder and then farther downstream from the tip. One can remark that all the thermocouples of a given model indicate a transition Reynolds number in the same range with consistent results i.e. the values are either a slightly decreasing or a constant function of Y^+ which means that, as soon as the transition occurs at a given location Y^+, it spreads downstream along the attachment line. Some of the configurations, referenced with subscripts *a*, *b* or *c*, have been tested several times, giving fairly coherent results with discrepancies lower than 15% for $\bar{R}^*_{tr1}$. Moreover part of the apparent discrepancies might be attributed to a difference in the wall to stagnation temperature ratio.

As cylinders of different diameters have been used it is possible to perform some cross-check of the results, for instance with models C1480 and C1980 or with C1978 and C3379. Though these tests correspond to different wind tunnel stagnation conditions, the differences for $\bar{R}^*_{tr1}$ are less than 10%. For instance, transition on model C3379 occurs for stagnation conditions $P_i = 43 \times 10^5$ Pa and $T_i = 790$ K instead of $P_i = 57.5 \times 10^5$ Pa and $T_i = 680$ K on model C1978; it is clear that, for such variations, one can expect significant changes in the level of free-stream disturbances. Unfortunately a systematic investigation of the free-stream turbulence level is not available for the wind tunnel used in the present study. According to the synthesis of Harvey (1978) one can expect a typical free-stream pressure fluctuation level $p'/p_\infty \simeq 3\%$ by comparison with other wind tunnels of the same type. Usually the free-stream noise level does not influence very much attachment line transition as demonstrated by Creel et al. (1986, 1987). However if we follow the analysis of Gaponov (1983) some sensitivity to the free-stream fluctuations can be expected in the inception region where the attachment line boundary layer is still growing in the spanwise direction.

Fig. 5 gives the spanwise evolution of the value $\bar{R}^*_{tr2}$ corresponding to end of transition. One must notice that this value is usually determined more accurately than for $\bar{R}^*_{tr1}$ because it corresponds to a more sudden change of the slope of the curve of the Stanton number with respect to the Reynolds number. Unfortunately it was not always possible to reach Reynolds numbers sufficiently high to observe the end of transition and this explain why the set of results in Fig. 5 is much smaller than in Figs. 3 and 4. However the available results are clearly independent of the spanwise location. The difference between the Reynolds numbers for the beginning and end of transition is therefore nearly independent of the spanwise location at least for the range of Y^+ values available in the present experiments. It is perhaps worthwhile to compare the spanwise extent of the models used in the present study with those used in subsonic or incompressible flow studies: the typical values of s/η, where s is the distance from the cylinder tip and η the

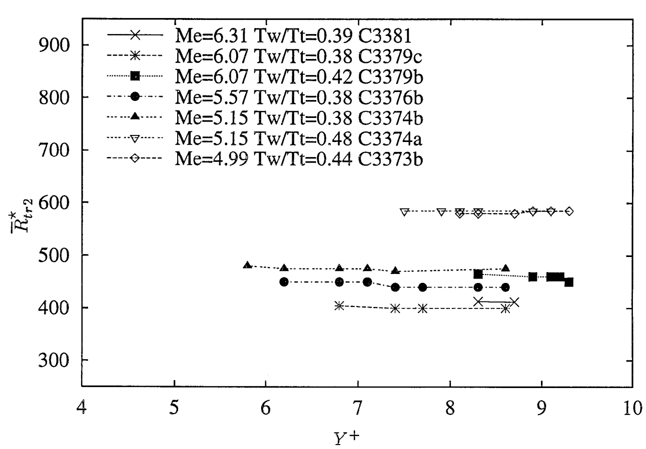

Fig. 5. Influence of the spanwise location on the end of transition ($D=33$ mm)

boundary layer thickness scale, are in the range $s/\eta = 3667$ ($s/\eta^* = 774$) for C3381 up to $s/\eta = 7411$ ($s/\eta^* = 2218$) for C3373. For comparison the study of Cumpsty and Head (1969) was performed at $s/\eta \simeq 10000$ and the study of Poll (1979) at $s/\eta \simeq 8000$, that is at much larger values, specially if one assumes that the proper reference scale is η^* instead of η. Therefore, although the present experiments do not indicate any tendency for a decrease of the difference $\bar{R}^*_{tr2} - \bar{R}^*_{tr1}$ when Y^+ increases, it is still possible that the intermediate state between a laminar and a fully turbulent flow does not survive farther downstream.

3.2
Influence of the wall temperature

Some tests were performed with a low wall temperature achieved by cooling the model with liquid nitrogen before running the wind tunnel. In order to prevent icing on the model, which could introduce some wall roughness, the test section was filled with gaseous nitrogen during the cooling process. Figure 6 shows that the Reynolds number value for the beginning of transition has only a weak dependence upon the spanwise coordinate and reach a nearly constant value for $Y^+ > 8$.

Comparison of the values $\bar{R}^*_{tr1}$ of test cases C3379, C3376 and C3374 with the corresponding cases for a very cold wall (last three cases with subscripts C in Table 2) shows that wall cooling has only a rather weak influence being perhaps slightly stabilizing for C3374 at $M_e = 5.15$ and rather destabilizing, either in terms of $R_{D\infty,tr1}$ or in terms of $\bar{R}^*_{tr1}$, for C3376 and C3379 at higher values of the spanwise Mach number. If one assumes that transition occurs at a constant value of $\bar{R}^*$ then the results suggest that the coefficient of T_w in Poll's reference temperature formula might be higher than 0.1: for instance, a value of 0.25 will allow to reach nearly equivalent values of $\bar{R}^*_{tr1}$ for the tests C3376 at the two wall temperatures. Clearly these results are not in agreement with the strong stabilizing influence of wall cooling observed in the numerical investigation of Malik and Beckwith (1988) confirmed by Arnal et al. (1990). However these numerical predictions are based on a TS waves approach which is perhaps incorrect as discussed in Section 1. One must also notice that, for a flat plate boundary layer, Vigneau (1985) has shown that the influence of wall cooling on stability was strongly dependent upon the free-stream Mach number and rather weak in the hypersonic range.

3.3
Comparison with other works

The present results for $\bar{R}^*_{tr1}$, plotted with respect to the spanwise Mach M_e, are compared with results from other authors in Fig. 7. We have taken the mean value or our results when several values were available for the same Mach number. The low values obtained by Yeoh (at $M_e = 1.78$) and Skuratov (at $M_e = 1.87$) correspond perhaps to a contamination by the nozzle boundary layer for Yeoh's tests or to roughness induced by the thermosensitive coating for the Skuratov's tests. Our results suggest a strong destabilizing influence of an increase of the spanwise Mach number above $M_e \simeq 5$. As shown in Fig. 8, the same behavior is observed if we plot the value of the criterion suggested by Bushnell $R_{D\infty,tr1}$ with respect to M_e. In Fig. 9 we have also plotted the values of the Reynolds number based on the spanwise momentum thickness and the attachment line conditions $R_{\theta,tr1}$ which is also a decreasing function of the spanwise Mach number. Of course these results must be considered with caution because it is always possible that the transition was induced by some very efficient source of parasitic excitation not detected during the experiments. However one should stress the following points:

- The results have been obtained from several test campaigns by two experimenters (but of course in the same test facility).
- Below $M_e = 5$ we find fairly high values (rather higher than other experimenters). As the same models were used for low and high spanwise Mach numbers (by changing only the sweep angle) this demonstrate that transition was not induced by some local roughness on the model wall. Moreover it was checked several times that the height h of the wall defects was not higher than 10^{-3} mm. This corresponds to $h/\eta \leqslant 0.03$ and it is generally believed that the smooth regime is achieved below 0.5. If one assumes, like Holden and Kolly (1995), that the proper reference length is η^* the values of h/η^* are typically about 5×10^{-3}.

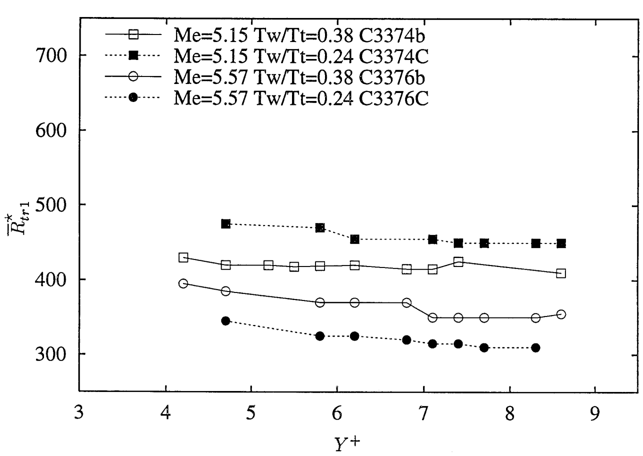

Fig. 6. Influence of wall cooling (filled symbols for $T_w/T_t \simeq 0.24$ and hollow symbols for $T_w/T_t \simeq 0.38$)

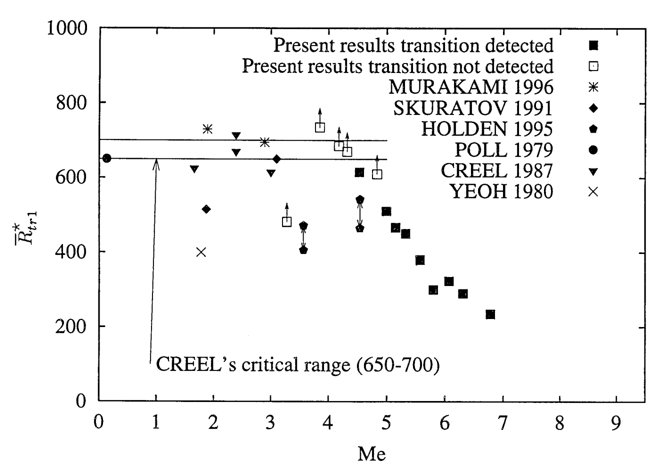

Fig. 7. Influence of the Mach number on the beginning of transition

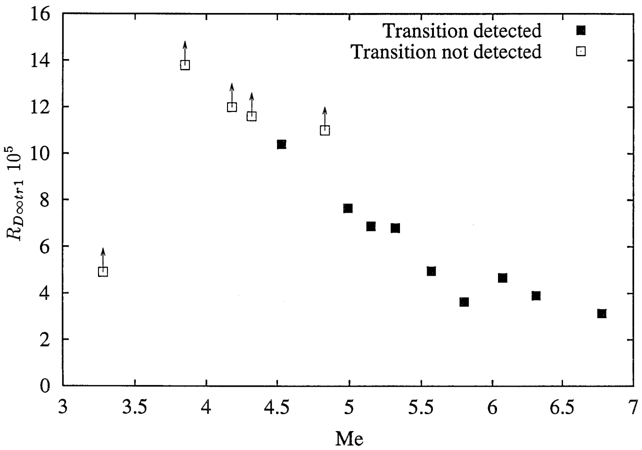

Fig. 8. Influence of the Mach number on the beginning of transition

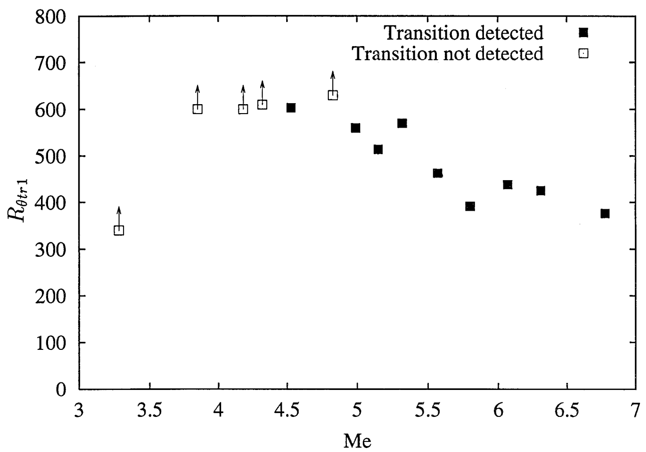

Fig. 9. Influence of the Mach number on the beginning of transition

- The results from a consistent set with a fairly continuous evolution with respect to M_e, confirmed by the results obtained at very low wall temperatures.

Among the possible explanations for this dependence upon the Mach number, one must notice that the high spanwise Mach numbers results correspond to high sweep angles. For the same free-stream conditions the ratio η/D, which characterizes the curvature effects, is then an increasing function of the sweep angle and therefore of the Mach number. However an explanation related to the influence of curvature is rather unlikely because Lin and Malik (1997) have shown that, at least in incompressible flows, an increase of curvature is stabilizing. A more likely explanation is an increasing receptivity to some form of free-stream perturbations. According to Gaponov (1983) a significant receptivity by free-stream sound perturbations can be expected when the boundary layer is growing: one can notice that the inception length from the model tip is an increasing function of the sweep angle and it is therefore possible that the dependence upon the spanwise Mach number comes from a high receptivity of boundary layer in the upstream part of the model.

4
Conclusion

Smooth leading edge transition in hypersonics was investigated with local Mach numbers $M_e = 3.28$ to 6.78. Poll's transformed Reynolds number corresponding to the beginning of transition seems to be a decreasing function of the spanwise Mach number M_e above $M_e \simeq 5$. The values of the momentum thickness Reynolds number corresponding to the beginning of transition are also decreasing with respect to the spanwise Mach number.

Although the flow is, in principle, independent of the spanwise coordinate, a significant difference was found between the Reynolds numbers corresponding to the beginning and end of transition. A collapse of this difference might be expected for an infinite long cylinder. However, it was not observed in the present experiments presumably because of too low values of the ratio of the spanwise length to the cylinder diameter.

The influence of wall cooling is rather weak with a slight tendency to be destabilizing with respect to Poll's criterion for the highest values of the spanwise Mach number. This suggests perhaps to increase slightly the coefficient of the wall temperature in Poll's reference temperature formula.

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