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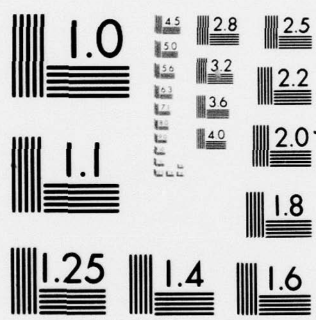

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TOWARDS NEW TRANSONIC WINDTUNNELS

Edited by

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This AGARDograph is dedicated to
the memory of Dietrich Küchemann.

DIETRICH KÜCHEMANN

DIETRICH KÜCHEMANN; AN APPRECIATION

Professor Dr. Dietrich Küchemann, CBE, Dr.rer.nat., FRS, F.Eng., FRAeS, FIMA, FAIAA, Hon.D.Sc.(CIT), Dr.Ing.E.h.(Berlin), Hon.D.Sc.Eng(Bristol), Hon.M.Aero. Soc.(India), was born in Göttingen on 11th September 1911 and died, a naturalised British Subject, on 23rd February, 1976. Much honoured in his later years*, the career in which he gained so much distinction began wholly by chance. For when he first entered the University of Göttingen in 1930, it was understood that he would ultimately become a student of Max Born, a close friend of his father. But, by the summer of 1933, Born, Courant and Franck had all gone, expelled by the Nazis, and Göttingen's school of mathematics and physics had virtually collapsed. Of the great men on whom its reputation had been built, only Prandtl remained. And it was under Prandtl and W. Tollmien that Küchemann studied for his doctorate; at a time when the field of theoretical and applied aerodynamics which he had entered was ripe for the rapid development that was to be demanded of it by the spectacular advances in aircraft design and propulsion during the war years.

He submitted his doctoral dissertation, "Stability of gas flows with linear velocity distribution along a flat plate", in 1936, and then joined the AVA, Göttingen, as a scientist in its theoretical aerodynamics department. For the next two years his departmental head was Irma Flügge-Lotz. But it was a department in which Professor A. Betz, the Institute's director, took a special interest. And when Flügge-Lotz left in 1938, Betz took it over as his own special responsibility and somehow managed to keep its members together and active in research throughout the war and, at the same time, to offer some sort of protection to the instinctive anti-Nazis among its members, of whom Küchemann was undoubtedly one. The effectiveness of this protective influence must surely account for the appearance in 1941 of two papers by Küchemann and his friend Erich von Holst on the aerodynamics of bird flight: at a time when von Holst was being employed by Betz for 'special work', and was being kept out of the army for it.

Küchemann himself had volunteered for the army in 1938, largely because it seemed prudent to do so, and served, after a fashion, in the Luftnachrichtentruppe (Signals). He was called up for training for short periods in 1938 and 1939, and held the rank of Unteroffizier from 1942 to 1945, but saw no real active service because Betz claimed him back for the AVA. Joining the army seems to have been a calculated gesture: one of the few positive actions that could be taken by a German patriot, at that time, without the implication of any approval of the Nazi party. To have appeared to wish to avoid any form of war service would have been particularly dangerous for him, for his family had already suffered for its political beliefs, and it would have been pointless to present the authorities with what might have been seen as further evidence against them. As it was, his father, a distinguished liberal-minded and progressive schoolmaster, had already been deprived of his school, the Oberrealschule in Göttingen, where the young Dietrich was a pupil from 1921 to 1930, and was having to struggle to make a living through private teaching.

In the main, however, Küchemann immersed himself in his research. By 1940 he had taken the first steps in a study of the aerodynamics of propulsion that was to occupy him for the rest of the War period. It began with his initiation of research on air intakes, through which was formed a working partnership with Dr. Johanna Weber that was to remain active, and extremely fruitful, for the rest of their working lives. Their prolific output of research reports, both theoretical and experimental, during the period 1940-1945 bears witness not only to their industry but to the comprehensive nature of their attack on the propulsion problem. And it was what they established then, on the properties of annular aerofoils, on the installation of coolers, on ducted propellers and on the installation of jet propulsion units, that was to provide the core of their book "Aerodynamics of Propulsion", published by McGraw-Hill in 1953.

So, by 1945, Küchemann had become and authority, perhaps the authority, on propulsion aerodynamics. But he was no narrow specialist, and it is safe to assume that he had kept himself informed, very well informed, of all that had been going on around him. It is not surprising, therefore, to find that when the War ended, and the AVA was closed by the occupying forces in May 1945, it fell to Küchemann to guide visiting scientists around the Establishment, and to inform them generally of the work that had been going on there. Otherwise, he and his colleagues remained in a state of some suspense; still nominally on the staff, but without pay, and reduced to labouring work as the only source of income available to them, until the AVA reopened in August of that year.

The end of the War in Europe marked the end of an epoch in which scientists had concentrated their attention, almost exclusively, on the application of their knowledge and techniques to the waging of war. And the time had come for the scientists themselves, and for the governments that employed them, to take stock of what had been achieved, and to project those achievements into the post-war world. Aviation, in particular, had undergone massive advances since 1936, when Küchemann first joined the AVA, and, with the invention of the gas turbine engine, was on the brink of a revolution that would bring high-speed, and even supersonic, flight within the grasp of the aeronautical engineer. But aircraft are not, inherently, weapons of war. And what had been learned and successfully applied in war could now be applied also to the development of civil aviation.

It was an exciting prospect. Scientific delegations descended upon Germany, hot on the heels of the occupying armies, intent on rescuing the records of German research and development to complement their own experience. But they need not have worried. Men like Küchemann were as determined as they were to ensure that nothing of value to a peaceful world should be lost, and had even taken active steps to preserve those records to which they had access. But they had more to offer than their reports and working files, and the better experienced among them were given the opportunity to continue their work in one or other of the allied nations: an opportunity that many accepted and that, for some, would amount to permanent emigration. However, Küchemann at first elected to remain at the AVA. He rejected the opportunity to move to the United States, to which he had been 'allocated', not through dislike of America or Americans but because he saw himself, and wished to remain, a European. And so, with Göttingen and the AVA occupied by the British, he began an association with British scientists that was to dominate the rest of his life.

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* Elected Fellow of the Royal Society in 1963, appointed Commander of the Order of the British Empire in 1964, awarded the Silver Medal of the Royal Aeronautical Society in 1962, its Gold Medal and the Enoch Thulin Medal of the Swedish Society for Aeronautics and Astronautics in 1969, the Ludwig-Prandtl-Ring of the Deutsche Gesellschaft für Luft and Raumfahrt in 1970, and honorary degrees by the Cranfield Institute of Technology in 1973 and by the Technische Universität, Berlin, and the University of Bristol in 1975.

He was not persuaded to come to Britain until September 1946, and then only for six months. But when that first short-term contract expired, he accepted another, and persuaded Dr. Weber to join him to renew their collaboration of the war years. There then followed a series of yearly contracts, during which they became increasingly absorbed into the work of the Aerodynamics Department of the Royal Aircraft Establishment, Farnborough. Küchemann's family joined him in England in 1948, and he expressed a willingness to accept a long-term contract in the knowledge that this might involve naturalisation. But the yearly contracts continued until 1951, when he was finally assimilated into the normal staff structure of the establishment as a Principle Scientific Officer. His outstanding contributions to the work of the R.A.E. were rewarded in 1954 by promotion to Senior Principal Scientific Officer, on individual merit, a post that carried with it no administrative responsibilities, and was a recognition solely of his personal status as a research scientist.

But Küchemann's value to the R.A.E. depended at least as much on his influence upon others as on his own personal research. This side of him was recognised in 1957 by promotion to Deputy Chief Scientific Officer and the appointment of Head of Supersonics Division of Aerodynamics Department, followed by a further promotion to Chief Scientific Officer and Head of Aerodynamics Department in 1966. He finally retired from this last post in 1971, was awarded the rank of Chief Scientific Officer on Special Merit, and then remained in the Aerodynamics Department until his death, once again as an individual research scientist.

Küchemann's contributions to aeronautical science* were concerned, almost exclusively, with the aerodynamic design of aircraft: the subject of his last book, which he described in a sub-title as "A detailed introduction to the current aerodynamic knowledge and practical guide to the solution of aircraft problems". It was a subject that he made very much his own, and that he pursued with immense vigour and a remarkable single-mindedness. So much so that the physical insight and mathematical skill, on which his undoubted mastery of aerodynamics was based, were reserved, in his mind, solely for the aeroplane. That alone occupied his attention, concentrated his concern for the future of the world, gave point to his work. With the exception of his wartime study of bird flight, he refused to allow his interests to be deflected by the many applications of aerodynamics to fields other than aeronautics to which he could have turned and doubtless made significant contributions.

His career began at a time when aeronautics was dominated by the classical low-speed aircraft, with its high aspect-ratio unswept wings, and aerodynamics by Prandtl's boundary layer concept and the lifting-line theory of finite wings that derived from it. Prandtl's ideas were, by then, well established. They led to the conclusion that, provided the boundary layer remained thin, and unseparated, everywhere over the surface of a high aspect-ratio wing, any chordwise section of that wing would possess, to a first approximation, the same aerodynamic properties as the same wing section in a related two-dimensional flow. Thus at that level of approximation, aerodynamic design reduced to the design of two-dimensional aerofoils, in which the first principle of the design process was to seek a shape for which the thin boundary-layer, or streamline, flow could be maintained over a practically acceptable range of attitudes. A second principle, the incorporation of a sharp trailing edge in the aerofoil shape, derived in part from the first: from the observation that streamline flow appeared to demand such a shape, and in part from the practical need for a unique relation between the aerofoil's properties and its attitude.

Thus in relation to the classical aircraft, the engineer's demands on aerodynamic design could be met by the assembly of a catalogue of aerofoils and their properties, from which he could readily select shapes that would suit his particular needs. To go further demanded improved understanding of the mechanics of complex three-dimensional compressible flow fields, and at least some awareness on the part of the research scientist of the nature of the problems of application facing the engineer. But Küchemann seems to have recognised, more clearly than most of his contemporaries, that a simple generalisation of the first design principle for the classical wing, to regard it initially as a demand for streamline flow over a wing as a whole, and later for a type of flow that was in some sense, practically acceptable, appeared to allow the original restriction to the high aspect-ratio unswept wing to be dropped completely, to open the way to the design of wings of much more general planform, and to allow the interactions between the major elements of an aircraft to be understood and accounted for. He, more than anyone, ensured that advancing knowledge was matched by corresponding advances in design philosophy, and by the development of techniques, both theoretical and experimental, that would make those advances applicable.

The development of that philosophy gave continuity both to his own research and to the work that he inspired in others and through it he was led to recognise the way ahead in the design of sweptback wings: to recognise, in particular, the importance of the loss of the 'sweep effect' at the centre and tips of such a wing, and to devise ways of countering it: to accept readily the practicability of leading-edge vortices as a potential basis for the design of slender wings, which played so important a part in the guidance of the research that led ultimately to Concorde: and to accept also the 'wave-rider' as a possible aerodynamic solution to flight at hypersonic speeds.

But many other examples could be cited of the practical consequences of Küchemann's design philosophy. It underlay almost everything he did. And while it developed steadily through the years, its basic principles never changed significantly. To those who worked with him they largely define Dietrich Küchemann the aeronautical scientist. And for most of us they have become basic principles of our own.

At the same time, in our relationship with Dietrich, we enjoyed something else, felicitously shared by all who came into contact with him: an immediate affection towards him, strengthened and made more profound by continued acquaintance. Reverence for his wisdom and knowledge, his distinctive approach to scientific and engineering questions, was inescapable; but his modesty, at times a self-effacing humility, an interest in people, compassion, sensitivity, an inexhaustible generosity in listening to the problems of others, helping them, encouraging and teaching them, were qualities that generated love: not the selfish love born from the expectation of approval by a great man: something simpler, deeper, more responsive to his warmth, his conviction that indeed "alle Menschen werden Brüder". Yet, Küchemann was not a person genuinely to be thought about or related to in simple terms. He was too big a man to permit his qualities to be thus described in the facile language of every-day life. He possessed what so often accompanies greatness: an inner

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* of which a permanent record is provided by his massive output of published works. In addition to the books, "The Aerodynamics of Propulsion" (with J. Weber, McGraw-Hill, 1953) and "The Aerodynamic Design of Aircraft" (Pergamon, 1978), a comprehensive list of papers and articles is contained in "The Publications of Dietrich Küchemann" (R A E Library Bibliography 357, by R.W. Slaney, 1976).

contradiction, in his case between a belief in the essential harmony of mankind and a hesitancy, at certain periods of his life, to commit himself fully to its society. It was an inconsistency that added to his charm: a source of perpetual wonder to his friends: a frailty which intrigued them and attracted them still more strongly towards him.

The circumstances which nurtured that wish for detachment can only be guessed at. Perhaps the guess is wrong, but, as we have seen, the elements in his history which could have tended to set him apart a little from his fellow men are incontestable. He attended a school of which his father was the Headmaster; a relationship difficult enough for a child to manage when it devolved upon office alone, but the elder Küchemann did more than occupy a relatively exalted office: he was overtly anti-Nazi, resolute enough in his opposition to the movement to make his dismissal from the post inevitable. Accordingly, the young Dietrich was deprived of the opportunity to behave and be accepted as an ordinary schoolboy. Later, as a University student and then as a research-worker, he experienced further isolation, growing-up in a society with which he identified himself but whose leadership he rejected emotionally and intellectually: an attitude, for someone of his integrity, that allowed no moral compromise but nevertheless demanded caution, a habit of circumspect aloofness in his relationship with other students and most of his colleagues. Perhaps, that habit of being ostensibly part of a social system without belonging unreservedly to it remained with him for, more than thirty years afterwards, as Head of the Aerodynamics Department at Farnborough, he complied impeccably with the procedures and administrative rules of the British Civil Service whilst being critical of them, frequently to the point of impotent despair.

What might have decisively contributed to his feeling of singularity, even more than the tortured structure of Germany at that period, was his own originality; because, as a research-worker at Göttingen, his special way of looking at problems, of refining aerodynamic theory in order to adapt it to some well-defined practical aim, left him vulnerable to criticism. He must often have had to defend an approach as unconventional then as, later, it was to be influential in creating a new system of aerodynamic design. In doing so in the early 1940's, he was undoubtedly sustained by the conviction that he was right. But even that had to be complemented by a strength of character, a purposefulness, in pursuing ideas sometimes hostile to his scientific colleagues and probably beyond the comprehension of those who might have been able to make use of them.

The need to preserve a belief in his ideas and to encourage their acceptance by others never left him. But the task could only be accomplished by a partial disengagement from the community in which he worked. He was never dogmatic about his theories: indeed, he listened patiently, courteously, when attempts were made to demolish them. Nonetheless, he was unresponsive, unyielding. His manner of dealing with is critics, in conversation or at a large, formal meeting, was as beautifully simple, reticent, dignified as it was effective. He would expound an idea, quietly listen to its rebuttal, then repeat his argument with an imperturbable and innocent disregard for all the attempts made to refute it. Thereby, the critic was distracted, persuasively urged to examine his own logic rather than continue to attack Küchemann's. At the same time, it might well have afforded Dietrich an opportunity to indulge privately in a sense of humour that was a constant delight to his friends and colleagues.

His protective shell, a sort of intellectual armour encumbering a man whose need for emotional links with people around him was insistent, made him somewhat bemused by the many honours that came his way; as if their purpose were slightly suspect. Did they genuinely represent a personal recognition, or were they elaborate social artefacts to install him in an establishment to which he did not, would not willingly, belong? All the same, he did not belittle those honours individually. One of them, his Visiting Professorship at Imperial College, provoked the remark, poignant and sincere: "This is just what my family had hoped for me. It would have made them very happy".

Whatever dilemmas he posed for himself in his relationships with people outside his home, and certainly not reciprocated by them, were unobtrusive in his attitude to music. It represented a form of expression unconstrained in passion, neither needing nor receiving the guarded appraisal his upbringing had accustomed him to make of other activities. He was a gifted 'cellist, who could readily have fashioned a professional career as a performer, very nearly deciding to do so at the end of the War in Europe had his scientific work not been compelling and the opportunity to pursue it available. Even so, he subjected his talent to merciless criticism, complaining with characteristic modesty that his technique of fingering was impaired by his hand having too small a span: an impediment those who delightedly heard him play were never aware of. His knowledge of music was vast, his tastes discriminating but showing nothing of the snobbery that sometimes goes with the amateur. He would play a piece from a Sullivan light opera with a similar dedication and enjoyment that he showed for a Bach partita: although it was the latter that engaged his rapture and communicated to his audience something of the expansive emotional qualities he possessed and which he compelled himself strictly to discipline.

His belief in the unity of Europe was unequivocal, to an extent that led him to accept seemingly menial tasks: for fear that they might not otherwise be performed scrupulously. Whether they involved the secretaryship of a body such as Euromech of which he was one of the original architects, or the more revered, comparably influential, position as chairman of the AGARD Fluid Dynamics Panel, his commitment was complete: his ideas on policy precise: and his determination to secure their acceptance and realisation inexhaustible. Even after his health had begun to fail, he travelled incessantly, often confining himself to a hotel room late into the night in order to compose the minutes of an international meeting; when completed, they set out with clarity in a language difficult to conceive not to be originally his own, conclusions and arguments that must have aroused the envy of those initially responsible for them.

Above all else in engineering, he had faith in the future of Aviation, not just in the mechanistic sense, but as a force for good in the World, a vehicle for communication and understanding between its peoples. In his last work, "The Aerodynamic Design of Aircraft", a monumental summing-up of his scientific philosophy and the way he adapted it to the solution of engineering problems, he asks "Is our work significant and worthwhile with regard to human society and the way we live? What is the social motivation of aviation?" Strangely, he does not really give the answers, but assumes, in the final pages of the book, that the reader has worked them out for himself and thereby shares the conviction that they are benign and assuring of a permanent future for the aeroplane. Dietrich Küchemann to the end has used the debating technique he so often found successful at meetings. He was right then. Let us hope that he will continue to be proved right.

Whether or not that turns out to be so, he has bequeathed to Aeronautics an indestructible testimony to his greatness: permanent evidence of the revolution he brought about in aerodynamic design. His name will remain familiar to future generations. But there are those of us who were more fortunate in having an opportunity to work with Dietrich Küchemann; in our warm friendship with him; in possessing a memory of him which we cherish. Some perplexities remain; and it is a tribute to his character and stature as a man that we ponder them still. But we do so in spirit of the deepest and most lasting admiration, respect and affection.

P.R.O.
E.C.M.

KÜCHEMANN'S ROLE IN THE PROMOTION OF NEW WINDTUNNELS

Küchemann's invaluable endeavours for a new transonic windtunnel as an international concept can probably best be described by setting them in historical order.

Following the Specialists' Meeting on "Transonic Aerodynamics" in 1968, organized by AGARD's Fluid Dynamics Panel (FDP), a small Evaluation Committee was set up, with Küchemann as chairman, which produced the Technical Evaluation Report[2] of that meeting. Amongst others, one of the conclusions was that the meeting had shown the need for better experimental facilities and techniques and that "the Fluid Dynamics Panel should undertake to examine the needs of the NATO nations (particularly Europe) with regard to providing wind-tunnel facilities for testing large models at high Reynolds numbers". It was also suggested that "AGARD might be instrumental in bringing it about".

Küchemann then continued along these lines to push the case for large European windtunnel facilities, both inside the UK as well as outside. On his recommendation[2] the FDP held a Round Table Discussion in September 1969 on "High-Reynolds-Number Problems"[3]. Küchemann gave a presentation on the transonic flow over swept wings. Many other speakers covered such subjects as the types of windtunnels which could be considered and also the possibility of a European project.

As a direct consequence of this Round Table Discussion the FDP chairman wrote to the Director of AGARD and proposed the setting up of a Working Group (High Reynolds Wind Tunnel or HiRT Group). The HiRT Group (with membership from four European nations and from the USA) produced the results of its study[4] in the timespan of about one year. The report concentrated on transonic tunnels and it was concluded that there existed a critical need for provision of high Reynolds number tunnels within NATO and that two new types of transonic windtunnels were required, a Ludwieg Tube and a blow-down tunnel.

In November 1979 the FDP chairman made a Special Report[5] to the AGARD National Delegates, calling their attention to the results of the HiRT Group study.

A second follow-up from the above mentioned Round Tabel Discussion was the organisation, by the FDP, of a Specialists' Meeting having the primary aims of clarifying what facilities shoud be specified for NATO purposes and what techniques should be developed for use in the near future.

This meeting was held in Göttingen during April 1971[6], and Küchemann was instrumental in its success by acting as chairman of the Programme Committee and as chairman of the Round Table Discussion held at the end of the meeting. In reading the transcript of that discussion it is clear that the technical forum supported Küchemann in his conclusion: ".....so hasn't the time come when we ask our Masters to act, where we can say we have gone far enough to ask them to take this matter seriously and to support further work which would include financial support,.....".

One week later, the Defence Research Group of NATO organized (at the suggestion of their French representative) a Seminar in St. Louis, France, to discuss problems relating to aerodynamic testing facilities[7]. Here, Küchemann brought forward the conclusions of the FDP Specialists' Meeting: "The best technical advice available within AGARD leads to the conclusion that one or several large new windtunnels would contribute immensely to the effectiveness of a large number of aerospace systems now planned or contemplated within the NATO nations" and "... if we want to face up to our responsibilities for the future, we need to pool our resources... Now is the time to act and to go ahead together".

At the end of the Seminar a resolution was agreed[7], which was acted upon by both the Defence Research Group (DRG) and AGARD.

The DRG formed an Ad-Hoc Working Group on "The Collaborative Possibilities for the Provision of Major Aerodynamic Testing Facilities in Europe" (AEROTEST) and AGARD formed a Working Group of the Fluid Dynamics Panel on the subject of "Large Windtunnels" (LaWs). The latter was concerned with the technical definition of the facilities needed in the future for research and development of conventional aircraft (both combat and transport), helicopters and V/STOL aircraft.

Küchemann took upon himself the chairmanship of the LaWs Group. Under his vigorous leadership the Group, with a membership from 8 nations (including the USA and Canada), met 10 times in the period from December 1971 till December 1972. The active help of more than 75 scientists from research establishments and industry was organized and a total number of 132 papers were produced especially for the LaWs Group. The most important were collected in a number of AGARD reports[9,10,11]. The first LaWs paper[12], written by Küchemann, already gave in detail the plans he had towards the realisation of the first LaWs-Group report[13]. The report defined the needs for new aerodynamic testing facilities, covering all speed ranges. It concluded, however, that first priority should be given to the provision of a new large transonic windtunnel of nearly equal importance.

Furthermore, upon Küchemann's suggestion and supported by the LaWs Group members, it was recommended that the European countries involved should devote some part of their National Research Programmes to a Collaborative Programme of Work is defined by the Group[13], which should provide technical information needed for the design and operation of the European windtunnels.

As four possible options were proposed for the drive system of the large transonic tunnel it was recommended that Engineering Studies should be undertaken on these options and that the LaWs Group should meet again to consider these technical options and recommend a preferred drive system.

On behalf of the four nations which supported the studies (France, Germany, UK and the Netherlands), AGARD subsequently placed a contract for the Engineering Studies with the consulting firm DSMA of Toronto. A report[14] appeared in April 1974 and, under Küchemann's chairmanship, the LaWs Group produced its second report[15], confining itself to the required transonic tunnel. It was concluded that, on the basis of the results of the work done by DSMA and by the originators of the proposed drive systems a recommendation on a preferred drive system could not yet be made, that further work would be necessary. In addition the required flow quality had to be difened and examined for each of the options. It was recommended to set up a Technical Project Group of full-time professional engineers for an independent assessment of the information provided.

In the meantime the above mentioned AEROTEST Group of the DRG had produced its report[16] in early 1973. Küchemann had maintained the liaison between the LaWs and AEROTEST Groups. It was recommended to set up the Project Group for Transonic Windtunnel Definition and to draw up a general Memorandum of Understanding for co-operation in the field of provision and utilization of aerodynamic testing facilities, under the aegis of a Co-ordinating Committee. Both recommendations were acted upon, the DRG set up Project Group AC/243 (PG.7) and the general MoU between the four participating countries was signed.

As recommended in the first LaWs Report[13] the Fluid Dynamics Panel took the initiative in setting up a small Working Group (Mini LaWs) to find out what was being done, to identify gaps and to recommend regions for closer co-operations and for work-sharing in the field of operation and design of new windtunnels.

Again we find Küchemann as the active chairman and inspiration of this group, which worked for two years from March 1973 until March 1975, producing two reports with a review of current work and recommendations on what should be done in a collaborative way[17,18]. Upon his initiative experts from both sides of the ocean worked together to report on the current situation and to recommend further activities in many important fields, such as noise measurements, transonic test-section design, model design, etc.

When the Mini LaWs Working Group was reshaped in to a Sub-Committee by the Fluid Dynamics Panel, because it looked as though the work was going to be a worthwhile, continuing effort, Küchemann gave up his chairmanship of this activity. This was mainly because he had accepted the chairmanship of the Fluid Dynamics Panel in the autumn of 1973 and he felt it difficult to combine the two activities. There had been signs that his health was failing.

The further activities of Küchemann in connection with large windtunnels included a Course Directorship for a Lecture Series "Course on Large Transonic Windtunnels" at the Von Kármán Institute in Brussels in early 1973[19]. This course was mainly based on the inputs to the work of the LaWs Group.

Finally, under his chairmanship of the Fluid Dynamics Panel a Specialists' Meeting was held in London in October 1975 on the subject of "Windtunnel Design and Testing Techniques"[20], a suggestion which originated from the Mini LaWs Group work. In his closing-speech at the meeting Küchemann again made a strong case for collaboration in the field of aerodynamic research and for a European Transonic Windtunnel and finished by saying: "Now let us do it". Two days later he resigned as chairman of the Fluid Dynamics Panel, in which function he had also tried to make a strong case for the advocated large European windtunnels and for strong collaboration and an exchange of data with the USA.

Until five months later, when his unexpected death came as a shock to the many people who knew him, Küchemann stayed highly interested in the progress made in connection with the proposed European transonic windtunnel.

He could get very pessimistic when things were not moving in the difficult field of international co-operation as quickly as he would have liked. On the other hand he was highly delighted when things began to move again unexpectedly.

When, as we hope, the European Transonic Windtunnel is realized, the aerospace industries in the European countries will owe Küchemann a large debt for his excellent ideas, his continuing initiatives, his leadership and his perseverance.

J.P.H.

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AN INVESTIGATION OF THE QUALITY OF THE FLOW GENERATED BY THREE TYPES OF WIND TUNNEL (LUDWIEG TUBE, EVANS CLEAN TUNNEL AND INJECTOR DRIVEN TUNNEL)

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SYMBOLS

A), B)	Constants in calibration of hot film probes
b	Number of filters (see section 2.4)
E	Output voltage
f	Frequency
Δf	Band-width
M	Mach number
n	Reduced frequency
p	Static pressure
q	Kinetic pressure of flow in test-section
T	Absolute temperature
U	Fluid velocity (mean)
u	Fluctuating longitudinal component of velocity
v	Fluctuating lateral component of velocity
V_i, V_o, V_{o_p}, V_{o_m}	Voltages associated with analysis of fluctuating signals (see section 2.4)
α, β	Angles between local flow direction and centre-line of test-section or settling chamber in pitch and yaw planes respectively
ε	Relative bandwidth ($\Delta f/_f$)
ρ	Fluid density

SUPERSCRIPTS

*	Corrected values (see section 2.3)
$\sim$	rms value of fluctuating component

SUBSCRIPTS

1	immediately upstream of contraction
O	stagnation values
s	sensor
∞	test-section, free-stream values
u_1, u_2	reference values (see section 2.3)

1 INTRODUCTION

Previous studies[1] have addressed the problem of defining the flow quality required if accurate aerodynamic data are to be obtained quickly. In these, flow quality was characterised by the level and spectra of static pressure fluctuations and overall turbulence levels. This usage was enforced by previous experimental practice which, largely for reasons of practical convenience, had assumed that the levels of fluctuations in static pressure were an index of flow quality generally. The same assumption had to be made when relating rates of data acquisition for various types of test to flow quality in existing facilities and, hence, interpreting this data into flow quality requirements for given speeds of data acquisition.

Although some evidence is available to suggest that the unsteady aerodynamic excitation of models under test is related to the levels of free-stream static pressure fluctuations[1,2], such a correlation cannot be regarded as totally reliable under all circumstances. Changes in static pressure, particularly at typical wavelengths implied by the natural frequencies of most models, are unlikely to cause significant changes in aerodynamic forces on models. Such excitation is more likely to be due primarily to changes in angles of pitch or yaw or to changes in dynamic pressure. The use of static pressure fluctuations as an index of flow quality thus presumes a correlation between this type of flow unsteadiness and other, probably more significant, types of unsteadiness. That such a correlation will always exist is far from self-evident and may be especially doubted when comparing wind-tunnels having very different and novel forms of drive system and, hence, different sources of unsteadiness.

The Technical Working Group for the Large European High Reynolds Number Transonic Wind-tunnel (LEHRT) (NATO DRG AC243 (PG7/WG1)) thus considered that conventional measurements of static pressure fluctuations needed to be supplemented by other data when comparing the qualities of the flow generated by pilot versions of 3 types of wind-tunnels then being considered as options for LEHRT. Clearly, static pressure and turbulence data were necessary for comparison with the requirements laid down in Ref 1; but additional measurements were desirable in order to confirm that comparisons drawn on the basis of static pressure fluctuations were not misleading because of differing relationships between this characteristic and other types of flow unsteadiness. They considered that the most appropriate forms of additional data would be unsteady analogue of those measurements that are normally made in order to assess the spatial uniformity of the temporal mean flow. Such data could most readily be interpreted in the light of available experience of tolerable non-uniformity of the mean flow and could most easily be translated into its probable effect on the obtaining of aerodynamic data.

The quantities to be measured were thus specified as:-

a) fluctuating static pressure on the sidewall of the test-section

b) turbulence immediately upstream of the contraction and in the test-section

c) fluctuations of flow angle (both pitch and yaw) in the test-section

d) fluctuations of both pitot and static pressures in the test-section flow

These measurements were carried out in 3 wind-tunnels (Ludwieg tube (LT)), Evans Clean Tunnel (μECT), and injector driven tunnel (IDT, T_2) and supplemented by additional data from a smaller IDT (T_2').

Although these data have their genesis in the specific purpose of assisting an evaluation of three candidate drive systems for a particular application, the comprehensive sets of flow quality measurements thus obtained are of wider interest.

2 EXPERIMENTAL METHODS

2.1 Salient characteristics of the pilot wind-tunnels

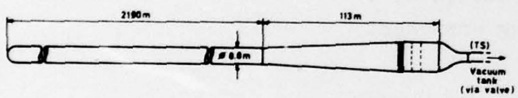

Fig 1 Ludwieg tube (LT)

The Ludwieg tube (LT) was constructed by modifying tube "A" of the Göttingen Ludwieg tube tunnel and operates on the principles described in Ref 3 and 4. Its airline is shown as Fig 1. Flow is initiated by a double diaphragm downstream of the diffuser and second throat - the latter being formed by a translating plug. The test-section has solid walls and is cylindrical with an internal diameter of 200mm and a length to diameter ratio of 3.0.

The flow is generated by a charge tube of length 80mm and a diameter of 399mm having about the same length to diameter ratio as the proposed design for LEHRT[4]. A 2.5° half-angle conical diffuser is attached to the downstream end of the charge tube having an exit cross-sectional area 3 times that of the charge tube. This is followed by a honeycomb, 2 wire screens, and an axisymmetric convergent nozzle (of contraction ratio 4 : 1) leading to the test-section.

The LT is designed for operation, with choked second throat, at Mach numbers between 0.4 and 1.0 and at stagnation pressures between 1.5 and 6.0 bars. Prior to the measurements of flow quality the facility was calibrated by measuring static pressures immediately upstream and downstream of the screens and in the test-section. Operation at test-section Mach number (M_∞) between 0.36 and 0.92 was demonstrated at stagnation pressures of 150kPa and 250kPa. The upper limit on Mach number was imposed by boundary layer growth in the cylindrical test-section and fell to 0.82 with introduction of the flow quality probes so that pitch and yaw angle data at the highest Mach number was obtained at conditions close to choking. The steady run time was approximately 0.4s thus demonstrating that the addition of the settling chamber did not significantly prolong the starting process as the theoretical run time[5] for instantaneous flow establishment is 0.44s.

Associated experiments of relevance to the quality of the flow produced by a Ludwieg tube have been reported by Grauer-Carstensen and concern measurements of the growth of boundary-layer and consequent pressure changes in the 399mm internal diameter (ID) charge and recovery tubes[6]. It was found that the edges of the boundary-layers moved at approximately constant rates towards the tube axes. At the downstream end of the charge tube, the edge of the boundary layer reached the tube axis at times (from initiation of flow) of 0.34s and 0.26s for charge tube Mach numbers of 0.2 and 0.3 respectively - the former corresponding to the LT used for the flow quality experiment. Corresponding pressure changes during the steady run time of this LT (at 250kPa stagnation pressure) were a 1% reduction in the stagnation pressure of flow through the test-section and a 7% rise in the pressure at the upstream end of the recovery tube. These observations are in good agreement with theory[7,8] which implies that much smaller effects are to be anticipated at the scale of LEHRT (5m x 4.2m test-section).

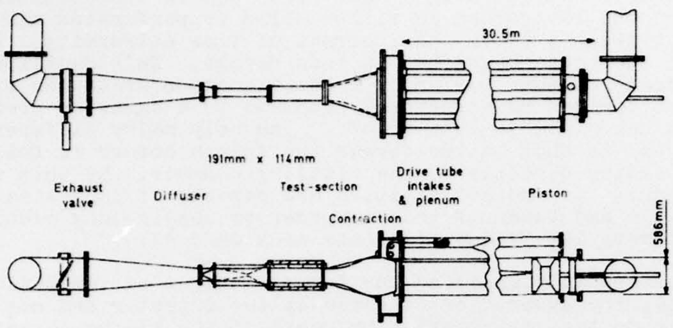

Fig 2 ECT pilot tunnel

The μECT was constructed at RAE (Bedford) following a successful preliminary demonstration of wave cancellation[9]. It is illustrated in Fig 2 and includes a test-section of rectangular cross-section with 10% open slotted roof and floor and solid sidewalls. Test-section width is fixed at 114mm and the height was 145mm giving a contraction ratio of 16.3 and a maximum piston speed of $15.9 ms^{-1}$.

The charge tube has a diameter of of 586mm and a length of 30.5. The 4 drive tubes each have a bore of 152mm. An adjus-adjustable second throat is located downstream of the test-section and is followed by a short, equal area, transition from rectangular to circular cross-section and a conical diffuser which ends at the exhaust valve. This valve is a 0.508m diameter, edge sealed, butterfly valve opened by a mechanical actuator in a time which is adjustable between 190ms and 400ms. The return circuit, whose internal diameter is 508mm, includes 4 mitred corners (without turning vanes) and brings the flow back to the upstream side of the main piston. This piston runs in the charge tube and is connected to smaller drive pistons by wire ropes. Air to feed the drive tubes is drawn from the pulley housing, which acts as a plenum chamber, via throttles formed by interchangeable orifice plates. The air in the pulley housing is replenished by air drawn from either the charge tube or the return circuit. For the tests reported here the feed to the drive tubes was taken from the charge tube and the throttles were wide open (corresponding to maximum pressure ratio).

Operation and performance of the μECT has been extensively described elsewhere[10,11] and it is sufficient to note here that changes in mean stagnation pressure during a run are confined to cyclic variations at frequencies of about 1.2Hz and 10.0Hz of magnitudes (at $M_\infty = 0.9$) of 0.1% rms and 0.25% rms respectively. The facility has been operated throughout its design envelope of stagnation pressure from 120kPa to 310kPa and Mach numbers between 0.30 and 1.10. Steady run time varies with Mach number, having a typical value of 1.45s at $M_\infty = 0.8$; but for the tests reported here fixed length records of about 1.0s duration were taken during the steady run.

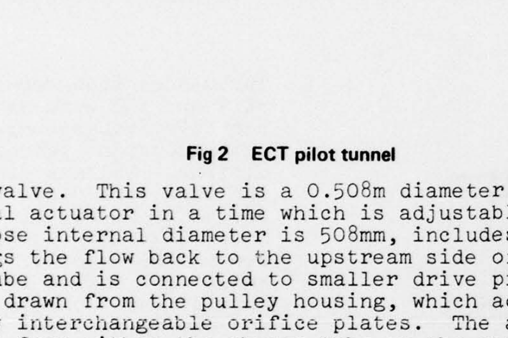

Fig 3 Injector driven tunnel (IDT): T_2

Two injector driven tunnels (IDT) have recently been built at ONERA/CERT in Toulouse. Although the basic principles of such a facility have been known for some time[12] and several examples have been built and used, the problem of designing very large facilities of this type was readdressed in a comprehensive fashion by Carriere[13]. This reappraisal introduced a novel form of combined turning vanes and injector and showed that a large IDT might be made to start in an acceptably short time. Validation of theory and optimisation of circuit layout were accomplished in a small pilot IDT known as T_2'. Subsequently, a larger facility, known as T_2, was constructed and commissioned.

The smaller IDT (T_2') has a solid-walled test-section of rectangular cross- section 100mm high by 100mm wide. The contraction ratio is 9.0 and the test-section is followed by the first diffuser in which much of the diffusion is accomplished by abstraction of air through the porous walls. The air thus discharged to atmosphere balances the inflow of air at the injector which is combined with the turning vanes of the first corner. The induced airflow enters the corner at a Mach number of up to 0.6 and the injected air issues from the base of the corner vanes at a Mach number of 1.6. A transition from rectangular to circular cross-section is combined with further diffusion between the first and second corners. A final diffuser is located downstream of the second corner whence the third and fourth corners bring the flow back to the settling chamber.

The larger IDT (T_2) has a similar layout (Fig 3) but has a square test-section of 400mm width and height[14]. The roof and floors can be either solid or perforated and their angle of divergence can be varied between $\pm \frac{1}{2}^{\circ}$ in the interest of flow uniformity. A solid roof and solid floor were used for the tests described in this report. This facility is designed for operation at Mach numbers between 0.3 and 1.1 at stagnation pressures from 200kPa to 500kPa; but operation was limited by structural problems to stagnation pressures below 250kPa for part of the tests described in this report. The only major difference between the airlines of T_2 and of T_2' is that in the former the fourth corner is followed by a rapid expansion rather than leading directly to the settling chamber. By this means a contraction ratio of 20 is obtained in T_2. Both facilities are capable of long steady run times (30s for T_2) and full advantage was taken of this in order to obtain long records so as to obtain accurate data even at very low frequencies (see section 2.4).

The T_2' facility has been extensively employed for investigations of rapid flow establishment, injector performance, the generation of noise at the injector and amelioration of the influence of this noise on the test-section by means of the second throat and acoustic lining of the return circuit. This work is reported in detail elsewhere[14,15].

2.2 Probe designs

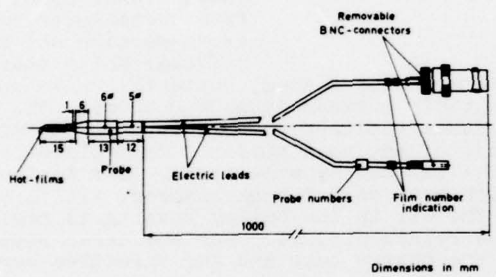

Fig 4 Hot-film probe (V-shaped)

Turbulence measurements in the μECT and IDT were made using a hot-film, wedge-shaped, V-probe under constant temperature conditions. Salient dimensions of the DISA type 55R72 probes used are shown in Fig 4. It will be noted that the two hot films lie on the edges of a wedge which is V shaped in plan. The apex of the V points into wind and has an included angle of 90°. Instantaneous velocities are deduced from the instantaneous power inputs necessary to maintain the probe at a constant temperature. These power inputs were provided by DSIA or TSI constant-temperature anemometer systems as used with most hot-wire or hot film probes. The probes were mounted on supports specially manufactured to suit each wind-tunnel, the tip of the probe being at least 90mm upstream of the leading edge of the support. The probes were provided by DFVLR-AVA.

In the μECT wind-tunnel the probes were mounted on the centre-lines of the facility in the test-section and just upstream of the contraction. Owing to difficulties described in section 2.5, the measurements in the IDT (T_2') test-section were made with the nose of the probe 55mm ahead of the centre-line of a 6mm diameter cylindrical support and 20mm from the test-section wall (ie the distance between the wall and the tip of the probe was 40% of the half-width of the test-section). However, this support was small and the probe was well outside the sidewall boundary-layer so that no significant differences were anticipated on account of this change in mounting arrangement.

Probe characteristics may be summarised as follows:-

TABLE 1

Characteristics of V-Probes

Probe	User/Calibrator	Resistance of film (Ω) at 20°C	Resistances of lead (Ω)	Temperature coefficient of resistance (% per °C)
No 1	ONERA/NLR	(10.26 (9.51	1.40 1.40	0.44 0.43
No 2	RAE/NLR	(8.78 (10.30	1.40 1.40	0.46 0.43
No 3	ONERA/ONERA	(14.71 (12.92	1.00 1.00	0.32 0.33

Hot wire measurements were made by ONERA in the settling chambers of T_2' and T_2 in order to supplement the hot-film data.

In the LT facility the use of hot-film probes was not successful, seemingly due to their thermal response characteristics, and hot-wire probes were used instead. With these, tests were performed at various radial positions in the settling chamber just upstream of the contraction. Although it had been anticipated that hot-wire probes would not withstand the aerodynamic loads within transonic test-sections - which was the reason for supplying hot film probes - some tests were made in the test-section with especially designed hot-wire probes, but failed.

Measurements of static pressure on the sidewalls of the test-sections of the LT and both IDT T_2' and T_2 were made using Kulite transducers having a very high natural frequency (circa 70kHz). In the μECT frequency response was sacrificed in favour of increased sensitivity by using a NEP type 1040 differential pressure transducer with the reference pipe connected to an adjacent static tapping via a long length of 1.5mm internal diameter, pvc tubing. This arrangement confined measurements to the frequency range of 6Hz to 8Hz; but gave a sensitivity of 0.52v/kPa as compared to the 300μV/kPa typically obtained from the Kulite transducers.

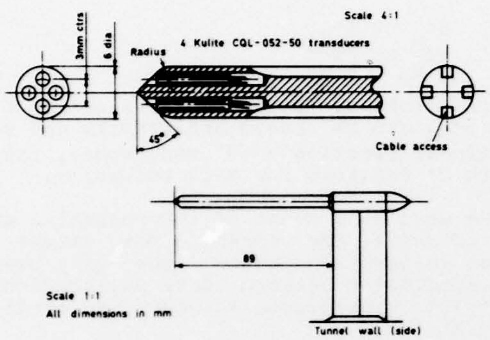

Fig 5 Yaw meter

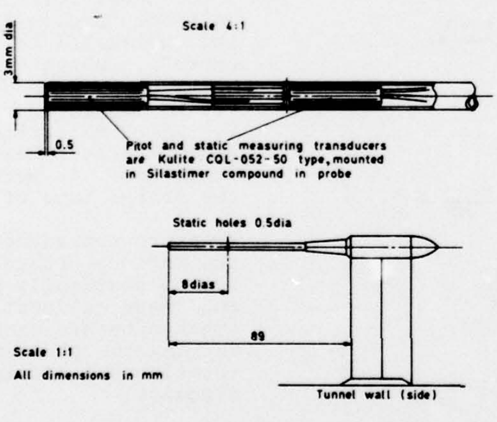

Fig 6 Pitot static probe

Fluctuations in flow angle, pitot pressure, and free-stream static pressure were measured by probes manufactured at the RAE and illustrated in Figs 5 and 6. These are adaptations of types of probe commonly used for measurements of the mean flow; internally mounted transducers of high frequency response being substituted for tubing to remote instrumentation. The Kulite transducers used (type CQL-52-50) had sensitivities around 290μV/kPa and resonant frequencies of 70kHz. The dimensions of the yawmeter were kept small so that individual transducer readings could be combined to give flow angles without fear of errors due to phase differences due to the finite time for disturbances to pass across the probe. For example, an acoustic disturbance takes only 18μs to travel a distance equal to the diameter of the yawmeter. A 90° phase error between readings of the transducers, which were at 3mm centres, was not likely to occur, therefore, until a frequency of about 28kHz. Likewise, the fundamental frequency of standing waves in the inlet pipes to all transducers was above this frequency which represents the approximate upper limit to valid data from these probes. The Kulite transducers used employ a strain gauge bridge of 750Ω overall resistance. The required 5v excitation voltage was obtained from a 20v dc power supply using a 2250Ω resistor in series with each transducer. This conformed to the manufacturer's recommendations; the change of sensitivity being thereby reduced to 0.45% per 100°C. For the tests in the LT the excitation voltage was obtained from a stabilised 5v dc power supply.

2.3 Calibration of probes

The output of hot films are very sensitive to temperature variations and can vary with differences in the equipment associated with their use (power supplies etc). Thus, a single calibration could not be used for all tests. Instead, probes 1 and 2 were calibrated at NLR over the range $43 < \rho U < 490$ (ρU in kg $m^{-2}s^{-1}$) in order to check the correct functioning of the probes, to establish the alignments of the two films to the axis of the probe, and to determine a "universal calibration curve". For the latter purpose, the recovery factor (r) was first determined as r = 0.87. Since the temperature of the sensor (T_s) could be obtained from the film resistance, this enabled measured output voltages (E) to be corrected for variations in temperature, the corrected voltage E* being given by

$$\frac{E^*}{E} = \sqrt{\frac{T_s - T_{o_{ref}}}{T_s - T_r}}$$

where $T_{o_{ref}}$ is a reference stagnation temperature, say that of the first measuring point.

The measured temporal mean velocity (U) was corrected for density changes to form a corrected velocity U* given by

$$U^* = \frac{\rho U}{\rho atm}$$

Corrected voltages plotted as a function of corrected speeds then formed a calibration curve. To allow for variations in ancillary equipment, this calibration curve $E^* = f(U^*)$ was used to form a "universal calibration curve" $E_u^* = f(U^*)$ by the linear transformation:-

$$\frac{E^* - E^*_1}{E^*_2 - E^*_1} = \frac{E^*_u - E^*_{u_1}}{E^*_{u_2} - E^*_{u_2}}$$

in which suffices 1 and 2 denote calibration points at low and high values of U* respectively. The corresponding values $E^*_{u_1}$ and $U^*_{u_2}$ can be chosen arbitrarily and were fixed at values which made E^*_u an approximately linear function of U* and, hence, facilitated the evaluation of the derivatives of E* with U* required for data reduction.

The "universal calibration curve" was used as a means of interpolating data obtained during each series of tests. For each set of tests, the corrected mean values of E* at a high and a low mean value of U* were used as calibration points. The "universal calibration curve" was then used as a means of interpolating between these calibration curves to obtain U* and the necessary derivative ($\partial E^*/\partial U^*$) appropriate to each test condition from the mean value of E* measured during that test.

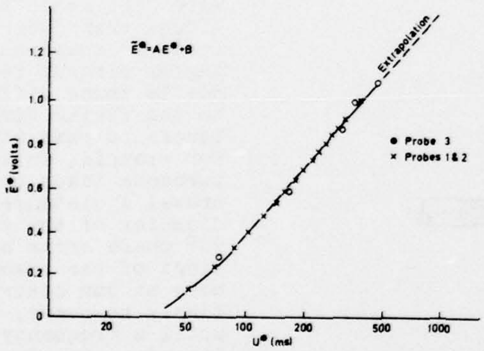

Fig 7 Universal calibration curve for hot film probes

Following the loss of probe No 2 and the substitution of probe No 3, a similar calibration process was followed by ONERA for the latter probe. Fig 7 shows how well data from probe 3 was correlated with data from probes 1 and 2 using the "universal calibration curve". Subsequently the calibration of probe No 2 was extended to values of ρU as low as 8.4 kg $m^{-2}s^{-1}$ using the RAE Boundary-Layer Research Tunnel preparatory to measurements in the charge tube of the μECT.

Pressure transducers used in the various probes were calibrated statically prior to use and these calibrations were checked before each run by varying the pressure of the (stationary) air in the tunnel circuits.

The pitot-static and yawmeter probes were checked in a precursor series of tests in the μECT in order to confirm that their steady aerodynamic characteristics were typical of the conventional probes from which they were derived, and, hence, to gain confidence in the measurements made with them. These tests were not intended as exhaustive aerodynamic calibrations; but rather sought assurance concerning possible specific problems. For example, it was postulated that high supervelocities around the lip of the pitot probe might, at high subsonic speeds, give rise to regions of supersonic flow terminated by a shockwave. The resulting disturbed flow could invalidate data obtained from the static pressure tappings. To this end, measurements were made of the mean values of pitot and static pressure indicated by the probe as well as stagnation pressure and pressure in the plenum chamber obtained from the fixed instrumentation of the μECT.Mach numbers were obtained from the probe data and from the tunnel instrumentation. The differences between these 2 Mach numbers are shown as a function of M_∞ in Fig 8. It is probable that some of the variation shown in this figure is due to variation of the pressure difference across the slotted liners. Thus, Fig 8 is by no means an absolute calibration of the pitot static probe. What is does show is that there is no abrupt change in the difference between probe and tunnel instrumentation data at Mach numbers below 0.8. The variation for M > 0.8 is probably due to the development of local regions of transonic flow. It was concluded that

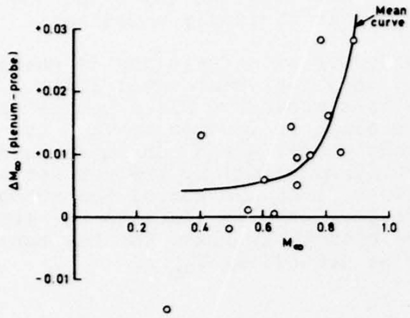

Fig 8 Comparison of data from pitot static probe and from fixed tunnel instrumentation (μECT)

errors in static pressure measured by the pitot static probe were likely to be small at least for $M < 0.80$. In the event, data obtained with this probe did not exhibit any marked difference in pressure fluctuations for $M_\infty < 0.8$ and $M_\infty > 0.8$. Hence, effects of supercritical flow on the measurements seem to have been small at speeds. The primary question concerning the yawmeter was whether its sensitivity would be materially altered by having the pressure transducers so close to the surface of the probe. The local flow angle is computed by forming a pressure coefficient from the difference in the pressures measured by each pair of transducers and the free-stream dynamic pressure. This pressure coefficient is translated into a flow angle using the probe sensitivity ie the rate of change of this pressure coefficient with angular displacement of the probe. The sensitivity was determined at $M_\infty = 0.7$ by performing runs with the probe mounted at various angles of incidence to the free-stream. These tests yielded a sensitivity of 0.0447 per degree which, being in reasonable accord with the sensitivity of 0.040 per degree quoted in Ref 16 for a similar probe at $M \simeq 0$, was used throughout all the data reduction.

2.4 Methods of data reduction

The concept of analysing any complex periodic signal into a series of sinusoidals, known as its Fourier components, is too well known to require reiteration. However, it is useful to rehearse the findamental ideas involved in the extension of this principle to aperiodic signals - such as those encountered in the present work.

Consider an idealised apparatus in which a signal is processed by a number of filters in parallel. These filters are ideal bandpass filters with their centre-frequencies f and bandwidths Δf chosen that were the input a pure sinusoid it would, whatever its frequency, pass through one, but only one, of the filters. The outputs from the filters are each measured by a meter recording its root-mean-square (rms) value. All practical means of Fourier analysis are attempts to reproduce the function - if not the form - of this idealised apparatus.

If a pure sine-wave is input to the apparatus a null reading will be seen on all the output channels except one - which will indicate the rms value of the input signal. The centre frequency of this channel will provide an estimate of frequency of the input signal. If the number of filters (and meters) is increased, while still maintaining a complete, but non-redundant, coverage of the frequency range, the result will not change except that a better estimate of the input signal frequency will be obtained.

Now, consider input of a periodic, but non-sinusoidal, signal. Non-zero readings will now be observed on several meters. These readings will be the rms value of the Fourier components of the input signal and the centre frequencies of the corresponding filters provide estimates of the frequencies of these components. The result of such a test can be represented graphically by a histogram giving rms values for each centre frequency. If, as before, the number of filters is increased the result is unchanged in that the same set of rms readings will be observed but the reduced bandwidth of the corresponding filters will imply greater certainty in the estimates of the frequencies. Thus, apart from the resolution on the frequency scale, the histogram representation of the results is independant of the bandwidth used in the filtering. Since any individual Fourier component will pass through one, but only one, filter the input signal power is equal to the sum of the powers of the outputs ie if the rms of the input is V_i and the rms of the outputs are V_{o1}, V_{o2}, ... V_{om}, then:-

$$V_i = \sqrt{\sum_{m=1}^{m=n} V_{o_m}^2}$$

So far only strictly periodic signals have been considered. Any aperiodicity involves a fundamental change in the underlying mathematics[17]. Application of Fourier analysis to an aperiodic signal yields Fourier components at an infinite number of frequencies. Hence, application of an aperiodic input to the idealised apparatus will produce non-zero readings on all of the output meters. The equality of the input signal power and the sum of the output powers is unaffected by the aperiodicity of the input since this equality depends only on the complete, but non-overlapping, coverage of the frequency range by the filters. The result of the test can be represented by a histogram of output readings as a function of frequency; but such a presentation is no longer meaningful as may be seen by considering the effects of increasing the number of filters. With an increased number of filters, non-zero readings will be recorded on everyone of the meters despite their number having been increased. As the sum of the output signal powers must remain constant, the individual readings will be decreased in magnitude due to the increased number of output readings. Thus, the presentation of the results is now dependant upon the number of filters ie upon details of the method of analysis. This is obviously undesirable.

We now presume that the distribution of signal power with frequency is reasonably flat - leaving precise definition of what is meant by this statement until later. Consider now the case in which one of the filters in the idealised apparatus has a centre frequency f and a bandwidth Δf. The corresponding output rms reading is denoted by V_o. Let this filter be replaced by a number of filters each of bandwidth $\Delta f/_b$ and centre-frequencies such that the range Δf is completely covered (without overlapping) by the new set of filters. It is evident, from the preceeding discussions, that the sum of the output powers is unchanged by the substitution of the new set of filters for the old. As the distribution of output power with frequency is supposed to be reasonably flat, the new output readings can be taken to be equal to each other and can be denoted by V_{o_b}

Then:-

$$V_o^2 = b\, V_{o_b}^2$$

or

$$V_{o_p} = \frac{V_o}{\sqrt{b}}$$

This immediately suggests the use of a variable formed by dividing the meter readings by the square root of the bandwidth of the associated filter. In practice, integration of the data is facilitated by using the relative bandwidth (ie $\varepsilon = \Delta f/f$) rather than the absolute value Δf. In this case, the output variable at frequency f becomes $V_o/\sqrt{\varepsilon}$ with the wide bandwidth filter and $\sqrt{b}\, V_{o_b}/\sqrt{\varepsilon}$ with the set of narrow bandwidth filters. Since $V_{o_b}\, \sqrt{b} = V_o$ it is clear that the new variable is independent of Δf and a histogram of its variation with frequency is a useful way of displaying the output data.

All of the varieties of Fourier analysis equipment used in the analyses of the data described in this report are attempts to realise an idealised arrangement as described above. In addition, the bandwidth is made sufficiently small that it is smaller than the expected bandwidth of the sharpest peak in the spectrum (thus elucidating what was meant earlier by the term "reasonably flat"). The measured rms values then provide a close approximation to a continuous spectrum. However, even excluding the possibility that the bandwidths used are insufficiently small in relation to the peaks in the spectra (this does not seem to have been troublesome except where spurious mechanical or acousrical effects were involved), a choice remains as to the precise value of bandwidth to be used. A narrow bandwidth gives high precision in the definition of the frequencies; but, for a given duration of signal to be analysed, implies large, and unavoidable, statistical uncertainties in the measured amplitudes. Thus, a trade-off exists between record length, accuracy of amplitude measurement, and discrimination of frequencies. The actual values used in the various experiments are listed below:-

Institute	Frequency range (kH_z)	Bandwidth (H_z)	Record length (s)	Standard deviation of measured amplitudes
DFVLR	0 to 2.0 0 to 10.0	4 20	0.40 0.40	79% 35%
ONERA	0 to 0.10 0.01 to 2.0 0.20 to 40.0	0.5 10.0 200.0	20.0 1.00 0.05	32% 32% 32%
RAE	0.10 to 10.0 0.20 to 20.0	6% of centre f 156.0	1.00 1.00	41% to 4% 8%

The standard deviation shown in the above table is the statistical error inherent in the method of analysis. This was improved upon in most cases by analysis of several records thus effectively extending the length of record available for analysis. It does not include errors due to noise on the signals etc.

Analysis at the DFVLR was performed by a Nicolet Scientific Corporation Spectrum analyser type UA500A using signals recorded on a magnetic tape recorder and replayed at 1/8th of the recording speed. A similar type of recording system (and speed step-down) was used in the analysis performed at RAE which, however, used a Hewlett-Packard type 5451A Fourier analyser.

At DFVLR, differential signals such as from pairs of transducers in the yawmeter were derived during the run using differential amplifiers and were recorded in parallel with the individual transducer outputs. At RAE, only the individual transducer signals were recorded and the differencing was performed by replaying these signals into a differential amplifier. The input channel gains of this amplifier were adjusted so as to obtain null output when replaying a calibration signal which had been supplied simultaneously to all recorder channels immediately prior to each run - thus compensating for differences in recorder gains. The ONERA method of analysis was centred on a Hewlett-Packard 2100 mini-computer. This technique is fully described in Ref 18 which includes a listing of the fast Fourier transform program used.

2.5 Special experimental problems and their resolution

Two mechanical failures of probes were encountered. In the first of these a hot-film probe being used for a test in T_2' at $M_\infty = 0.8$ developed a split in the probe itself and the leads to one film were broken. This was attributed to mechanical vibration of the probe. This probe was replaced by one available at ONERA which, upon calibration, was found to have a very similar characteristic to that possessed by the original probe prior to its loss. The tests were continued successfully with a shortened probe support which reduced probe vibration at the cost of moving the probe away from the tunnel centre-line. Data for the IDT facilities are thus for the turbulence probe mounted 20mm away from a tunnel side-wall rather than on the centre-line of the test-section. In the second failure, a Kulite transducer in the pitot-static probe failed during tests in the LT. The transducer was replaced and the probe recalibrated prior to subsequent tests.

All the facilities were designed to produce a very low level of flow disturbances. Consequently, it was sometimes difficult to analyse the test data since the very low signals were contaminated by electrical noise. This problem was especially severe in the case of tests in the μECT using the pitot static probe and was attributed to the operation of heavy electrical plant near the test site. This was resolved using records taken immediately prior to the test (without air flow over the probe) and by quadratic differencing of "wind-off" and "wind-on" measurements. In order to obtain the requisite accuracy of amplitude measurements it was necessary, on this occasion alone, to use relative bandwidths of up to $\Delta f/f = 0.20$ thus sacrificing much frequency resolution for accuracy in amplitude. As a final check overall rms measurements were made for the range $0.1 < n < 1.0$ and for $\Delta f = 0.5$ kHz at $n > 1.0$. Although quite obscuring any details of the spectrum, this enabled accurate rms values to be obtained so that the aerodynamic contribution to these could be evaluated with confidence. Comparison between the two sets of data thus obtained is shown in Fig 9. The fair measure of agreement between mean levels indicates that the corrections applied to the narrow band data were reasonably successful.

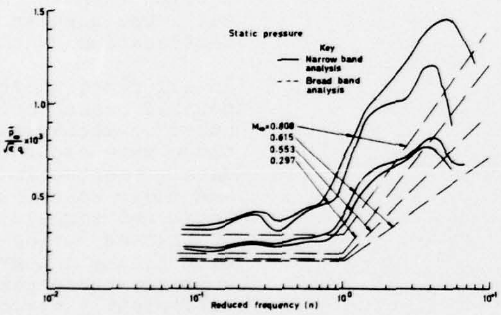

Fig 9 Comparison of data obtained by two methods: pitot static probe, μECT

At DFVLR a signal to noise ratio was achieved which a clear separation of signal and noise up to frequencies of 15kHz.

In several cases, difficulties were encountered with the appearance of large, sharp peaks in the measured spectra. An example from the LT is given in Fig 10 which shows a typical pressure fluctuation spectrum obtained from a pressure probe situated on the axis of the circular test section. The position of the obvious peaks depends on test section diameter and Mach number. A plot of frequencies of peaks as a function of Mach number as given in Fig 11 indicates that there are two different sources for such intensity peaks. The peaks for which frequency increases with Mach number can be related to the shedding of von Karman vortex streets from the probe support.

The others, for which frequency decreases slightly with increasing Mach number, were found to correspond to cylindrical acoustic waves travelling upstream through the test section. Closer investigation of this phenomenon shows that this second group of peaks are not representing discrete resonance frequencies but that they are the edges of frequency bands.

It seems that the acoustic waves are generated by the local turbulent wall boundary layer. Theoretical considerations[19] suggest that - in the near vicinity of each n-th peak - the intensity is proportional to

$$\frac{1}{\sqrt{\left(\frac{\omega}{a}\right)^2 - (1 - M^2) l_n^2}} \, .$$

It can clearly be seen that the given fluctuation intensity has peaks for

$$\left(\frac{\omega}{a}\right)^2 - (1 - M^2) l_n^2 = 0$$

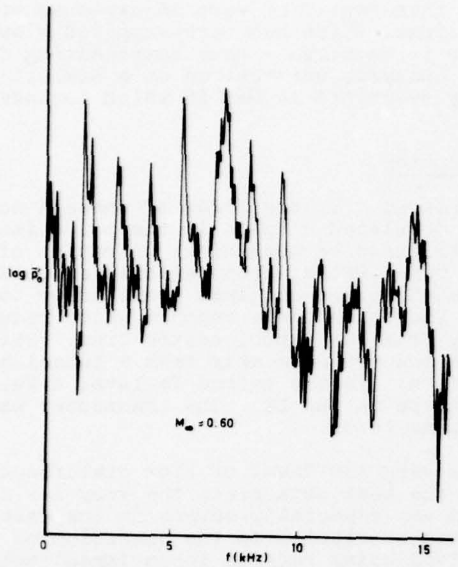

Fig 10 Fluctuation of pitot pressure in test section of LT

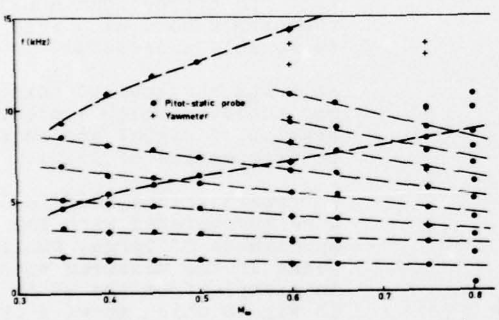

Fig 11 Frequencies of peaks in measured spectra from pressure probes: Ludwieg tube

where l_n is given by

$$J_o'(l_n \cdot R) = 0$$

A comparison of theoretically predicted peak frequencies with actual experimental results is given in Fig 12. The decrease of amplitude with higher frequencies as indicated by the above relation can also be recognized in the test results, and it follows that other effects contributing to the measured total fluctuation intensity are completely obscured by these acoustic waves.

It seems that conditions for development of acoustic waves are unusually favourable in the special LT test section in which they were observed first. As, however, the origin of these waves is independent of the drive system, occurrence of acoustic waves is not specific for LT tunnels and should be regarded for any type of wind tunnel. As shown in Fig 13, on which is plotted the frequency of peaks in the turbulence spectra measured in the μECT, most of such frequencies found in this facility were independant of the test-section Mach number and, accordingly, were attributed to mechanical vibrations of the probe and its support. The remaining peaks, ie those whose frequencies increased with M_∞, were attributed to genuine aerodynamic effects. It may be that some of these had a similar acoustical cause to the peaks found in the LT. However, this is unlikely as the analysis of Ref 19 indicates that such effects are much more pronounced in an axisymmetric, closed test-section than in the slotted-wall, rectangular cross-section, test-section of the μECT.

In all cases where sharp pronounced peaks could be attributed to mechanical resonances, these were excluded from the data presented in this report and their contributions to the total rms signals have been abstracted during the derivation of any integrated values, wherever these peaks were of sufficiently narrow bandwidth to suggest that the associated disturbances were confined to the immediate vicinity of the peaks and, hence, did not obscure the neighboroughing signals. Unfortunately, this was not the case with the acoustic peaks observed in the signals of pitot and static pressures recorded in the LT test-section discussed above. Results of these measurements have to be omitted because the signals are totally obscured by these broad-banded acoustic waves.

3 EXPERIMENTAL RESULTS

Typical results obtained in the various facilities are presented in this section, with little comment. Analysis is reserved for section 4.

3.1 Acoustic disturbances

As described earlier, fluctuations in static pressure were measured on the sidewalls of the test-sections and on the centre-lines of the tunnels using a pitot-static probe. Figs 14, 15 and 16 present measurements of fluctuating sidewall static pressures for the LT, μECT, and T_2 respectively. Corresponding data from the pitot-static probe are given in Figs 17 and 18.

3.2 Turbulence

Data were obtained for lateral and longitudinal velocity fluctuations both upstream

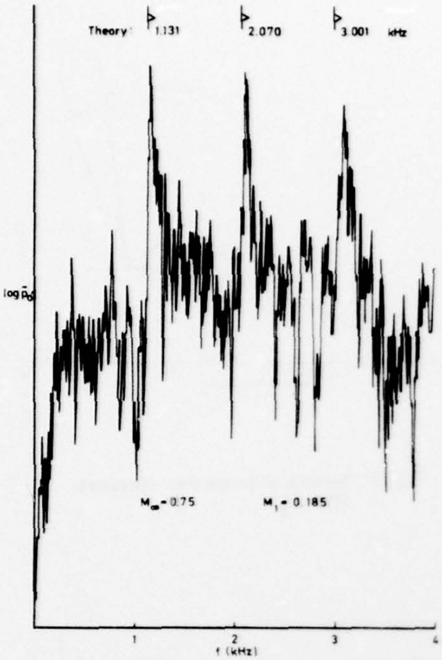

Fig 12 Theoretically predicted peak frequencies and fluctuation spectrum from charge tube of LT

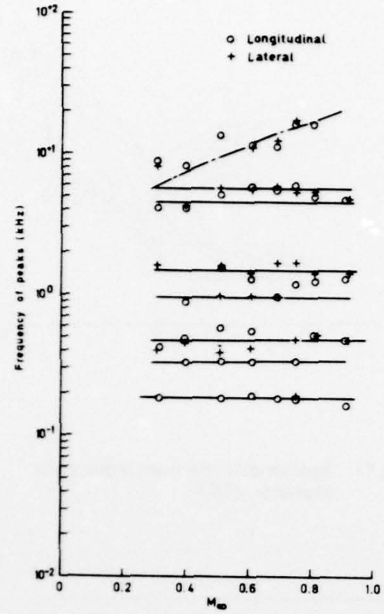

Fig 13 Frequencies of peaks in measured turbulence spectra: μECT

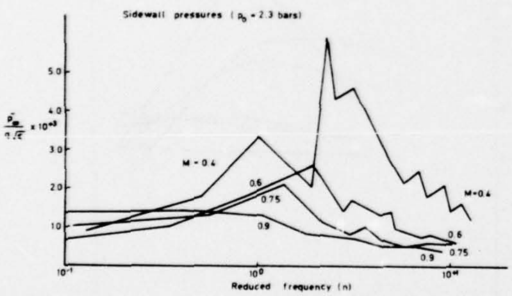

Fig 14 Fluctuations in sidewall static pressure: Ludwieg tube

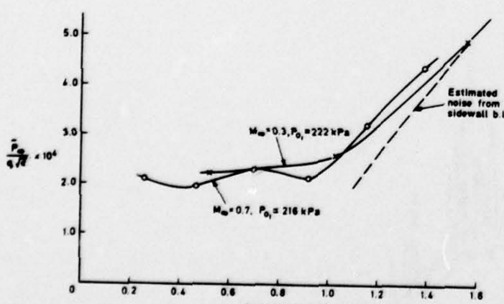

Fig 15 Frequency spectra of sidewall static pressures: μECT

of the contraction and in the test-section. These are given in Figs 19 to 27 inclusive.

3.3 Flow angle fluctuations

Spectra of flow angle fluctuations, both pitch (α) and yaw (β) from the 3 facilities are presented in Figs 28 to 33 inclusive.

3.4 Pitot pressure fluctuations

Spectra of this quantity are presented in Figs 34 and 35.

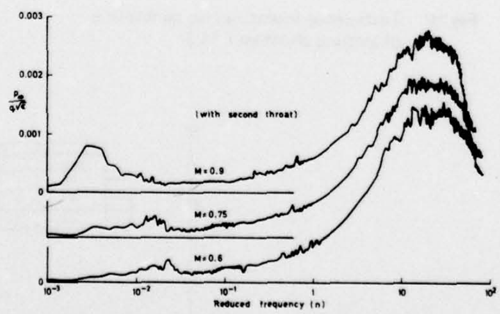

Fig 16 Sidewall static pressure fluctuations: T_2 (IDT)

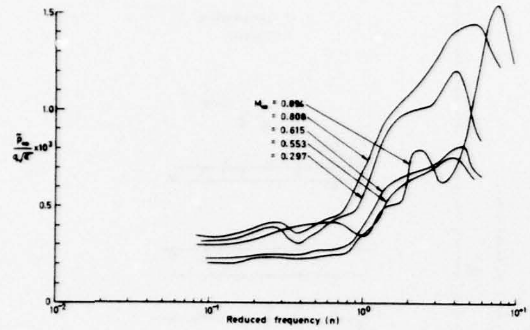

Fig 17 Spectra obtained from probe static pressure: μECT

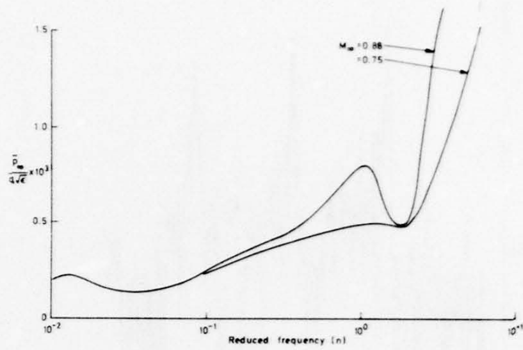

Fig 18 Spectra of probe static pressure: IDT T_2

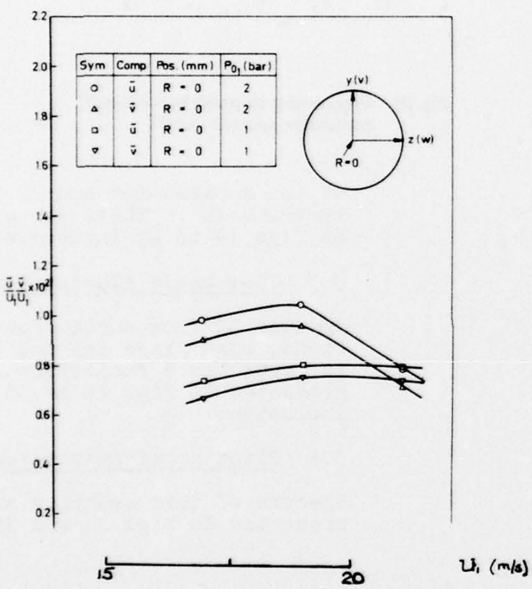

Fig 19 Turbulence intensities on centre-line of settling chamber of LT

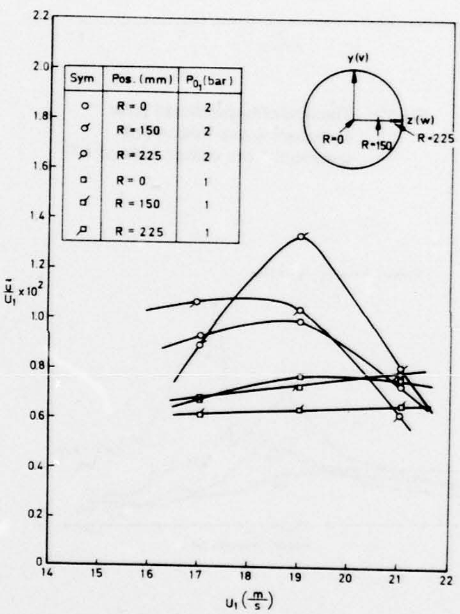

Fig 20 Turbulence intensity as function of location within settling chamber of LT

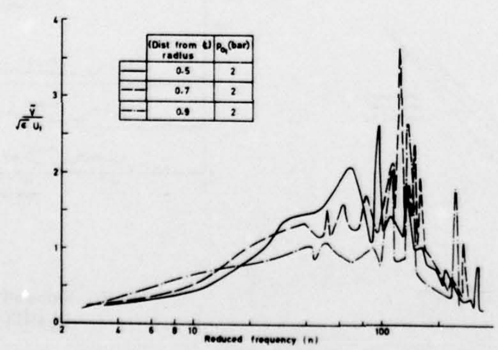

Fig 21 Lateral turbulence in settling chamber of LT

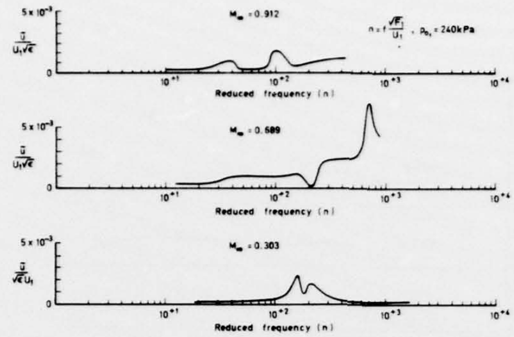

Fig 22 Longitudinal turbulence component immediately upstream of contraction: μECT

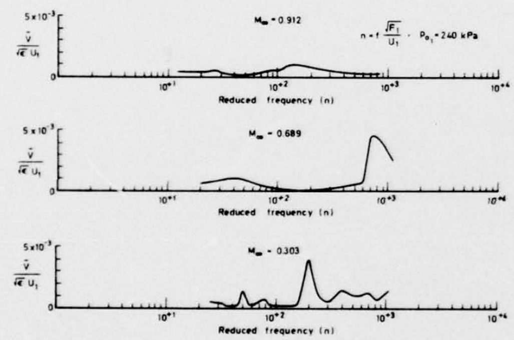

Fig 23 Lateral turbulence component immediately upstream of contraction: μECT

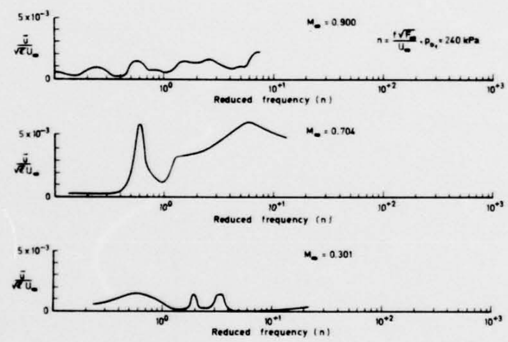

Fig 24 Longitudinal turbulence component in test section: μECT

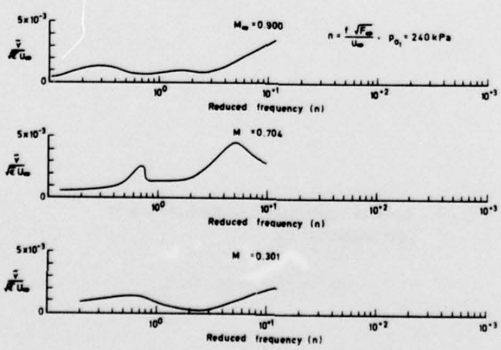

Fig 25 Lateral turbulence component in test section: μECT

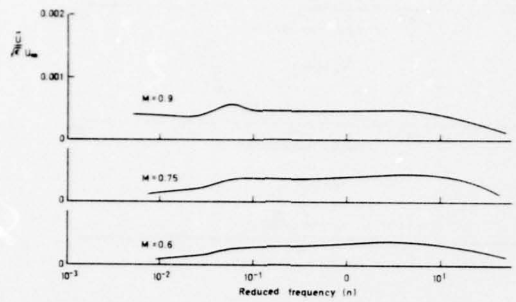

Fig 26 Longitudinal component of turbulence in test section: T_2

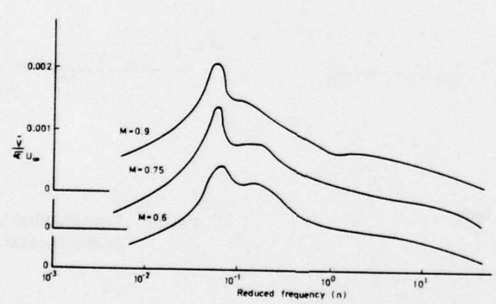

Fig 27 Lateral component of turbulence in test section: T_2

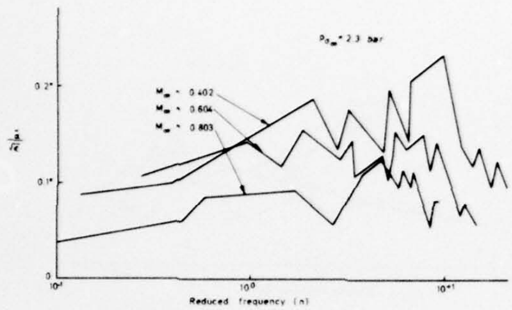

Fig 28 Spectra of fluctuating pitch angle: LT

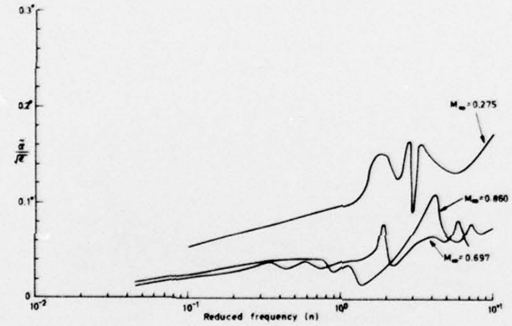

Fig 29 Typical measured spectra of pitch angle: μECT

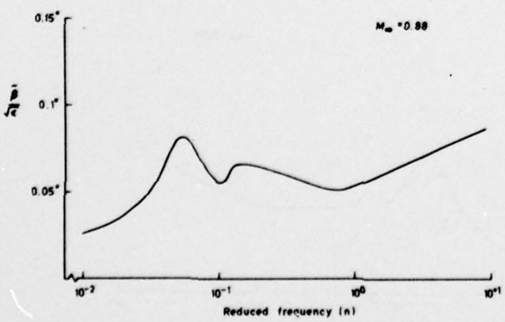

Fig 30 Frequency spectrum of fluctuating pitch angle: IDT (T_2)

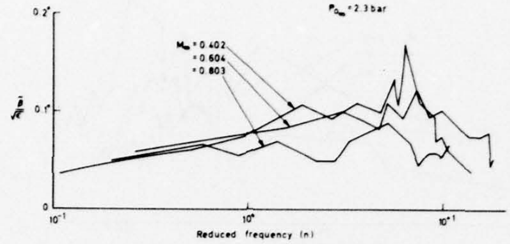

Fig 31 Spectra of fluctuating yaw angle: LT

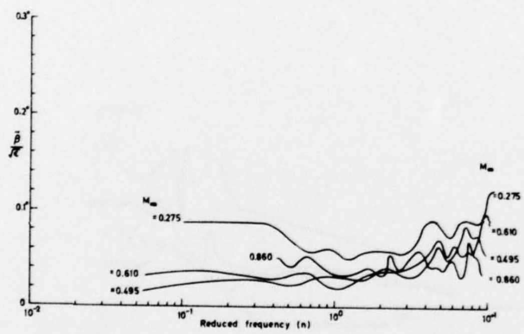

Fig 32 Spectra of fluctuations of yaw angle: μECT

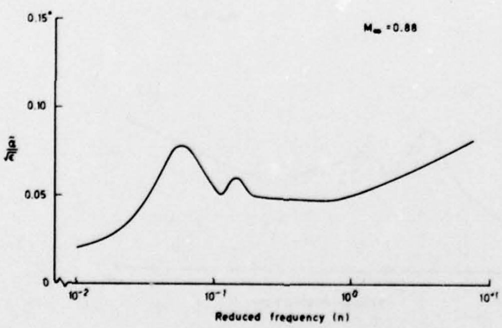

Fig 33 Spectrum of fluctuating yaw angle: IDT (T_2)

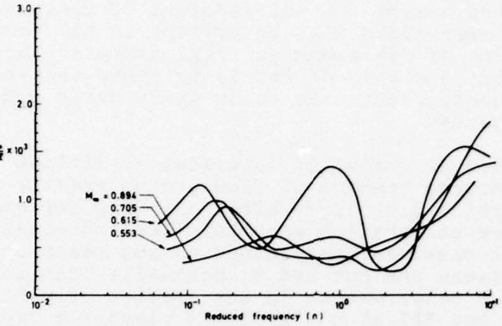

Fig 34 Spectra obtained from pitot pressure: μECT

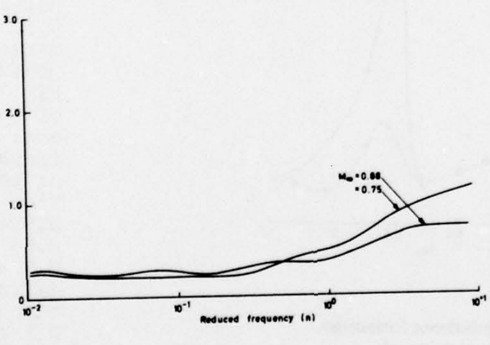

Fig 35 Spectra obtained from pitot pressure: IDT (T_2)

4 DISCUSSION OF RESULTS

4.1 Influence of drive-system, second throat and test-section on flow quality

While obtaining the data reported in the previous section, a number of interesting observations and additional tests were made. These are reported here when they either shed light on the functioning of the individual types of wind-tunnel or illustrate points of importance to wind-tunnel activities in general.

Perhaps the most interesting of such observations to come out of the work on the LT was the discovery of the severity of flow disturbances associated with acoustic resonances in the cylindrical, solid-walled, test-section of that facility. While it is believed that these effects are far less severe in test-sections of rectangular cross-section and/or ventilated walls, similar mechanisms must be present in all transonic wind-tunnels and provide means for amplification of disturbances originating at the downstream end of the test-section. Extension of the analysis of Ref 19 to other test-section configurations and means of suppressing acoustic resonances, such as by using compliant walls[20], appear to be worthwhile objects of study.

Analysis of runs repeated at nominally identical conditions on the μECT revealed a run to run scatter in the measured spectra of flow angles greatly in excess of the statistical errors discussed in section 2.4. Even after allowing for other measurement inaccuracies, including transducer calibration uncertainties and electrical noise, a run to run variation in measurements remained unexplained by any means other than a genuine variation in flow conditions between one run and a, nominally identical, successor. The standard deviation of repeated measurements in the spectra of pitch fluctuations was 24% at n = 0.46, 23% at n = 0.92 and 38% at n = 1.38. A plausible explanation is that a major source of flow unsteadiness in this facility is the turbulence generated during filling of the charge tube. In the μECT filling of the charge tube with air was accomplished by admitting air through a stub pipe immediately upstream of the exhaust valve. Clearly, considerable turbulence is likely to have been generated as this air flowed through the test-section and contraction on its way to the charge tube. The intensity of this turbulence would depend upon the rate of filling and would not entirely decay to zero during the short interval between completion of filling the charge tube and initiating a run. Neither the rate of filling nor the time between its completion and a run were carefully controlled - thus accounting for the observed variability of flow quality. There could be merit, during the design of another facility of this type, in arranging for the admission of air to the charge tube in such a way as to minimise the turbulence generated thereby.

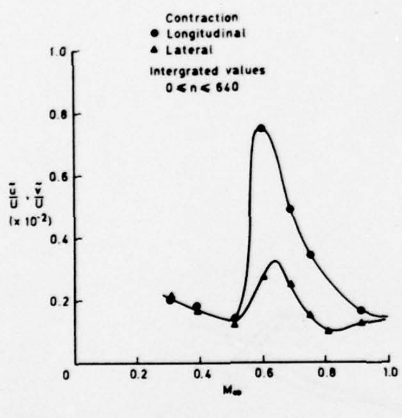

Fig 36 Integrated turbulence intensities immediately upstream of contraction: μECT

Throughout the data gathered from the μECT there is a tendency for flow unsteadiness to be larger at very low and intermediate Mach numbers ($M_\infty \simeq 0.7$ and $M_\infty \simeq 0.3$) than at the highest or low Mach numbers ($M_\infty \simeq 0.9$ and $M_\infty \simeq 0.5$). This tendency is most marked in the turbulence measurements as illustrated in Fig 36 which shows rms values of both longitudinal and lateral turbulence components (measured immediately upstream of the contraction) as a function of test-section Mach number. The maxima at $M_\infty \simeq 0.65$ and the minima at $M_\infty \simeq 0.5$ and $M_\infty \simeq 0.9$ are well defined. Comparison of the turbulence spectra for $M_\infty \simeq 0.7$ and $M_\infty \simeq 0.90$ reveals that the maxima are primarily due to substantial contributions in the range of reduced frequencies (based on local conditions) of from 300 to 700. This range corresponds to reduced frequencies based on test-section conditions of about 3.3 to 7.8. Increases in unsteadiness as Mach number is reduced from circa 0.9 can also be seen in the static pressure, pitch angle, and turbulence measurements in the test section and are largest (in absolute terms) for reduced frequencies greater than 1.0. However, the data does not allow of any more precise identification of the range of reduced frequencies in these cases and there is little, or no, increase in unsteadiness manifest in the pitot pressure or yaw angle data as M_∞ decreases from 0.9 to circa 0.7 although pitot pressure fluctuations are higher for $n \simeq 1$ at $M_\infty = 0.615$ (Fig 34).

The reason for these effects cannot be stated with certainty; but two pieces of evidence are persuasive indications that the increased flow unsteadiness is associated with the slotted roof and floor of the test-section and the use of diffuser suction. Firstly, the turbulence intensities measured in the test-section were generally higher than those measured just upstream of the contraction. This suggests the presence of a source in the test-section rather than further upstream especially as turbulence originating upstream of the contraction would be attenuated by its passage through this section. Secondly, the reduced frequencies (based on test-section conditions) of 3.3 to 7.8, quoted earlier, correspond to reduced frequencies based on free-stream test-section velocity and slot width of 0.029 to 0.069 which range is compatible with the fundamental and overtone excitation frequencies found by Mabey[2] to be associated with unsteady flow originating at the downstream end of slotted liners incorporating diffuser suction. In his case the fundamental frequency corresponded to a reduced frequency, based on slot width, of 0.035.

As to the variation of this additional unsteadiness with Mach number, Hartzuiker[21] has pointed out that the most likely origin of pressure fluctuations associated with the slotted liners is at the downstream end of the liners. He postulates that these waves propogate upstream and are partially reflected back to the end of the slots (from the entrance to the settling chamber) and these prompt the generation of new disturbances. This prop agation is assumed to be of a spherical type so that attenuation varies roughly quadratically with time. The time for a pressure pulse to make the round trip is:-

$$t = \frac{2\ell}{(1-M_\infty{}^2)M_\infty a_\infty}$$

(where ℓ is the distance from the end of the slots to the contraction) which, with attenuation proportional to t^2 gives:-

M_∞	0.2	0.4	0.6	0.8	1.0
Relative amplitude	0.25	0.77	1.0	0.56	0.0

This distribution is broadly similar to the observed variation of flow unsteadiness with the Mach number in the test-section and, hence, suggests that the basic mechanism is of the type postulated by Hartzuiker. Transmission of the disturbances through the contraction would then account for the variation shown in Fig 36.

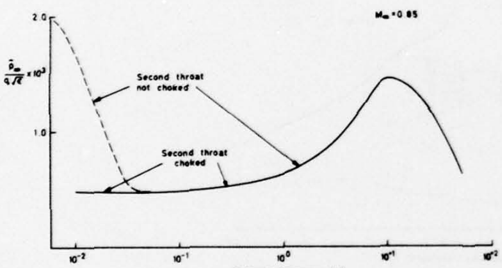

Fig 37 Influence of second throat on sidewall static pressure fluctuations: IDT (T_2)

In the injector driven tunnels a number of interesting additional experiments have been conducted. The influence of a second throat was investigated in T_2 by comparing data (such as that presented earlier) obtained with a choked second throat with data obtained without this throat in use. Static pressure fluctuations measured on the test-section sidewall at M = 0.85 with and without the second throat in use are shown in Fig 37. The large reduction in low-frequency fluctuations effected by the second-throat amply demonstrates the effectiveness of such devices in blocking off the shortest path by which disturbances from the diffuser can reach the test-section.

Turbulence measurements in both T_2 and T_2' are unlikely to have been confused by substantial sources of turbulence within the test-sections as these have solid walls. Comparisons, such as that shown in Fig 38, of turbulence measured upstream of the contraction and in the test-section are, thus, of interest as they show the influence of the contraction upon the turbulence of flow passing through it - unobscured by other effects. The large reduction in longitudinal turbulence is demonstrated. It will be noted that turbulence in the test-section of T_2' is isotropic. This suggests that the differing influences of the contraction upon longitudinal and lateral components of turbulence - thus producing anisotropy - are quickly cancelled out by redistribution of turbulence energy between the various modes once the flow enters the test-section. The turbulence in the test-section of T_2 is not isotropic; but this is a manifestation of another effect, discussion of which follows.

Differences between turbulence levels in T_2 and T_2' were investigated with particular attention to the flow in the settling chambers. Surveys of the temporal mean flow in the settling chambers of these tunnels had revealed that the more compact layout of rapid diffuser at the upstream end of the settling chamber in T_2 resulted in flow separations from the walls of this component. The resulting effect on turbulence levels is illustrated in Fig 39 which shows spectra of the ratio of lateral to longitudinal components of turbulence within the test-sections of T_2 and T_2'. Whereas turbulence in T_2' (when in its normal configuration) is essentially isotropic, the separations at the entrance to the settling

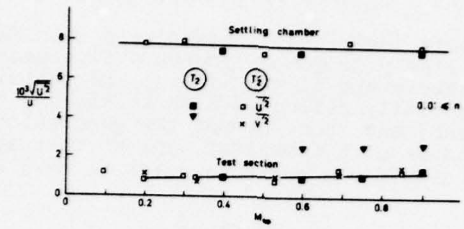

Fig 38 Turbulence intensities in T_2 and T_2'

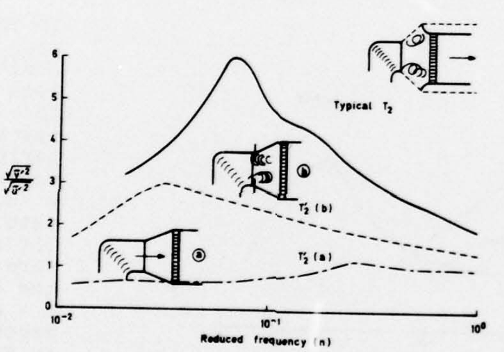

Fig 39 Influence of the entrance into the settling chamber on velocity fluctuations (T_2 and T_2')

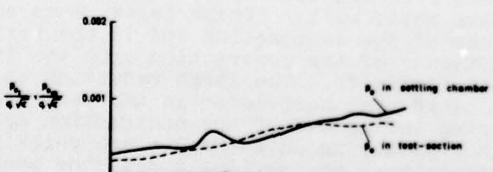

Fig 40 Pitot pressure in settling chamber and test section of T_2

chamber of T_2 promoted large amounts of lateral turbulence resulting in non-isotropic turbulence with a predominance of lateral components. Attribution of this result to the influence of the separations in the rapid diffuser was confirmed by the introduction of spoilers into the diffuser of T_2' thus forcing a separation to occur and, as indicated by the intermediate curve in Fig 39, producing in the test-section an isotropy of the turbulence of a similar character - albeit of smaller magnitude - to that found in T_2.

Finally, Fig 40 illustrates the isentropic nature of the flow through the contraction by showing excellent correspondence between spectra of pitot pressure fluctuation: measured in the settling chamber and in the test-section of T_2.

4.2 Relationships between acoustic, turbulence, flow angle and pitot pressure fluctuations

A complete investigation of this topic would require numerous cross-correlations to be evaluated. These were not included in most of the data obtained. However, some interesting lessons can be learned from the data available. This is summarised in the table below in which mean values over the range $0.1 \leqslant n \leqslant 1.0$ of the various fluctuating quantities are given for the Mach number closest to 0.9 for which data is available. Attention is concentrated upon these reduced frequencies and Mach numbers because these are of the greatest interest to data gathering and the probable users of transonic wind-tunnels[1]. Also, the range $0.1 \leqslant n \leqslant 1.0$ is one in which the quality of the flow in the test-section is normally determined by the drive system rather than by details of the design of the test-section or by noise radiated from the turbulent boundary-layers on its walls.

$M_\infty \simeq 0.9$

Mean levels ($0.1 \leqslant n \leqslant 1.0$)							
Facility	$\tilde{p}_\infty/q\sqrt{\varepsilon}$ walls	probe	$\frac{\tilde{p}_o}{q\sqrt{\varepsilon}}$	$\frac{\tilde{\alpha}}{\sqrt{\varepsilon}}$	$\frac{\tilde{\beta}}{\sqrt{\varepsilon}}$	$\frac{\tilde{u}}{U\sqrt{\varepsilon}}$	$\frac{\tilde{v}}{U\sqrt{\varepsilon}}$
LT	1.4×10^{-3}	-	-	1.2×10^{-3}	0.8×10^{-3}	-	-
μECT	0.2×10^{-3}	0.3×10^{-3}	0.4×10^{-3}	0.5×10^{-3}	0.8×10^{-3}	0.7×10^{-3}	0.9×10^{-3}
T_2	0.4×10^{-3}	0.6×10^{-3}	0.3×10^{-3}	0.8×10^{-3}	1.0×10^{-3}	0.4×10^{-3}	1.0×10^{-3}

($\tilde{\alpha}$ and $\tilde{\beta}$ in radians)

One question to be answered is whether the use of fluctuating wall static pressure measurements as an index of flow quality remains valid even when dealing with such disprate types of drive systems. This can be examined by normalising each measure of flow unsteadiness by the value of fluctuating wall static pressure measured in the same facility. This gives:-

Levels relative to $\tilde{p}_\infty/q\sqrt{\varepsilon}$ (measured on wall)							
Facility	$\frac{\tilde{p}_\infty}{q\sqrt{\varepsilon}}$ walls	probe	$\frac{p_o}{q\sqrt{\varepsilon}}$	$\tilde{\alpha}/\sqrt{\varepsilon}$	$\tilde{\beta}/\sqrt{\varepsilon}$	$\frac{\tilde{u}}{U\sqrt{\varepsilon}}$	$\frac{\tilde{v}}{U\sqrt{\varepsilon}}$
LT	1.0	-	-	0.8	0.6	-	-
μECT	1.0	1.5	2.0	2.5	4.0	3.5	4.5
T_2	1.0	1.5	0.8	2.0	2.5	1.0	2.5

The table above shows that there is very little correspondence between variations (between facilities) in fluctuating wall static pressure levels and any other quantity except for fluctuating static pressures measured with a probe. However convenient such measurements may be to make, it is clear that unsteadiness of wall static pressures should not be used as a general index of flow quality when dealing with wind-tunnels having differing types of drive system or, possibly, even widely different types of test-section. This may also be seen by comparing Figs 28 and 31 with Figs 29 and 32 together with a comparison of, say, Figs 14 and 15. Such comparisons show how two facilities having very different levels of unsteadiness of flow static pressure have similar levels of fluctuations in pitch and yaw angles.

If fluctuating static pressure alone cannot be used as an index of flow quality then it is necessary to enquire what additional parameters, should be considered. While the set of measurements described in this report give a virtually complete description of the quality of the flow in any wind-tunnel, they are probably too time consuming for routine use; but the supplementing of static pressure fluctuations by some other simple, but more reliable, index of flow quality would be desirable. The merits of supplementing static pressure fluctuations by some other quantity is demonstrated if the significance of flow quality for data gathering is considered further.

For the usual types of wind-tunnel measurement (forces, pressures etc)[1] it is clear that the wavelengths of pressure fluctuations within the range of interest ($0.1 \leqslant n \leqslant 1.0$) are so large compared to the dimensions of typical models that unsteadiness in static pressure is unlikely to have an important direct effect on measured steady (integrated) quantities. Fluctuations in pitot pressure will have a direct, and approximately proportional, effect on the data being gathered. However, the typically 0.02% to 0.05% variations in pitot pressure are unlikely to be of practical significance as changes of this order in measured pressures and forces are usually undetectable with normal instrumentation except within narrow frequency bands associated with resonances of balances etc. However, changes in flow angles will also have a direct, and approximately proportional, effect on the data. Further, when it is recalled that significant changes in measured forces may be caused by variations in incidence or sideslip of a fraction of a degree, it is clear that fluctuations in pitch or yaw of 0.05^{o} are of practical significance.

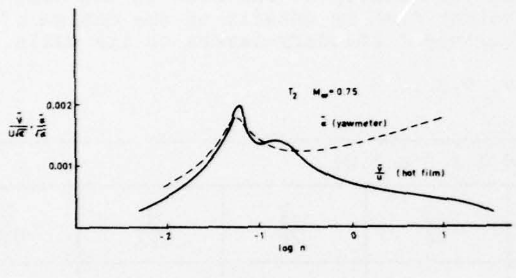

Fig 41 Comparison between $\tilde{\alpha}$ and $\tilde{v}$ measurements (T_2)

Accordingly, it would seem that changes in flow angle are the prime cause of fluctuations in the data gathered during normal aerodynamic testing. The work reported here has shown that it is unnecessary to infer the unsteadiness in flow angle from other measurements. Rather, this may be measured directly. The tables above indicate that, as would be expected, it is not of utmost significance whether the general level of fluctuations in flow angle are measured by a suitable yawmeter or by measuring lateral components of velocity. Within reasonable experimental accuracy, broadly the same results are obtained. This is further illustrated in Fig 41 which compares spectra of yawmeter and lateral turbulence measurements made in T_2. Good agreement was obtained at the lower frequencies; but the agreement is poor at somewhat higher frequencies. Choice of whether to make yawmeter or turbulence measurements can depend on the instrumentation available and the frequency range of interest in any particular application. In larger facilities the demands on frequency response of pressure measurements will be less severe (for given reduced frequency) and measurements of unsteady pressures is becoming an increasingly common requirement. Hence, yawmeters may then be the more convenient type of instrument. However, resolution of the reasons for the different results obtained at high frequencies by the yawmeter and the hot film methods is desirable.

5 CONCLUSIONS

Although initially conceived as a means of comparing three drive systems then under consideration as options for the Large European High Reynolds number wind-tunnel, the work described in this report has yielded several conclusions of importance in a much wider context.

Firstly, it has been demonstrated that it is possible to obtain a much more complete description of the unsteady components of the flow within the test-section of a transonic wind-tunnel than has hitherto been customary. What can be accomplished in small wind-tunnels of novel design can obviously be repeated with comparative ease within established facilities.

Secondly, this report provides ample evidence of the need for scrupulous attention to details of the design of the working section and adjacent components if the quality of the flow provided by a good drive system is not to be degraded by flow unsteadiness generated in the test-section. Examples illustrated in this report include acoustic resonances in closed axisymmetric test-sections, unsteadiness associated with the use of imperfectly designed means of applying diffuser suction to slotted walls, the effectiveness of using a second throat to prevent noise from the diffuser reaching the test-section, and the effects of flow separations at the entrance to the settling chamber.

Finally, it has been shown that the use of fluctuating static pressure as an index of flow quality is invalid when comparing wind-tunnels having different forms of drive system or, possibly, even widely different types of test-section. Fluctuations in flow angle are of much more direct consequence to the gathering of the usual types of data and can be measured using either appropriately designed yawmeters or with hot-film probes. Both of these methods give comparable results at low frequencies and for overall levels of fluctuations for $0.1 \leqslant n \leqslant 1.0$ and the choice between them depends upon the frequency range of interest, and the instrumentation available in any individual application. However, clarification of the causes of the different results obtained by these two techniques at high frequencies is desirable.

6 ACKNOWLEDGEMENTS

The work reported herein would not have been possible without the enthusiastic help of many people. Principal amongst these were Mr A Elsenaar who calibrated the hot-film probes and investigated their applications at transonic speeds in detail and Mr G F Lee who designed the pitot-static and yawmeter probes. Also, Mr T G Gell performed the static and aerodynamic calibration of the latter probes and was responsible for the analysis of data obtained in the μECT; while Mr W A Beckett conducted the actual testing in this facility. Mr D G Mabey was a ready source of valuable advice at several stages in the work.

At ONERA M A Mignosi and M A Seraudie conducted the testing in T_2 and T_2' and took a great part in the analysis of the results.

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DEVELOPMENT OF THE CRYOGENIC TUNNEL CONCEPT AND APPLICATION TO THE U.S. NATIONAL TRANSONIC FACILITY

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SUMMARY

Based on theoretical studies and experience with a low-speed fan-driven tunnel and with a pressurized transonic tunnel, the cryogenic wind-tunnel concept has been shown to offer many advantages with respect to the attainment of full-scale Reynolds number at reasonable levels of dynamic pressure in a ground-based facility. The unique modes of operation available in a pressurized cryogenic tunnel make possible for the first time the separation of Mach number, Reynolds number, and aeroelastic effects. By reducing the drive-power requirements to a level where a conventional fan-drive system may be used, the cryogenic concept makes possible a tunnel with high productivity and run times sufficiently long to allow for all types of tests at reduced capital costs and, for equal amounts of testing, reduced total energy consumption in comparison with other tunnel concepts.

A new fan-driven high Reynolds number transonic cryogenic tunnel is now under construction in the United States at the NASA Langley Research Center. The tunnel, to be known as the National Transonic Facility (NTF), will have a 2.5 by 2.5-m test section and will be capable of operating from ambient to cryogenic temperatures at stagnation pressures up to 8.8 atm. By taking full advantage of the cryogenic concept, the NTF will provide an order of magnitude increase in Reynolds number capability over existing tunnels in the United States.

LIST OF SYMBOLS

a	Speed of sound
c	Chord of two-dimensional airfoil
$\bar{c}$	Mean geometric chord
C_D	Drag coefficient, $\frac{\text{Drag}}{qS}$
C_L	Lift coefficient, $\frac{\text{Lift}}{qS}$
C_m	Pitching-moment coefficient, $\frac{\text{Pitching moment}}{qS\bar{c}}$
C_p	Pressure coefficient, $\frac{p - p_\infty}{q_\infty}$, or specific heat at constant pressure
d_m	Maximum diameter of model
E	Modulus of elasticity
ℓ	Linear dimension of model or test section
h	Enthalpy
M	Mach number
p	Pressure
q	Dynamic pressure, $1/2\ \rho V^2$
R	Reynolds number, $\rho V \ell/\mu$
$\mathcal{R}$	Universal gas constant
S	Reference wing area
T	Temperature
$\bar{T}_t$	Mean value of total temperature
v	Specific volume
V	Velocity
x	Linear dimension
Z	Compressibility factor
α	Angle of incidence
γ	Ratio of specific heats
σ	Standard deviation
μ	Viscosity
ρ	Density

Subscripts

β	Boattail
max	Maximum
min	Minimum
t	Stagnation conditions
1	Upstream
2	Downstream
∞	Free stream

*Head, Advanced Experimental Methods Group, Subsonic-Transonic Aerodynamics Division.

1. INTRODUCTION

The present interest in the development of transports and maneuvering aircraft to operate efficiently in the transonic range has resulted in a review of the problems of flow simulation in transonic wind tunnels. Among the more serious problems is that related to inadequate test Reynolds number. It is this problem and an attractive solution to the problem that is the subject of this paper.

The need for increased testing capability in terms of Reynolds number has been well documented.[1,2] A major problem is that of scale effect on boundary-layer shock-wave interaction. Not only is aircraft performance affected by the interaction, but other characteristics such as loads and load distribution, stability, buffeting, and maneuverability are also affected.

The possibility of simulating high values of Reynolds number in existing wind tunnels has not been overlooked, and many devices for causing boundary-layer transition are in use in the various aeronautical laboratories. However, fixing the location of boundary-layer transition is in itself seldom sufficient to duplicate the flow associated with full-scale Reynolds number.[3]

Recognition of such problems has resulted in a consensus, both in the United States and in Europe, that there is an urgent need for wind tunnels having greatly increased Reynolds number capability. The need is especially acute at transonic speeds where, because of the large power requirements of transonic tunnels, economic forces have dictated the use of relatively small tunnels. With ever increasing aircraft size, the existing transonic tunnels are becoming even more inadequate in test Reynolds number capability.

The inadequate Reynolds number capability of existing tunnels was the underlying theme of many of the papers presented at the AGARD fluid dynamics panel specialists' meeting held at Göttingen, Germany, in 1971. One of the fundamental difficulties is to define the level of Reynolds number which is required for valid transonic testing, and this was discussed in several papers during that meeting.[4,5] As a result of these studies, several proposals[3,6,7] were made at the Göttingen meeting for new wind-tunnel facilities which would be capable of providing a Reynolds number at transonic speeds of at least 40×10^6, based on the mean geometric chord of a typical swept-wing aircraft, which is considerably greater than existing wind-tunnel capability.

At a given Mach number, the Reynolds number may be increased by using a heavy gas rather than air as the test gas, by increasing the size of the tunnel and model, by increasing the operating pressure of the tunnel, and by reducing the test temperature. The method chosen to increase Reynolds number will, in general, also affect dynamic pressure, mass flow rate, and the energy consumption of the tunnel for a given amount of testing. The use of a heavy gas is a well-known method of achieving high Reynolds number. Freon-12 is one of the most suitable of the heavy gases for use in a wind tunnel.[8] However, the differences in the ratio of specific heats become important when compressibility effects become significant, thus making Freon-12 a questionable transonic test medium.[9] An obvious way to increase Reynolds number is to increase the model size. In order to avoid increasing the wall interference effects, however, there must be a commensurate increase in tunnel size. Design studies for tunnels large enough to give full-scale Reynolds number at normal temperatures and moderate pressures show that they would be very large, and therefore very costly, and would make heavy demands on power.[3] An alternate solution is to restrict the tunnel and model sizes and increase the operating pressure. This method is feasible, of course, but the aerodynamic forces on the model, balance, and support system are greatly increased at the operating pressures that are required to achieve the desired Reynolds number. From a power standpoint, a high-pressure tunnel is preferable to a large, moderate-pressure tunnel. However, for the required increase in Reynolds number, the power requirements are still undesirably large.

Operating a tunnel at cryogenic temperatures, first proposed by Smelt[10] in 1945 offers an attractive means of increasing Reynolds number while avoiding many of the practical problems associated with testing at high Reynolds numbers in conventional pressure tunnels. Personnel of the NASA Langley Research Center have been studying the application of the cryogenic concept to various types of high Reynolds number transonic tunnels since the autumn of 1971. The results of a theoretical investigation aimed at extending the analysis of Smelt and the results of an experimental program using a low-speed wind tunnel have been reported in References 11 and 12. In order to provide information required for the planning of a large high Reynolds number transonic cryogenic tunnel, a relatively small pressurized transonic cryogenic tunnel was built and placed into operation in 1973. As a result of the successful operation of the pilot transonic tunnel, it was classified by NASA in late 1974 as a research facility, re-named the 0.3-meter Transonic Cryogenic Tunnel, and is now being used for aerodynamic research as well as cryogenic wind-tunnel technology studies.

In addition to reviewing the cryogenic concept, this paper presents some details of the design and operation of the 0.3-meter Transonic Cryogenic Tunnel and describes some of the research done in the tunnel related to validation of the cryogenic wind-tunnel concept.

Also presented are the future plans for the 0.3-meter tunnel as well as some of the design and performance characteristics of a fan-driven high Reynolds number transonic cryogenic tunnel now under construction at the NASA Langley Research Center. This new tunnel, to be known as the National Transonic Facility (NTF), will take full advantage of the cryogenic concept to provide an order of magnitude increase in Reynolds number capability over existing tunnels in the United States.

2. THE CRYOGENIC CONCEPT

The use of low temperatures in wind tunnels was proposed by Smelt[10] as a means of reducing tunnel drive-power requirements at constant values of test Mach number, Reynolds number, and stagnation pressure. Reynolds number, which, of course, is the ratio of the inertia force to the viscous force, is given by

$$R = \frac{\text{Inertia force}}{\text{Viscous force}} = \frac{\rho V^2 \ell^2}{\mu V \ell}$$

which reduces to the well-known equation

$$R = \frac{\rho V \ell}{\mu} = \frac{\rho M a \ell}{\mu}$$

As the temperature is decreased, the density ρ increases and the viscosity μ decreases. As can be seen from the above equations, both of these changes result in increased Reynolds number. With decreasing temperature, the speed of sound a decreases. For a given Mach number, this reduction in the speed of sound results in a reduced velocity V which, while offsetting to some extent the Reynolds number increase due to the changes in ρ and μ, provides advantages with respect to dynamic pressure, drive power, and energy consumption.

It is informative to examine the underlying mechanism through which changes in pressure and temperature influence Reynolds number. To the first order μ and a are not functions of pressure while ρ is directly proportional to pressure. Thus, increasing pressure produces an increase in Reynolds number by increasing the inertia force with a commensurate increase in model, balance, and sting loads. Also, to the first order, $\rho \propto T^{-1}$, $V \propto T^{0.5}$, and $\mu \propto T^{0.9}$. Thus, decreasing temperature leaves the inertia force unchanged at a given Mach number due to the compensating effects of ρ and V^2. The increase in Reynolds number with decreasing temperature is thus due strictly to the large reduction in the viscous force term as a result of the changes in μ and V with temperature.

The effect of a reduction in temperature on the gas properties, test conditions, and drive power are illustrated in Figure 1. For comparison purposes, a stagnation temperature of 322 K (+120° F) for normal ambient temperature tunnels is assumed as a datum. It can be seen that an increase in Reynolds number by more than a factor of 6 is obtained with no increase in dynamic pressure and with a large reduction in the required drive power. To obtain such an increase in Reynolds number without increasing either the tunnel size or the operating pressure while actually reducing the drive power* is extremely attractive and makes the cryogenic approach to a high Reynolds number transonic tunnel much more desirable than previous approaches.

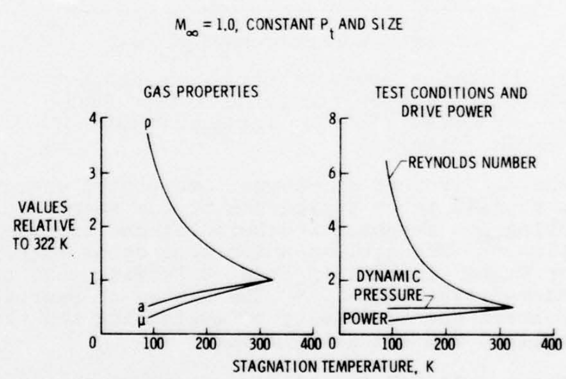

Figure 1. Effect of temperature reduction on gas properties, test conditions, and drive power. $M_\infty = 1.0$; constant stagnation pressure and tunnel size.

2.1 The Advantages of a Cryogenic Tunnel

2.1.1 Reduced Dynamic Pressure and Drive Power.

One of the most important test conditions which must be considered in the design of any wind tunnel is the dynamic pressure. The magnitude of the loads imposed on the model increases directly with dynamic pressure and so can cause such problems as increased model stress, which can seriously limit the achievable test lift coefficient for transport aircraft and prevent the accurate modeling of maneuver devices for fighter aircraft; more massive stings, which can produce flow distortion as well as require large departures from true aircraft-afterbody shapes; excessive balance loads; and nonrepresentative aeroelastic effects.

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*The tunnel drive power is shown in Figure 1 varying as $\sqrt{T}$ which is strictly true only for an ideal gas. Real-gas calculation of the work done in isentropic compressions at a given pressure and pressure ratio show drive power to decrease somewhat more rapidly than the simple $\sqrt{T}$ variation. In the calculation of drive power, it is important to use real-gas rather than ideal-gas properties. In particular, one must use the change in real-gas enthalpies, Δh, rather than $C_p \, \Delta T$ and the real-gas isentropic expansion coefficients[13] rather than the ideal-gas specific heat ratios.

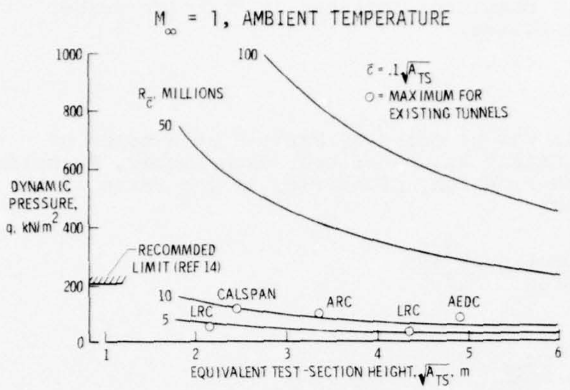


Figure 2. Dynamic pressure as a function of tunnel size for various values of Reynolds number.

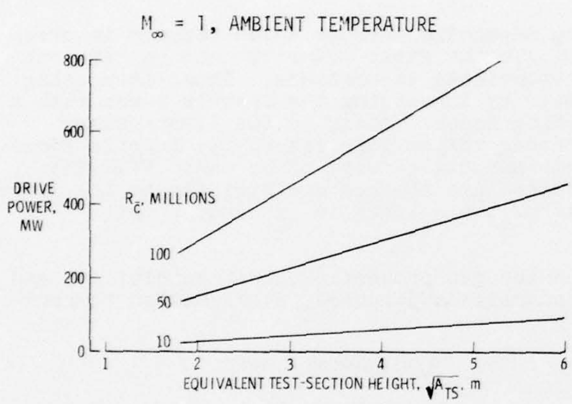


Figure 3. Drive power required for ambient temperature fan-driven tunnels as a function of tunnel size for various values of Reynolds number.

Figure 2 shows the variation in dynamic pressure with test-section size for various test Reynolds numbers† for ambient temperature tunnels at a Mach number of 1. The test-section size determines the model size and, hence, the tunnel stagnation pressure needed to produce a given Reynolds number and, since dynamic pressure is directly proportional to stagnation pressure, the test dynamic pressure is set.

The 5- and 10-million Reynolds number curves shown in Figure 2 are generally representative of the maximum values currently available in transonic tunnels in the U.S. These tunnels have maximum dynamic pressures of the order of 110 kN/m^2 (2300 lb/ft^2) or less. To get a test Reynolds number of 50 million in a 5.5-m test section at ambient temperatures would result in a dynamic pressure of about 555 kN/m^2 (11,600 lb/ft^2). Although the usable dynamic pressure depends upon the configuration and type of test, it is generally accepted that values greater than about 200 kN/m^2 (4200 lb/ft^2) would seriously limit the test capability for advanced transport aircraft.[14] For ambient temperature tunnels, then, the size of the test section would have to be greater than 5.5 m in order to achieve a test Reynolds number of 50 million at acceptable levels of dynamic pressure. Such a tunnel would be very costly due to both the shell cost and the cost of the drive system. The drive power required for ambient temperature tunnels is shown in Figure 3. As can be seen, a drive power of over 400 MW (more than a half-million horsepower) would be required to achieve the test Reynolds number of 50 million for a tunnel with a 5.5-m test section.

Such extreme power requirements have in the recent past forced designers of transonic tunnels to abandon the conventional ambient-temperature fan-driven tunnels in favor of short run time stored energy concepts. In Europe, the LaWs working group studied three approaches to the stored-energy intermittent tunnel with the goal of selecting one approach and building one transonic tunnel to serve the whole of European aviation.[15] The options which were being considered for the Large European High Reynolds Number Tunnel (LEHRT) included a Ludwieg tube tunnel,[16] a piston-driven tunnel,[17] and an injector-driven tunnel.[18] The number of approaches that were given serious consideration underscores the difficulty of evaluating the numerous trade-offs among test capabilities and capital and operating costs.

For a selected tunnel size and Reynolds number, the previously described effects of cryogenic operation are manifested in large reductions in the required tunnel stagnation pressure and therefore in large reductions in both the dynamic pressure and the drive power. These reductions are illustrated in Figure 4, where both dynamic pressure and drive power are shown as functions of stagnation temperature for a constant chord Reynolds number of 50 x 10^6 at M_∞ = 1.0 for a tunnel having a 2.5- by 2.5-m test section. As the tunnel operating temperature is reduced, the large reductions in both dynamic pressure and drive power provide the desired relief from the extremely high values that would be required for a pressure tunnel operating at normal temperatures.

The large reduction in drive power makes a fan-driven tunnel practical even at this high Reynolds number. The resulting efficiency and increased run time provide important advantages relative to intermittent tunnels, such as increased productivity, improved dynamic testing capability, and, as will be shown, reduced total energy consumption (and therefore operating costs) for equal amounts of testing.

An additional advantage of a fan-driven tunnel is realized by having run times sufficient to insure the avoidance of problems caused by heat transfer between the model and

†Unless otherwise noted, for consistency throughout this paper, Reynolds number is based on a wing chord equal to 0.1 times the square root of the test-section area; for wings of small aspect ratio, the actual values may be two or three times the value given.

the stream. In tunnels where the flow is generated by expansion waves, spurious scale effects due to heat transfer can only be avoided by cooling the model to the expected recovery temperature.[19] Such problems do not exist in a continuous-flow tunnel where the model is never far from thermal equilibrium with the stream. In general, no additional testing time is required in a fan-driven tunnel to allow the model to achieve thermal equilibrium since the flow initiation process is gradual and test conditions are not changed abruptly.

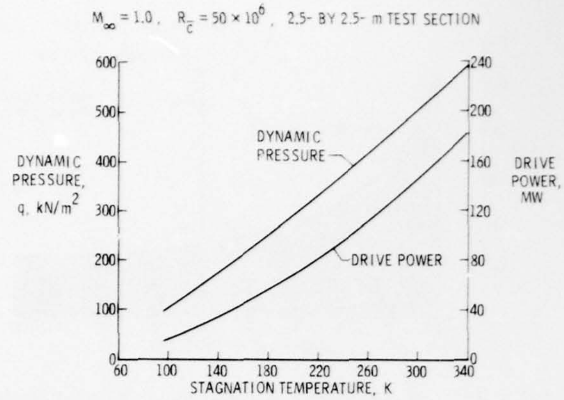


Figure 4. Effect of temperature reduction on dynamic pressure and drive power.

2.1.2 Reduced Capital Costs.

In Figure 4, the advantages of the cryogenic concept with respect to reduced dynamic pressure and reduced drive power are shown for constant Reynolds number and constant test-section size. For a constant tunnel size, both the shell costs, which may account for as much as two-thirds of the total cost of a wind tunnel, and the costs of the drive system for the tunnel vary nearly linearly with the maximum stagnation pressure of the tunnel. Therefore, for conditions of constant Reynolds number and tunnel size, the reduction in the stagnation pressure which is needed to achieve the desired Reynolds at cryogenic temperatures results in a reduction in capital costs even when the higher costs of the structural materials which are suitable for use at cryogenic temperatures is taken into account.

If the attainment of increased Reynolds number is accomplished by increasing stagnation pressure, for many configurations a pressure limit is reached beyond which the loads on the model will preclude testing at the desired lift coefficient. With this in mind, an alternate approach to the design of a high Reynolds tunnel is to establish the maximum usable pressure and allow tunnel size to decrease with design temperature in order to attain the desired Reynolds number. Under these conditions, there is a very strong impact of the cryogenic concept on capital cost due to the large reduction in tunnel size required for the attainment of a given Reynolds number.

At a constant pressure, the cost of the tunnel shell varies approximately with the cube of the tunnel size, whereas the cost of the drive system varies approximately with the square of the tunnel size. Thus, a reduction in tunnel size by a factor of 5 or 6, which, as can be inferred from Figure 1, may be realized by operating at cryogenic temperatures, represents a substantial savings in capital costs over the much larger ambient-temperature tunnel which would be required to achieve the desired Reynolds number at the same stagnation pressure.

2.1.3 Reduced Peak Power Demand and Total Energy Consumption.

As previously noted, because of the high peak power demands of large ambient-temperature transonic tunnels, the tunnel designer has, up until now, been forced to abandon the conventional continuous-flow transonic tunnel and adopt some form of intermittent tunnel using energy storage techniques. However, since a fan is the most efficient method of driving a tunnel, the reduction in peak power demand obtained by going to conventional energy storage techniques is realized only by accepting an increase in total energy consumption. The cryogenic tunnel concept shifts the primary energy consumption from the electric-drive system to the cooling system. Spraying liquid nitrogen directly into the tunnel circuit cools the tunnel and the stream in the cryogenic-tunnel concept developed at Langley. Plants which operate continuously at relatively low power produce liquid nitrogen which is stored at the tunnel site for use as needed. The cryogenic tunnel thus offers the tunnel designer a new type of "stored-energy" system in which the stored energy is used to reduce the drive power rather than provide it. Combined with the highly efficient conventional fan-driven closed-return tunnel, this approach greatly reduces the total power consumption as shown in Figure 5, in comparison with ambient-temperature fan-driven tunnels (which had been accepted as the most efficient from an energy standpoint). Thus, by reducing the drive-power requirements to a level where a fan drive again becomes practical even for large transonic tunnels, the cryogenic concept makes available not only the many technical advantages of the conventional continuous-flow tunnel but, at the same time, results in significant reductions in the total energy consumed during a test for a given Reynolds number and stagnation pressure. This reduction in total-energy requirement which results from cryogenic operation is especially significant in this age when the conservation of energy is assuming increasing importance.

2.1.4 Unique Operating Envelopes.

In addition to the advantages of reduced dynamic pressures and reduced drive-power and and energy requirements, the cryogenic wind-tunnel concept offers the aerodynamic researcher some unique and extremely useful operating envelopes. For a given model orientation, any

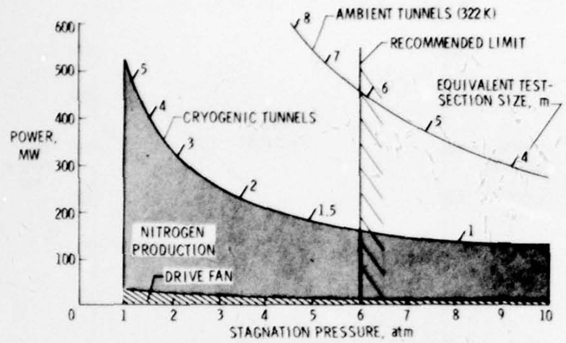


Figure 5. Total power required for continuous running as a function of stagnation pressure for ambient and cryogenic fan-driven tunnels. $M_{\infty} = 1.0$, $R_{\bar{c}} = 50$ million.

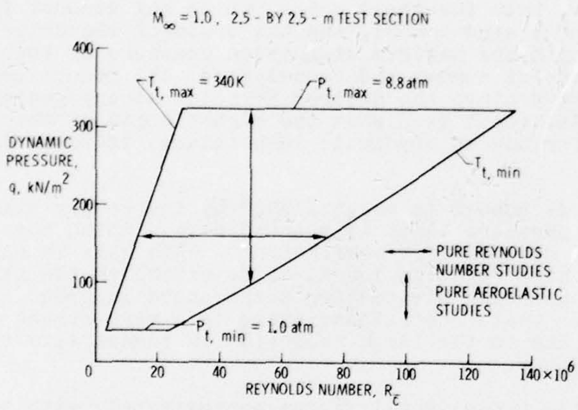


Figure 6. Constant Mach number operating envelope for cryogenic nitrogen tunnel.

aerodynamic coefficient is a function of, among other things, Mach number M, Reynolds number R, and the aeroelastic distortion of the model, which is, in turn, a function of the dynamic pressure q. A cryogenic tunnel with the independent control of Mach number, temperature, and pressure has the unique capability to determine independently the effect of Mach number, Reynolds number, and aeroelastic distortion on the aerodynamic characteristics of the model.

To illustrate the manner in which this is accomplished, operating envelopes for three modes of operation are presented for a cryogenic transonic pressure tunnel having a 2.5- by 2.5-m test section. The main purpose of presenting these operating envelopes is to illustrate the various modes of operation. However, the size of the tunnel and the ranges of temperature, pressure, and Mach number have been selected to show the anticipated characteristics of the future high Reynolds number National Transonic Facility.

2.1.4.1 Constant Mach Number Mode.

A typical operating envelope showing the range of dynamic pressure and Reynolds number available for sonic testing is presented in Figure 6. The envelope is bounded by the maximum temperature boundary (taken in this example to be 340 K), the minimum temperature boundary (chosen to avoid saturation at free-stream conditions), the maximum pressure boundary (8.8 atm), and the minimum pressure boundary (1.0 atm).

With such an operating capability, it is possible, for example, to determine at a constant Mach number the true effect of Reynolds number on the aerodynamic characteristics of the model, without having the results influenced by changing model shape due to changing dynamic pressure, as is the case in a conventional pressure tunnel. (There will be a slight variation of the modulus of elasticity E of most model materials with temperature. To correct for this variation in E, the dynamic pressure q may be varied by varying total pressure so that the ratio q/E remains constant over the Reynolds number range.) This ability to make pure Reynolds number studies is of particular importance, for example, in research on the effects of the interaction between the shock and the boundary layer. As indicated on the envelope, pure aeroelastic studies may be made under conditions of constant Reynolds number. In addition, combinations of R and q can be established to represent accurately the variations in flight of aeroelastic deformation and changes in Reynolds number with altitude. Similar envelopes are, of course, available for other Mach numbers.

2.1.4.2 Constant Reynolds Number Mode.

A typical operating envelope is presented in Figure 7, which shows the range of dynamic pressure and Mach number available for testing at a constant Reynolds number of 50×10^6. The maximum temperature limit and the maximum and minimum pressure limits used in the previous section have been assumed. The minimum temperatures were never less than those consistent with avoiding saturation under free-stream conditions. With this mode of operation, it is possible to determine true Mach number effects by eliminating the usual problems introduced by changes of Reynolds number or by changes in the aeroelastic effects.

2.1.4.3 Constant Dynamic Pressure Mode.

Although the three methods of testing have been illustrated in the preceding envelopes, an additional form of operating envelope is illustrated in Figure 8, which shows the range of Reynolds number and Mach number available at a constant dynamic pressure of 100 kN/m^2.

The cryogenic tunnel can furnish many other combinations of Reynolds number and dynamic pressure that may be desired to match a particular aircraft flight condition. The ability to isolate these effects is extremely desirable, since both aeroelasticity and

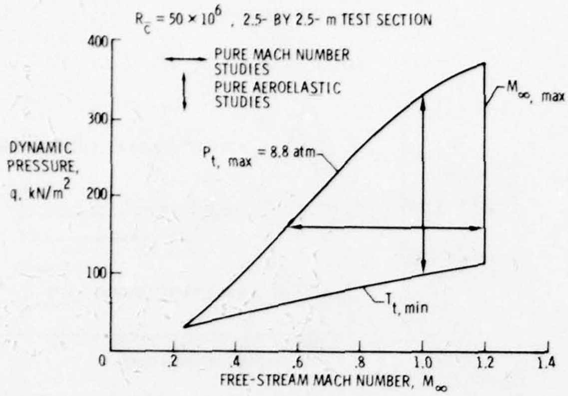


Figure 7. Constant Reynolds number operating envelope for cryogenic nitrogen tunnel.

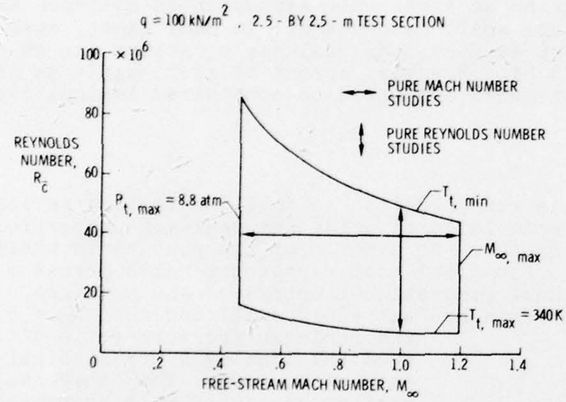


Figure 8. Constant dynamic pressure operating envelope for cryogenic nitrogen tunnel.

Reynolds number can produce profound effects on critical aerodynamic phenomena, such as shock boundary-layer interactions.

2.2 Real-Gas Studies

In the cryogenic tunnel concepts developed at Langley, the test gas is nitrogen rather than air. At ambient temperature and pressure there is no doubt that nitrogen would be an acceptable test gas since both air and nitrogen behave for all practical purposes like an ideal diatomic gas. At cryogenic temperatures, however, both nitrogen and air depart from ideal-gas behavior due to both thermal imperfections ($Pv \neq \mathcal{R}T$) and caloric imperfections (specific heats not constant). For this reason, analytical and experimental studies have been made to evaluate cryogenic nitrogen as a transonic wind-tunnel test gas. The results of these studies have been reported in Reference 13. The major findings are briefly reviewed in the following sections.

2.2.1 Minimum Operating Temperatures

The first part of the real-gas studies has been concerned with determining the minimum usable stagnation temperature. When testing at cryogenic temperatures, it is highly desirable to take maximum possible advantage of reduced temperature in order to increase test Reynolds number. As can be seen from Figure 9, the changes in Reynolds number per degree Kelvin change in stagnation temperature approaches 2% at the lower temperatures. An additional incentive to operate at lower temperatures is the reduction in fan-drive power and an attendant reduction in the amount of liquid nitrogen required for cooling.

Early theoretical studies of the minimum operating temperature were based on the assumption that condensation of the stream must be avoided under the most adverse flow conditions existing in the test section. Condensation is most likely to begin in the high local Mach number region over the model being tested where the pressure of the gas is at a minimum. Under the assumption that the gas is in static equilibrium at this low pressure, it can be shown that liquefaction of the stream will begin when the temperature associated with the low-pressure region just matches the saturated vapor temperature. Thus, under the assumption that condensation must be avoided, there exists for a given stagnation pressure and temperature a maximum local Mach number which must not be exceeded.

As noted in Reference 20, the assumptions made for the early look at minimum operating temperature were recognized as overly conservative. Based on theoretical considerations as well as on experimental results to be presented in a subsequent section, it is apparent that temperatures considerably lower than those based on maximum local Mach number considerations can, under certain circumstances, be used and still avoid any effects of condensation on the data.

2.2.2 Isentropic Expansions.

The thermodynamic properties for nitrogen were obtained from a National Bureau of Standards (NBS) program based on work by Jacobsen.[21] The NBS program was modified so that isentropic expansions could be made. The various ratios which describe an isentropic expansion were calculated by using the real-gas properties of nitrogen and were compared with

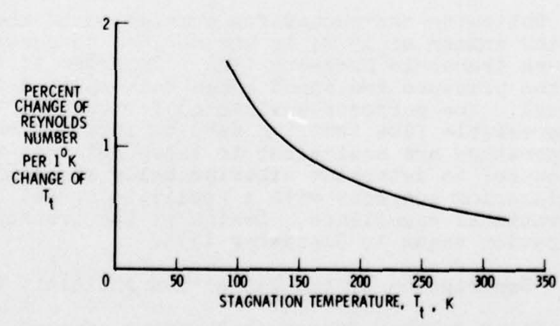


Figure 9. Change in Reynolds number per 1° change in stagnation temperature.

ratios derived from ideal-gas equations and ideal values of the compressibility factor ($Z = 1$) and the ratio of specific heats ($\gamma = 1.4$) for a diatomic gas. An example of the results is presented in Figure 10, where the ratio of the real and ideal pressure ratios necessary to expand isentropically to $M_\infty = 1.0$ is presented as a function of tunnel stagnation temperature and pressure. As can be seen, the real-gas effects are extremely small and, for $R_{\bar{c}} = 50 \times 10^6$ at cryogenic temperatures, the real-gas pressure ratio differs from the ideal-gas pressure ratio by only about 0.2%. It is interesting to note that the real-gas effect at cryogenic temperature is actually less than the real-gas effect at ambient temperatures, where a considerably higher stagnation pressure is required to obtain $R_{\bar{c}} = 50 \times 10^6$.

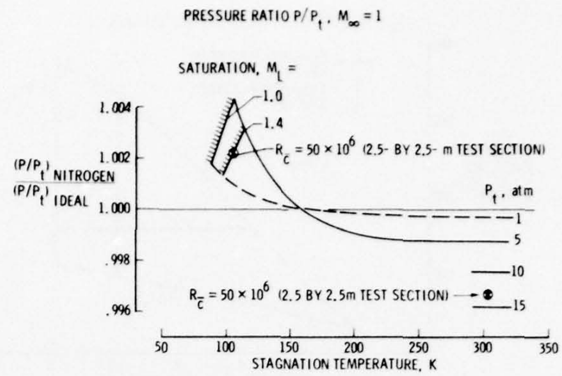


Figure 10. Pressure ratio for isentropic expansion of nitrogen to $M_\infty = 1.0$ relative to ideal diatomic gas value.

The other real-gas ratios used to describe an isentropic expansion in nitrogen also differ from the ideal-gas ratios by this same small percentage. In many cases, such as the determination of tunnel Mach number, for example, the real-gas equations can be used to avoid even this small error of 0.1% or 0.2%. However, errors of such magnitude are of the same order as the uncertainty in measurements and would be considered insignificant in most wind-tunnel work.

2.2.3 Normal-Shock Flow.

The NBS program previously mentioned was also modified so that the various ratios which describe normal-shock flow could be calculated by using the real-gas properties and compared with the corresponding ideal-gas ratios. An example of the results is presented in Figure 11 where the ratio of the real to the ideal static-pressure ratio across a normal shock is presented as a function of tunnel stagnation temperature and pressure. As in the case of isentropic expansion, the effects are extremely small and for $R_{\bar{c}} = 50 \times 10^6$, the real-gas pressure ratio differs from the ideal-gas pressure ratio by only about 0.2%. The other real-gas ratios associated with normal-shock flow in nitrogen also differ from the ideal ratios by this same small percentage. As in the case of isentropic expansion, even in those situations where the real-gas equations cannot be used to take these effects into account, an error of this magnitude would usually be considered insignificant. Thus, even though the values of Z and γ for nitrogen depart significantly from their ideal-gas values at cryogenic temperatures, both the isentropic flow parameters and the normal-shock flow parameters are insignificantly affected by these real-gas effects.

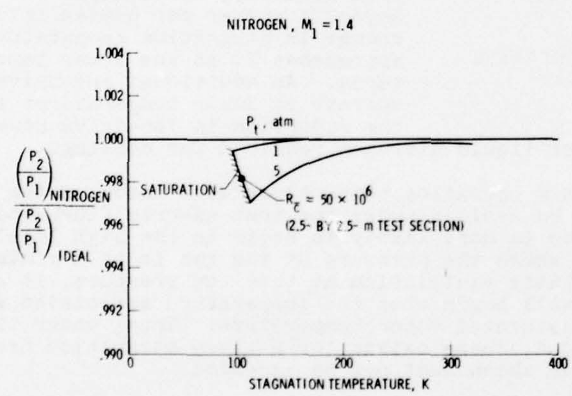


Figure 11. Pressure ratio across normal shock in nitrogen relative to ideal diatomic gas value.

3. THE LANGLEY 0.3-METER TRANSONIC CRYOGENIC TUNNEL

Following the successful completion of the low-speed cryogenic tunnel work at Langley in the summer of 1972, it was decided to construct a relatively small continuous-flow fan-driven transonic pressure tunnel in order to extend the design and operational experience to the pressure and speed range contemplated for a large high Reynolds number transonic tunnel. The purposes envisioned for the pilot transonic tunnel were to demonstrate in compressible flow that the results obtained when Reynolds number is increased by reducing temperature are equivalent to those obtained when Reynolds number is increased by increasing pressure, to determine experimentally any limitations imposed by liquefaction, to verify engineering concepts with a realistic tunnel configuration, and to provide additional operational experience. Design of the transonic tunnel began in December 1972 and initial operation began in September 1973.

3.1 Description of the Tunnel and Ancillary Equipment

The 0.3-meter Transonic Cryogenic Tunnel is a single-return fan-driven tunnel which can be operated at Mach numbers from near 0.05 to about 1.3 at stagnation pressures from slightly greater than 1 atmosphere to 5 atmospheres over a temperature range from about 77 K to 340 K (about -320° F to 152° F). The ranges of pressure and temperature provide the opportunity of investigating Reynolds number effects by temperature and pressure independently over almost a 5 to 1 range of Reynolds number.

The tunnel is designed to allow the use of interchangeable test sections. At present, there are two test-section legs available, each consisting of a contraction section, a test section and a diffuser section. The original test-section leg has a 3-D slotted octagonal test section measuring 34-cm from flat to flat. It is this 3-D test section which was used during the first three years of operation of the tunnel to obtain the proof-of-concept data and much of the operational experience to be described in subsequent sections of this paper.

Because of the renewed interest in airfoil research and the extreme sensitivity of many of the advanced airfoils to Reynolds number, a 2-D test-section leg has recently been installed. The 2-D test section measures 20-cm by 60-cm and can be used with either slotted or solid upper and lower walls. There is provision for removal of the sidewall boundary layer just ahead of the model through perforated plates extending from the floor to the ceiling. Pressure orifices on the model and a wake survey probe are used to provide the test data. In addition, a schlieren system is provided to allow for visual observation of the flow field. Preliminary tunnel-empty calibration of the 2-D test section has been completed and the testing of a series of calibration airfoils is under way.

This new 2-D test-section leg will provide a unique facility for fundamental fluid-dynamic research and airfoil development at test Reynolds numbers up to 50 million on a two-dimensional airfoil having a 15-cm chord.

A sketch of the tunnel circuit with the original 3-D test section is shown in Figure 12. Some details of the mechanical aspects of the tunnel have been reported in Reference 22. Some of the aspects of the tunnel related to its ability to operate at cryogenic temperatures are described in more detail in the sections which follow.

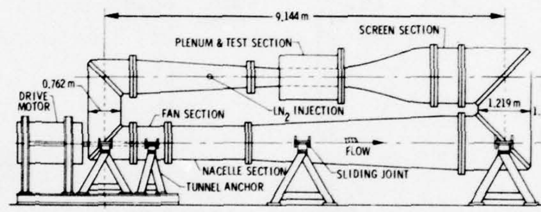


Figure 12. Layout of 0.3-meter Transonic Cryogenic Tunnel.

3.1.1 Materials of Construction.

The tunnel pressure shell is constructed of 0.635 and 1.270 cm thick plates of 6061-T6 aluminum alloy. The flanges used to join the various sections of the tunnel were machined from plates of this same material. The bolts for the flanges are made from 2024-T4 aluminum alloy. These particular aluminum alloys were selected because they have good mechanical characteristics at cryogenic as well as ambient temperatures and could easily be fabricated using equipment and techniques available at Langley. The flange joints, depending upon size, are sealed with either a flat gasket made from a mixture of teflon resin and pulverized glass fiber or a teflon-coated hollow metal "o"-ring.

3.1.2 Tunnel Support System.

The 3200-kg tunnel is mounted on the four "A"-frame support stands shown in Figure 12, one of which is a "tunnel anchor" support designed so that the center of the fan hub keeps a fixed position relative to the drive motor. Thermal expansion and contraction of the tunnel results in a change in overall tunnel length of about 4.0 cm between extremes in operating temperature. Sliding pads at each of the tunnel support attachments allow free thermal expansion or contraction of the tunnel structure. The sliding surfaces consist of 2.54 cm thick teflon sheets placed between the support attachments on the tunnel and stainless-steel blocks mounted on the upper portion of the carbon steel "A"-frame stands. Vertical and lateral movement at each joint is constrained by bolts passing through the tunnel support attachment and the teflon sheet into the "A"-frame. The tunnel support attachments are slotted in the longitudinal direction to allow free longitudinal expansion or contraction of the tunnel.

The object of the anchor support is to hold the center line of the tunnel at this support in a fixed position relative to the ground in the presence of relatively large amounts of thermal expansion. The undersides of all of the tunnel support attachments, including those at the anchor point, are on the horizontal plane through the axis of the return leg of the tunnel. With symmetrical expansion, the tunnel center line is held at a fixed height above the ground. A fork on the tunnel underside at the anchor point additionally prevents lateral or axial movement of the tunnel at this station. In this way, the axis at the anchor point is fixed relative to the ground and the tunnel expands and contracts symmetrically about this point on the axis. Since the fan and tunnel are both manufactured from aluminum, thermal expansion does not materially affect the tip clearance of the fan or generate any misalinement between the axis of the fan and the externally mounted drive motor. This support scheme has proved to be entirely adequate and no problems have been encountered.

3.1.3 Thermal Insulation.

Thermal insulation for most of the tunnel circuit consists of 12.7 cm of urethane foam applied to the outside of the tunnel structure with a glass fiber reinforced polyester vapor barrier on the outside. The urethane foam is not bonded directly to the aluminum tunnel wall but rather is separated from the wall by a shear layer consisting of two layers of fiberglass cloth. This allows the differential expansion between the aluminum and urethane foam to take place without causing the foam to fracture. This insulation has proved adequate and keeps the outside of the tunnel warm and dry under all operating conditions even during periods of high humidity.

3.1.4 Viewing Ports.

Seven ports were provided with the initial 3-D test section to allow inspection of the interior of the tunnel in the plenum and test-section areas and the nitrogen injection regions. A sketch of one of the ports showing details of construction is shown in Figure 13. Each of the ports consists of a 3.56-cm-diameter glass window which is designed to take the maximum differential pressure of 4 atmospheres at cryogenic temperatures. To provide protection against possible window failure and to provide thermal insulation, two 0.953 cm thick sheets of clear polycarbonate resin plastic, separated by air gaps, are fitted securely over each glass window. Should these sheets of plastic also fail following failure of the glass window, additional protection is provided by a third sheet of plastic fitted as a blast shield to standoff supports on the port assembly such that the shield is not subjected to the tunnel pressure. The tunnel is capable of operating indefinitely at cryogenic temperatures and therefore it is necessary to purge with ambient temperature dry nitrogen between the layers of plastic in order to prevent dew or frost from forming on the outer surface.

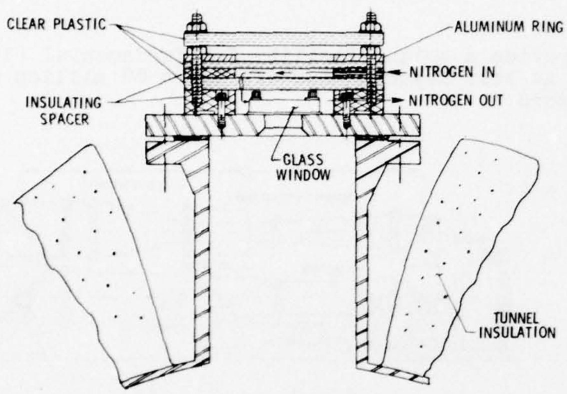


Figure 13. Viewing port of 0.3-meter tunnel.

It should be noted that there is no fundamental reason for using such small diameter ports. The small size of the ports was chosen to limit to a harmless level the pressure rise which would occur in the building which houses the tunnel in the event of failure of a glass window with the tunnel operating at maximum pressure and minimum temperature.

Initially, the inside of the tunnel was illuminated by directing the collimated output of three small incandescent lamps into the tunnel through three of the ports. However, a simple yet very effective light source was subsequently placed inside the plenum chamber which provided illumination for both the test section and plenum regions. The new light source consisted of two 12-volt automobile stoplight bulbs mounted in a plastic box which, in turn, was fastened to the plenum side of one of the test-section walls.

3.1.5 Drive-Fan System.

The tunnel has a fixed geometry drive-fan system which consists of seven prerotation vanes and a 12-bladed fan followed by 15 antiswirl vanes. The drive fan is powered by a 2.2-MW water-cooled synchronous motor with variable-frequency speed control. The motor, which is external to the tunnel, is capable of operating the fan at speeds from 600 rpm to 6200 rpm.

3.1.6 Liquid Nitrogen System.

A schematic drawing of the liquid nitrogen system is shown in Figure 14. Liquid nitrogen is stored at atmospheric pressure in two vacuum insulated tanks having a total capacity of about 212,000 liters (56,000 U.S. gallons). Immediately prior to running the tunnel, the supply tank is pressurized to about 1.7 atmospheres absolute pressure in order to maintain sufficient net positive suction head at the pump inlet to prevent cavitation. The pump has a capacity of about 500 liters per minute with a delivery pressure of 9.3 atmospheres absolute, and is driven by a 22.4-kW (30 hp) constant-speed electric motor.

When the pump is used, the liquid nitrogen supply pressure is set and held constant by the pressure control valve, shown in Figure 14, which regulates the amount of liquid returned to the storage tank through the pressure-control return line.

Tunnel total temperature is controlled by a closed-loop feedback control system which regulates the flow of LN_2 into the tunnel circuit. Four identical 10 bit-11 element digital control valves operate in accordance with command signals issued from a digital

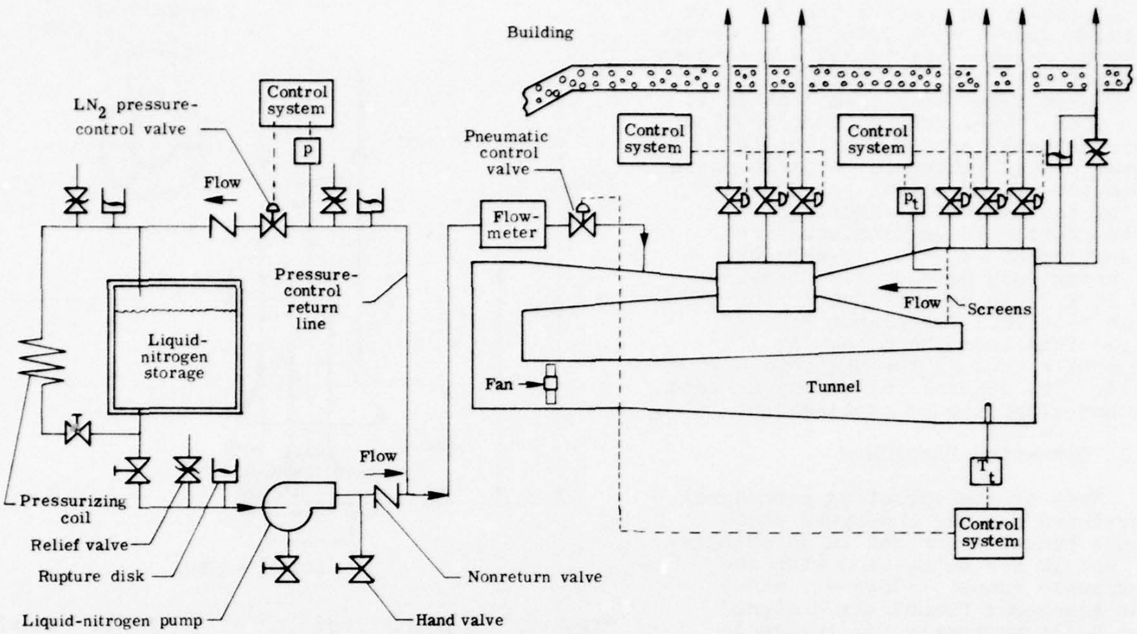


Figure 14. Schematic drawing of liquid nitrogen and exhaust systems of 0.3-meter tunnel.

microcomputer-controller. A platinum resistance thermometer located in the settling chamber serves as the temperature sensing element for the digital control system. Due to the wide ranges of tunnel pressure and Mach number, it is necessary to control LN_2 flow rates from about 4 liters per minute to about 500 liters per minute. The 1023:1 turndown ratio provided by the 10-bit digital system makes possible precise control of the LN_2 flow rate and, therefore, precise control of tunnel total temperature over this very wide range. In addition to providing optimum total temperature control, the microcomputer-controller also digitally calculates the instantaneous LN_2 flow rate with a high degree of accuracy.

In its original configuration, the liquid nitrogen system did not have the pressure control loop as shown in Figure 14 but rather was dead ended at the tunnel. This resulted in an excessive cool-down time for the LN_2 supply pipe leading to the tunnel. The minimum time required to cool the pipe between the LN_2 pump and the injection station with the original system was determined by the rate at which the gas generated during the cool-down process could be vented to the atmosphere through the tunnel exhaust system without damage to a turbine flow meter located in the liquid nitrogen supply pipe. The minimum supply pipe cool-down time with the original system was about 30 minutes. A recirculating loop supply system extending to the tunnel has reduced the cool-down time to 2 or 3 minutes and has also simplified the control of the liquid nitrogen flow rate by eliminating boiling in the pipe during tunnel operation which was also a problem with the original system at very low flow rates.

3.1.7 Nitrogen Exhaust System.

The system for exhausting gaseous nitrogen from the tunnel is shown schematically in Figure 14. Tunnel total pressure is adjusted by means of an 8-bit digital and two pneumatically operated control valves in exhaust pipes leading to the atmosphere from the low-speed section of the tunnel. In order to minimize flow disturbance the exhaust pipes are taken from the low-speed section at 120° intervals just ahead of the third set of turning vanes. A total pressure probe located downstream of the screens provides the reference pressure measurement when tunnel pressure is being controlled automatically.

As originally designed, the nitrogen exhaust from the tunnel vented directly to the atmosphere through pipes carried through the roof of the building housing the tunnel. A severe fogging problem existed with this original design during periods of high humidity and low wind speed. On several occasions, it was necessary to suspend operations until there was a favorable change in the weather. A simple and completely effective solution to this problem has been found and consists of an exhaust driven ejector as shown in the sketch in Figure 15. The low-pressure ejector induces ambient air which dilutes and warms the cold nitrogen exhaust gas. The resulting foggy mixture is propelled high into the air and dissipates completely. This simple exhaust ejector, which has an area ratio of about 5:1, induces sufficient ambient air to increase the O_2 level of the mixture to 11% and discharges the mixture so effectively that it has completely eliminated the fogging problem even under the most adverse weather conditions.

In order to operate the 3-D test section above $M_\infty = 1.06$, it is necessary to operate the tunnel a pressures sufficiently high to allow gas to be exhausted from the plenum chamber to the atmosphere. Manually controlled pneumatically actuated valves in three pipes leading to the atmosphere from the plenum allow approximately 1% of the mass flow entering the test section to be exhausted when operating at $M_\infty = 1$ or greater. By using this method, Mach numbers up to 1.3 can be obtained in the 3-D test section. The plenum exhaust pipes lead from the plenum at 120° intervals through the upstream plenum wall. The control valves may be used either singly or in combination.

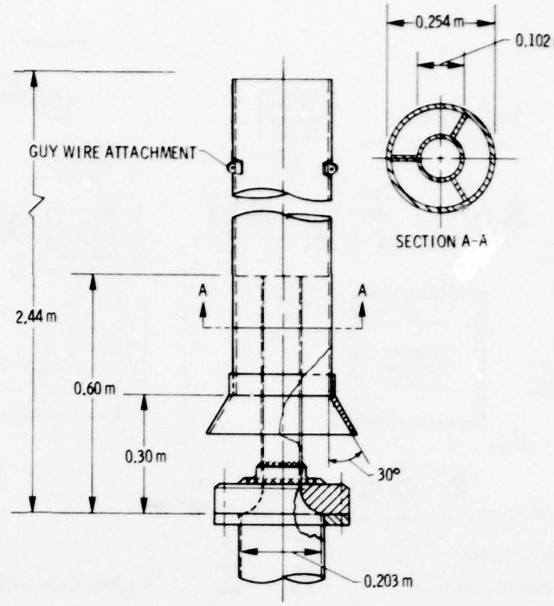


Figure 15. Exhaust ejector of 0.3-meter tunnel.

3.2 Operating Procedure

Many of the operating procedures developed for the low-speed cryogenic tunnel described in References 11 and 12 are being used with the transonic tunnel. However, since the transonic tunnel was designed and built purposely for cryogenic operation, the detailed operating procedures differ from these developed for the low-speed tunnel which required extensive modification to permit its use at cryogenic temperatures. The major conclusions with respect to the operation of the 0.3-meter tunnel after more than 1000 hours of running at cryogenic temperatures are as follows:

1. Purging, cool down, and warmup times are acceptable and can be predicted with good accuracy.

2. Liquid nitrogen requirements for cool down and running can be predicted with good accuracy.

3. Cooling with liquid nitrogen is practical at the power levels required for transonic testing. Test temperature is easily controlled and good temperature distribution can be obtained by using a simple liquid nitrogen injection system.

The experimental data on which these conclusions are based as well as other information related to the operation and performance of the 0.3-meter tunnel have been reported in References 23 and 24. A brief description of the operating procedures currently being used with the 0.3-meter tunnel is given in the sections which follow.

3.2.1 Purging.

If the tunnel has been opened and moisture from the atmosphere allowed to enter, it is essential that the moisture be removed prior to cooling the stream and tunnel in order to prevent blocking of the screens by frost. This purging of the moisture is accomplished by injecting liquid nitrogen into the tunnel circuit and allowing it to evaporate using the tunnel fan to maintain circulation and provide sufficient heat to keep the stream above the dew (or frost) point of the gas in the tunnel. The nitrogen exhaust system valves at the settling chamber are used to keep the tunnel total pressure at about 1.2 atmospheres during the prerun purge. After about 5 minutes or so the dew point is usually very close to the lower limit of measurement of the dew-point monitoring system. This limit is about 200 K (-80° F). Cool down of the tunnel then commences.

3.2.2 Cool Down.

Following the prerun purging process, the stream and tunnel are cooled to the desired operating temperature by injecting liquid nitrogen into the tunnel at the rate of about 75 liters per minute. The tunnel total pressure is held near 1.2 atmospheres and the drive fan is operated at a constant speed of about 700 rpm during the cool-down process. This low fan speed provides the necessary circulation in the tunnel during the cool-down process without adding a significant amount of heat to the stream. Under these conditions, cooling the tunnel and the stream from 300 K (80° F) to 110 K (-262° F) requires, on average, 2450 liters of LN_2 and takes about 30 minutes.

3.2.3 Running.

Following cool down of the stream and tunnel to the desired operating temperature, test Mach number is set by adjustment of fan speed. The setting of Mach number is made while the tunnel total pressure is near 1.2 atmospheres. A small computer automatically provides the operators with a continually updated display of Mach number based on the ratio of total pressure measured downstream of the screens to static pressure measured in the plenum. Once Mach number is set, the desired operating total pressure is obtained by adjustment of the nitrogen-exhaust control valves located at the settling chamber. While adjusting total pressure, the liquid nitrogen flow rate must also be adjusted to hold total temperature constant since the heat added by the fan is changing in direct proportion to total pressure. With the existing control systems, this procedure takes between 1 and 5 minutes with variation of pressure between 1.2 and 5 atmospheres.

During the initial operation of the tunnel, there was no major effort made to improve on the performance of the various control systems. However, improvements are now being made to provide automatic control of tunnel total temperature and pressure as well as Mach number. These improvements are being made in order to reduce the time required to obtain a desired test condition as well as to provide more accurate control once the desired condition has been achieved.

3.2.4 Warmup and Reoxygenation.

On many occasions, a test at cryogenic temperatures is not completed by the end of the day. The design of the 0.3-m tunnel is such that it may be left cold overnight and started up again the next day, thus avoiding the time and expense of a warmup and an additional complete cooldown cycle. At the conclusion of a test at cryogenic temperatures, the tunnel is warmed, depressurized, and reoxygenated so that model changes may be made at ambient conditions. The warming of the tunnel is accomplished by stopping the flow of LN_2 and continuing to operate the drive fan. The time required to warm the tunnel varies between 30 minutes and 1 hour, depending on the temperature range through which the tunnel is warmed and the total pressure and Mach number maintained during the warming process.

When the tunnel is warmed, and with the fan still running, the valves which are normally used to exhaust nitrogen from the plenum chamber and the settling chamber are opened. This results in an influx of ambient air into the tunnel through the plenum chamber, with corresponding efflux through the settling chamber vents, the plenum chamber holding slightly below atmospheric pressure while the settling chamber holds a pressure slightly above atmospheric. This pumping action brings the oxygen level in the tunnel to 20% by volume within 1 or 2 minutes, depending upon the Mach number. The fan is then stopped with the exhaust valves open, leaving the tunnel warm, depressurized, and reoxygenated.

3.2.5 Temperature Versus Time During a Typical Run.

A record of stream and tunnel wall temperature as a function of time is shown in Figure 16 to illustrate the various phases of tunnel operation during a typical run. The tunnel had not been opened to the atmosphere prior to the run illustrated so the prerun purging operation was not necessary. In order to avoid excessive thermal stresses in the tunnel structure, the cool down and warmup rates are generally held to less than 10° Kelvin per minute. Following the 40 minutes taken for this particular cool down, there is a 52-minute period of testing. In the example shown, there were eight different test conditions established at temperatures from 86 K to 103 K at pressures between 4.31 and 5 atmospheres and Mach numbers between 0.740 and 0.755.

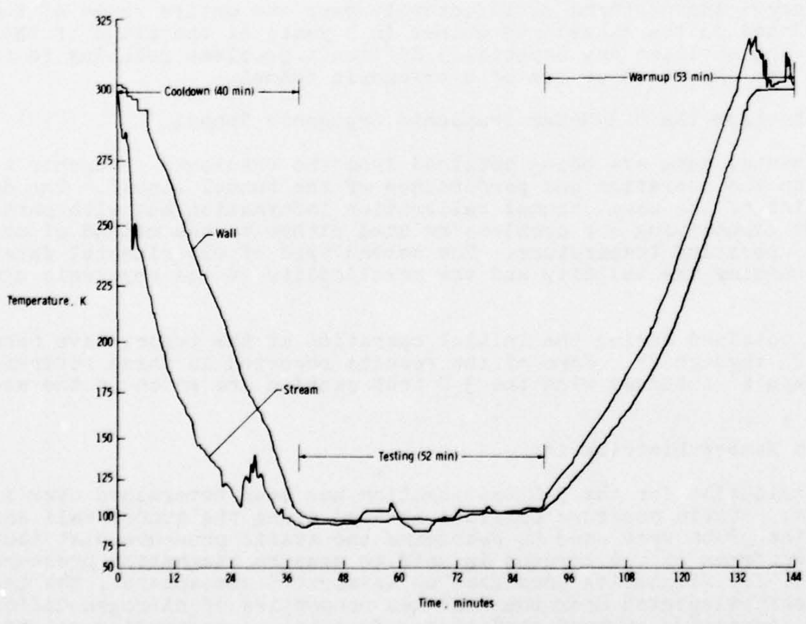


Figure 16. Stream and wall temperature as a function of time during a typical run.

3.3 Instrumentation

In addition to special instrumentation required for test-section calibration and special aerodynamic tests, the tunnel is instrumented to measure wall temperatures and static pressures around the circuit, dew point (or frost point) and oxygen content of the test gas, pressure in the LN_2 supply line, flow rate as well as total quantity of the LN_2 injected into the tunnel, flow rate of the gaseous nitrogen being exhausted from the settling chamber and the plenum chamber, fan speed, and torque at the drive motor shaft.

In working with the low-speed and transonic cryogenic tunnels, a philosophy has been adopted of maintaining, if possible, all pressure and force transducers at near ambient temperatures (≈300 K) in order to avoid possible problems with changes in sensitivity or changes in zero reading with temperature. In keeping with this philosophy, the various transducers are either mounted outside the tunnel in the ambient temperature environment or insulated and maintained at ambient temperatures by using electrical heaters if they must be used in the cryogenic environment of the tunnel. Examples of transducers maintained at ambient temperatures in the tunnel include a three-component force balance and an accelerometer used in conjunction with the testing of a force model to be described in a later section. Temperature controllers responding to changes in resistance of nickel resistance thermometer elements are used to hold the transducers at the selected ambient temperature set point within very close limits.

Temperatures over the entire range of operating temperatures (77.4 K to 340 K) are measured using either copper-constantan thermocouples or platinum resistance thermometers (PRT) depending on how accurately a particular temperature must be determined. Three PRT's are located in the settling chamber of the 0.3-meter tunnel. One of the PRT's is used to provide the tunnel operators with a continuous display of total temperature as the tunnel is brought to the desired total temperature. The other two PRT's are read directly into the data acquisition system and used in the reduction of the test data.

Pressures are measured using high accuracy (±0.25 percent of reading) capacitance-sensing diaphragm gages located outside the tunnel. For measuring the pressures around the tunnel circuit and for airfoil pressure tests, scanning valves are used to reduce the total number of transducers. Those scanning valves located inside the tunnel circuit are insulated and maintained at about 300 K by using a thermostatically controlled electrical heater. The tunnel operators are provided with a display of tunnel total pressure, plenum static pressure, and test-section Mach number calculated from these pressures by a small digital computer.

Very few problems with instrumentation have been experienced in the operation of the 0.3-meter cryogenic tunnel where, generally, off-the-shelf components and standard instrumentation techniques have been used and found to be satisfactory. The majority of the few problems that have been encountered have been due to the high operating pressure of 5 atmospheres rather than the low operating temperature. The only major problem due to the cryogenic environment that has been encountered were zero shifts of the normal force and pitching-moment components of the electrically heated balance due to temperature gradients across the gage section.[20] These gradients were due mainly to flow from the base of the model being circulated across the gage section. By modifying the original model-balance interface so as to minimize conduction, by optimizing the locations of the heaters and sensors, and by shielding the balance from all direct contact with the cold stream, a balance has been evolved from the original design which is completely free from zero shifts with temperature and performs satisfactorily over the entire range of temperatures and pressures. Based on the experience gained in 4 years of operation of the 0.3-meter tunnel, we do not envision any especially difficult problems relating to instrumentation in either the operation or use of a cryogenic tunnel.

3.4 Experimental Results From the 0.3-Meter Transonic Cryogenic Tunnel

Two types of experimental data are being obtained from the transonic cryogenic tunnel. The first type relates to the operation and performance of the tunnel itself. The data, for the most part, consist of the usual tunnel calibration information but with particular emphasis being placed on identifying any problems related either to the method of cooling or to the wide range of operating temperature. The second type of experimental data is primarily aimed at determining the validity and the practicality of the cryogenic concept in compressible flow.

Some of the results obtained during the initial operation of the tunnel have been reported in References 23 through 27. Some of the results reported in these references along with additional results obtained with the 3-D test section are given in the sections which follow.

3.4.1 Test-Section Mach Number Distribution.

The Mach number distribution for the 3-D test section has been determined over a wide range of test conditions. Static pressure orifices located along the tunnel wall and along a tunnel center-line probe were used to determine the static pressure distribution. A pitot tube located downstream of the screens is used to measure stagnation pressure. As noted in Reference 13, for stagnation pressures up to about 5 atmospheres, the isentropic expansion relations calculated from the real-gas properties of nitrogen differ by no more than about 0.4% (depending on test conditions) from the corresponding relations calculated from ideal diatomic gas properties and ideal-gas equations. However, since the pressure-measuring instrumentation being used is capable of resolving such differences, the appropriate real-gas relation is used in the data reduction process to determine Mach number from the stagnation to static pressure ratio. The nominal test-section Mach number is calculated from the ratio of stagnation pressure measured just downstream of the screens to static pressure measured in the plenum chamber.

The calibration of the 3-D test section indicates nearly identical tunnel wall and tunnel center-line Mach number distributions for all test conditions. In addition, there are no detectable differences between Mach number distributions at ambient and cryogenic temperatures. An example of the tunnel wall and tunnel center-line Mach number distribution is shown in Figure 17. Examples of the wall Mach number distribution over a range of Mach numbers is shown in Figure 18. Except at speeds greater than Mach 1, the Mach number distribution is generally good and no attempt has yet been made to improve the distributions by changes to slot geometry, wall divergence, or reentry flap position.

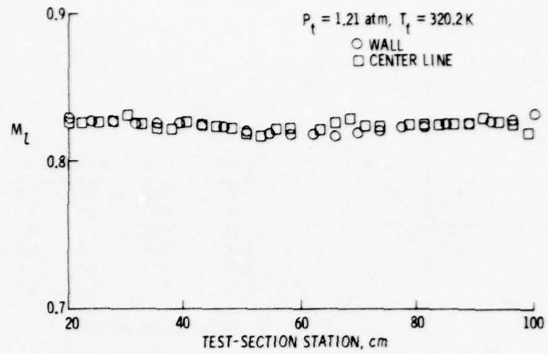


Figure 17. Example of test-section wall and center-line Mach number distribution at $M_\infty \approx 0.825$.

3.4.2 Transverse Temperature Distribution.

Since the heat of compression of the fan is removed by spraying liquid nitrogen directly into the tunnel circuit, there was some concern about the uniformity of the resulting temperature distribution, particularly at the power levels required for transonic testing where the mass flow rate of liquid nitrogen is in the order of 1% of the test-section mass flow rate. Therefore, it was decided to measure the transverse temperature distribution in the tunnel over a wide range of operating conditions in order to determine if there were any problems of uniformity of temperature distribution related to the method of cooling.

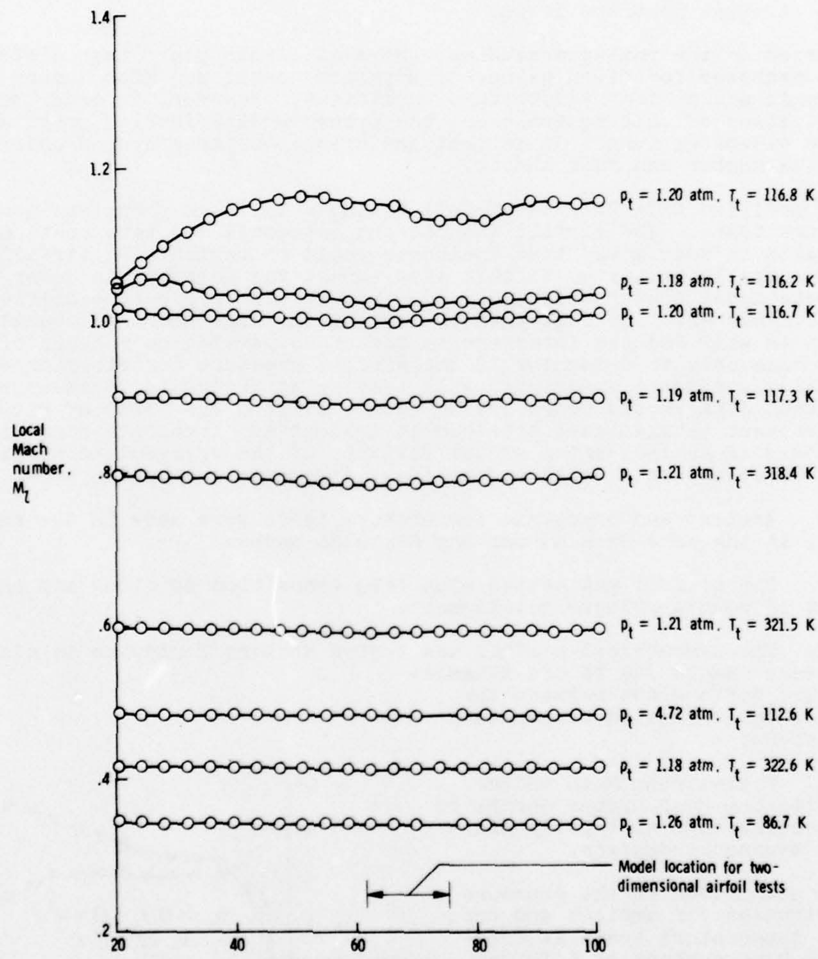
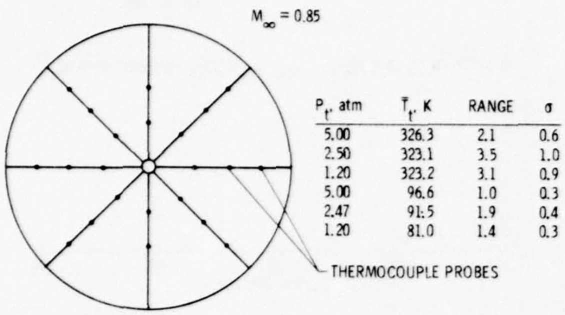


Figure 18. Examples of test-section wall Mach number distributions.

The temperature distribution in the tunnel is determined by a temperature survey rig which is permanently installed in the tunnel just ahead of the turbulence damping screens. A sketch of the temperature survey rig showing the general location of the 22 thermocouple probes is shown in Figure 19 along with a listing of some early results from the survey rig obtained at $M_\infty = 0.85$. The mean value (arithmetic average) of temperature $\bar{T}_t$ and standard deviation (measure of dispersion around the mean) σ are listed for several test conditions. As can be seen, there is a relatively uniform temperature distribution even at cryogenic temperatures where the tunnel is being operated within a few



P_t, atm	$\bar{T}_t$, K	RANGE	σ
5.00	326.3	2.1	0.6
2.50	323.1	3.5	1.0
1.20	323.2	3.1	0.9
5.00	96.6	1.0	0.3
2.47	91.5	1.9	0.4
1.20	81.0	1.4	0.3

Figure 19. Sketch of temperature survey rig with listing of typical results.

degrees of the test section free-stream saturation conditions. These values are typical of the data which were obtained over the entire operating envelope using the original LN_2 injection system as described in Reference 24.

Recently, a thru-the-wall injection system has been installed in the 0.3-meter tunnel. Four identical spray nozzles are mounted at 90° intervals around the tunnel at the LN_2 injection station shown in the sketch of the tunnel in Figure 12. The nozzles were mounted flush with the tunnel wall and positioned so the fan spray patterns from the nozzles were perpendicular to the flow. The new technique produces an improved temperature distribution with an average value for σ of 0.25 over generally the same range of test conditions as shown in Figure 19. By eliminating the need for plumbing inside the tunnel, the wall injection system permits a simpler and more efficient tunnel design than the original spray-bar injection system as described in Reference 24.

3.4.3 Airfoil Pressure Tests.

Based on the real-gas studies, there is little doubt that airfoil pressure distributions measured for given values of Reynolds number and Mach number should be the same at cryogenic and ambient temperature conditions. However, in order to provide experimental verification of this equivalence, the pressure distribution on an airfoil has been measured in the 0.3-meter tunnel at ambient and cryogenic temperatures under conditions of constant Reynolds number and Mach number.

A modified NACA 0012-64 airfoil having a 13.72-cm chord was used for the airfoil pressure tests. The airfoil spanned the octagonal 3-D test section and was fastened to the walls in such a way that incidence could be varied. An airfoil somewhat larger than would normally be tested in this size tunnel was selected in order to allow for more accurate model construction, a reasonable number of pressure orifices, and higher chord Reynolds number. The fact that the relatively high chord to tunnel-height ratio might result in wall induced interference was of no particular concern since the tests were being made only to determine if the airfoil pressure distribution was modified in any way by real-gas effects associated with testing at cryogenic temperatures. The pressure distribution data should therefore be looked at from the point of view of agreement or lack of agreement between data obtained at ambient and cryogenic conditions and the results used only as an indication of the validity of the cryogenic concept. The conditions selected to assure a valid and critical cryogenic evaluation were:

1. Ambient and cryogenic temperature tests were made in the same tunnel, on the same model, at the same Mach number and Reynolds number.

2. The airfoil was tested with free transition to allow any possible temperature effect on boundary-layer development.

3. The symmetrical airfoil was tested at zero incidence to eliminate any shape or incidence change due to the dynamic-pressure differences between the ambient and cryogenic temperature conditions.

4. Free-stream Mach number exceeded the Mach number normal to the leading edge of typical near-sonic transport designs.

A comparison of the pressure distribution for ambient and cryogenic temperature tests at free-stream Mach numbers of 0.75 and 0.85 are shown in Figure 20. For this comparison, the same chord Reynolds number was obtained at each temperature and constant Mach number by an appropriate adjustment of pressure with temperature. As can be seen, there is excellent agreement at both Mach numbers between the pressure distributions obtained at ambient and cryogenic conditions.

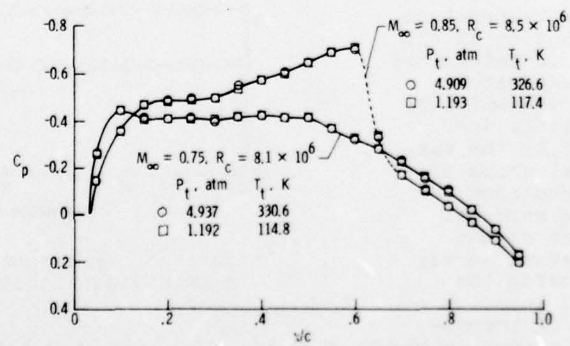


Figure 20. Comparison of pressure distributions for a symmetrical airfoil obtained at ambient and cryogenic conditions, $\alpha = 0°$.

This is considered to be a valid check in view of the large variation in gas properties over this large temperature and pressure range. In addition, this agreement is particularly significant with regard to setting tunnel conditions when one considers, for example, the large variation of speed of sound with temperature and the sensitivity of the airfoil pressure distribution to changes in Mach number.

The distribution at $M_\infty = 0.85$ is of perhaps greater significance since the pressure distribution shows the flow to be supersonic over a large portion of the airfoil, reaching a local Mach number of about 1.22 just ahead of the strong recompression shock. This type of flow, typical of supercritical flows, should be extremely sensitive to any anomalous behavior of the test gas due to operation at cryogenic temperature. The almost-perfect agreement in the pressure distributions provides experimental confirmation that nitrogen at cryogenic temperatures behaves like a perfect gas and is therefore a valid transonic test medium as predicted by the real-gas studies.

A comparison of pressure distributions obtained at temperatures of 186 K and 117 K at $M_\infty = 0.90$ is shown in Figure 21. The pressures were such that a constant chord Reynolds number of 18×10^6 was obtained for each set of data. This particular comparison is of interest since it is obvious from the lack of pressure recovery that the boundary layer had separated at the recompression shock. Although not strictly an ambient-cryogenic comparison, the excellent agreement between the two sets of data again provides experimental evidence that nitrogen at cryogenic temperatures is a valid test medium, in this case, even in the presence of separated flow.

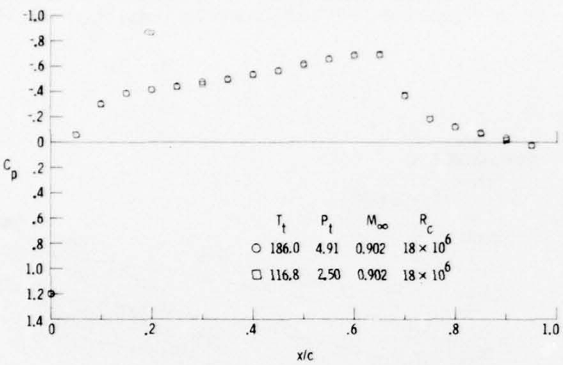


Figure 21. Comparison of pressure distributions for separated flow conditions, $\alpha = 0^\circ$.

3.4.4 Saturation Boundary Tests.

As noted in Reference 13, the saturation boundary for nitrogen is well defined and any possible effect of liquefaction of the test gas can easily be avoided if the maximum local Mach number on the model is known. For a given size tunnel and a constant operating pressure, a significant increase in test Reynolds number is possible if the saturation boundary may be crossed in these localized high Mach number regions. For example, if the test temperature for sonic testing is selected to avoid the saturation boundary based on free-stream Mach number, the test Reynolds number is about 30% greater than can be achieved at the same stagnation pressure had the test temperature been selected to avoid the saturation boundary based on a maximum local Mach number of 1.7. In order to determine if such an increase in test Reynolds number could safely be realized, an experiment was needed to investigate the feasibility of testing under conditions of local supersaturation.

The pressure distribution of an airfoil having a strong recompression shock should provide a sensitive indicator of condensation, which would be expected to occur in the low-pressure region ahead of the shock. Therefore, the airfoil pressure model described in the previous section was used also in a series of supersaturation tests.

Some data obtained during the initial operation of the 0.3-meter tunnel indicated that several degrees of local supersaturation had no effect on the pressure distribution over the airfoil.[23] However, these early tests covered only a limited Mach number range and total temperatures only low enough to result in operation down to free-stream saturation conditions.

Because of the potential increase in Reynolds number capability that results from operation below local saturation conditions, an experimental program has been undertaken in the 0.3-meter tunnel to extend the saturation studies to cover a wider range of test conditions. Studies have been made at constant test-section Mach numbers of 0.75, 0.85, and 0.95 at total pressures from 1.2 to 5.0 atmospheres with temperatures to within 2° Kelvin of settling chamber saturation. The temperature at which condensation effects perturb the flow is determined at a constant Reynolds number by reducing total temperature and pressure until the pressure distribution over the airfoil deviates from the pressure distribution obtained with unsaturated conditions. The results of these studies have been reported by Hall.[28] Some of the Mach number 0.85 results from Reference 28 are included herein to illustrate the technique used for these studies as well as to illustrate the potential benefit of operation beyond both the local and free-stream saturation boundaries.

For these tests with the airfoil at zero incidence, the maximum local Mach number over the airfoil was 1.22. The three saturation boundaries of interest — local, free-stream, and settling chamber — are shown as a function of total pressure in Figure 22.

To determine the total temperature at which condensation effects begin as a function of total pressure, six paths of constant Reynolds number were investigated. The Reynolds numbers were chosen so that the total pressures required spanned the pressure envelope of the tunnel. A reference pressure distribution was taken above the upper saturation boundary shown in Figure 22 if it was possible to do so within the pressure envelope of the tunnel. The six paths of constant Reynolds number and the total conditions sampled are shown in Figure 23.

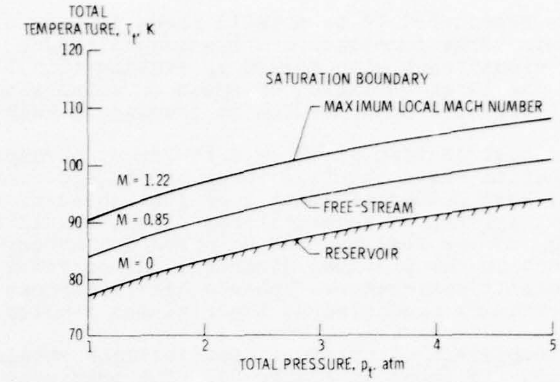


Figure 22. Saturation boundaries of interest as a function of total pressure, $M_\infty = 0.85$.

To determine if condensation effects were, in fact, perturbing the flow over the airfoil, pressure distributions obtained at the different total temperatures along a constant Reynolds number line were compared to the reference pressure distribution. Even though the agreement between the reference distribution and those distributions considered to be affected by condensation in many cases appeared to be within the experimental accuracy of the data, there was always a systematic positive shift in the pressure coefficient from the 20% chord position of the airfoil back to the recompression shock.

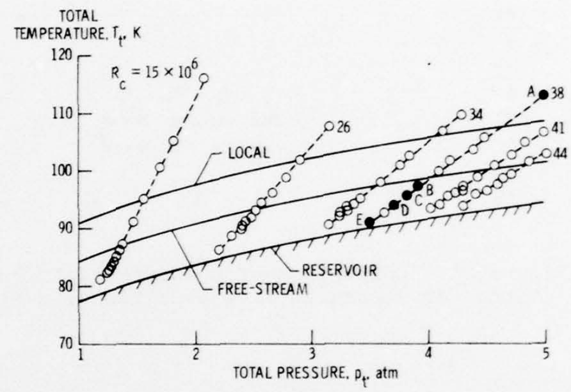


Figure 23. Paths of constant Reynolds number showing total conditions sampled, $M_\infty = 0.85$.

Four pressure distribution comparisons are presented herein. The test conditions for the individual pressure distribution all lie along the $R_c = 38 \times 10^6$ path in Figure 23 and are shown as filled symbols. The reference (condensation free) distribution, A, is compared, in turn, with the distributions B, C, D, and E, all of which lie below the free-stream saturation boundary.

The comparison shown in Figure 24 shows no effect due to operation under conditions that result in expansion to a temperature 1° Kelvin beyond the free-stream saturation boundary. A possible effect of condensation can be seen in the comparison shown in Figure 25 where the flow expanded to a temperature 2.5° Kelvin beyond the free-stream saturation boundary. The comparisons in Figures 26 and 27 both show definite effects of condensation on the pressure distribution when the flow expanded to temperatures 3.5° and 6° Kelvin, respectively, beyond the free-stream saturation boundary.

By making similar comparisons along each of the constant Reynolds number paths shown in Figure 23. Hall was able to define the conditions under which "possible" condensation effects were present over nearly the entire pressure range of the tunnel. A conservative experimental lower limit to tunnel operation was then obtained by plotting, as shown in Figure 28, the total conditions where the first "possible" effects of condensation were observed in the pressure distributions. As noted by Hall, the operating boundary defined by this locus of points is considered conservative since, in fact, the differences in pressure distributions even for those comparisons labeled "possible effect" may be insignificant to the aircraft designer.

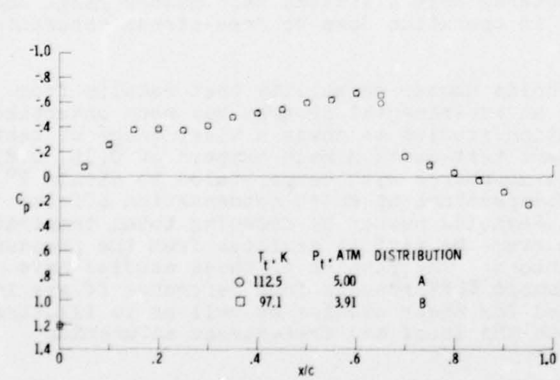


Figure 24. Pressure distribution comparison showing no effect with T_t = 97.1 K which is 1° K below free-stream saturation boundary.

For the particular airfoil used for these tests, there appear to be no serious condensation problems associated with testing at temperatures far below the saturation boundary associated with the high local Mach number over the airfoil. No low-temperature effects were observed

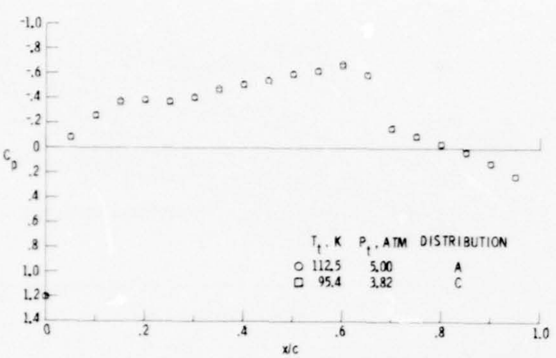


Figure 25. Pressure distribution comparison showing possible effect with T_t = 95.4 K which is 2.5° K below free-stream saturation boundary.

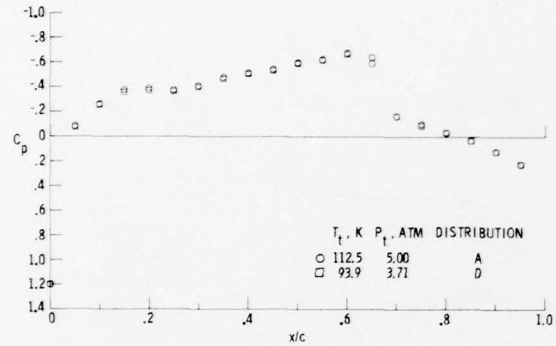


Figure 26. Pressure distribution comparison showing definite effect with T_t = 93.9 K which is 3.5° K below free-stream saturation boundary.

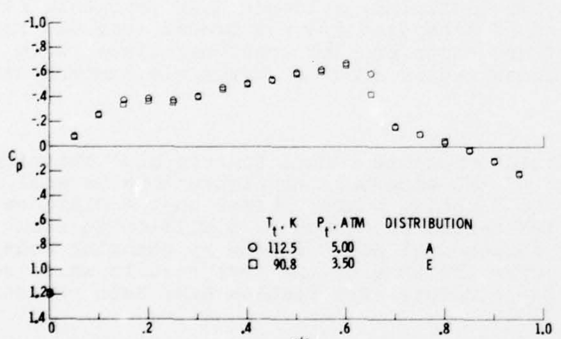


Figure 27. Pressure distribution comparison showing definite effect with T_t = 90.8 K which is 6.0° K below free-stream saturation boundary.

until total conditions were at least 2° Kelvin below those which produced saturated flow in the test section. For the conditions of this experiment, the ability to operate below the local saturation boundary allowed an increase in test Reynolds number of 17% for a constant total pressure over the entire pressure range from 1.2 to 4.5 atmospheres. If, instead of holding total pressure constant, test Reynolds number is held constant, the ability to operate with saturated flow allows a reduction in total pressure of more than 17%.

3.4.5 Three-Dimensional Force Tests.

Force tests have been made in the 3-D test section of the transonic cryogenic tunnel on a highly-swept delta-wing model with a sharp leading edge.

The three-dimensional test program was designed with the dual purpose of: (1) obtaining experience in a cryogenic wind tunnel with an electrically heated internal strain-gage balance, the accompanying sting, sting-support arc, and an accelerometer used to determine angle of attack; and (2) investigating any possible effects of cryogenic stagnation temperature on the aerodynamic characteristics of a configuration having flow characterized by a separation-induced leading-edge vortex. As reported in Reference 12, similar tests have been made previously at Langley in a low-speed cryogenic tunnel with satisfactory results using a water-jacketed internal strain-gage balance and a simple angle-of-attack mechanism.

The specific test program devised to meet the dual purpose consisted of using a complete force-measuring system — balance, sting, sting-support arc, and angle-of-attack accelerometer — for testing a delta-wing model at both ambient and cryogenic temperatures.

There were two primary considerations in the selection of the model geometry:

(1) Since the type of flow to be simulated was the leading-edge separation induced vortex flow with vortex induced reattachment, a highly-swept delta wing with sharp leading-edges was selected

(2) Since comparisons of ambient and cryogenic temperature results at the same Reynolds number required large differences in the tunnel stagnation pressure, a thick, diamond shape cross-section was selected for the delta-wing model in order to avoid aeroelastic distortions that could affect the aerodynamic results.

The model has a leading-edge sweep of 75°, is 0.1981 m long, and is 0.1067 m wide at the base. The surface of the model exposed to the airstream was aerodynamically smooth and, in order to assure freedom for any possible temperature effects on the boundary layer flow to occur, the tests were performed with free transition, that is, no transition strips were applied to the model. The model was machined from 17-4 PH stainless steel which was heat treated to condition H 1150-M. With this heat treatment, 17-4 PH can be used at temperatures as low as 77 K (-320° F).

As described in Section 3.3, a series of modifications to the original electrically heated balance was necessary in order to eliminate the zero shifts in some of the balance components when the balance was used at cryogenic temperatures. The balance that has evolved from these studies is completely free from zero shifts and performs satisfactorily over the entire range of temperatures and pressures.

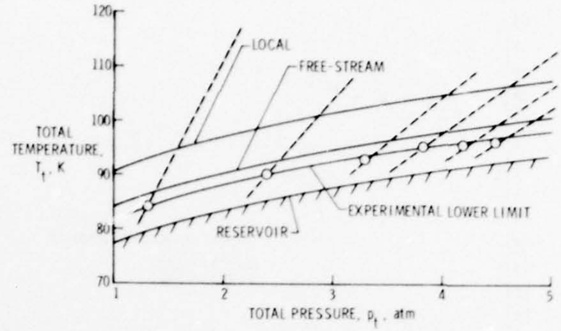


Figure 28. Experimental lower limit at which pressure distributions begin to differ from their references.

The test results obtained on the modified balance show clearly that flows with leading-edge vortex effects are duplicated properly at cryogenic temperatures.

An example of the results which have been obtained on the delta-wing model are presented in Figure 29 which shows the variation of pitching moment, drag, and lift-force coefficients with angle of attack at both ambient and cryogenic temperatures for a Mach number of 0.80. The circular symbols indicate experimental results obtained at a stagnation pressure of 4.60 atm and at a stagnation temperature of 301 K. The square symbols are data taken at a stagnation pressure of 1.20 atm and at a stagnation temperature of 114 K. With both sets of data, flagged symbols are used to indicate repeat data point. The Reynolds number, based on mean geometric chord, was 8.5 x 10^6 for the two sets of data. As can be seen, there is good agreement between the experimental results obtained at ambient and cryogenic temperatures.

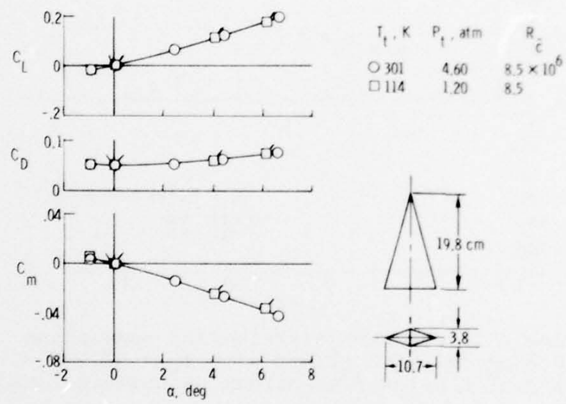


Figure 29. Delta-wing results.

The three-dimensional model results provide additional evidence that cryogenic nitrogen is a valid test gas even under conditions of separated and reattached (vortex) flow. In addition, there has been no indication of any major problem areas associated with obtaining angle-of-attack or strain-gage balance measurements at cryogenic temperatures.

3.4.6 Boattail Afterbody Pressure Tests.

Tests have been made in the 0.3-m transonic cryogenic tunnel to determine the effect of varying Reynolds number on the boattail drag of two wing-body configurations as well as on several isolated boattail configurations. The Reynolds number, based on the distance from the nose to the beginning of the boattail, was varied from about 2.6 million to about 132 million by changing unit Reynolds number by a factor of about 25 and by changing model length by a factor of 2. Included in this paper are some of the test results which relate to the validity of the cryogenic concept. The complete test results have been reported in papers by Reubush and Putnam.[29,30,31]

Shown in Figure 30 are the boattail pressure coefficient distributions for a circular arc-conic boattail at a Mach number of 0.6 and a Reynolds number of about 10.5 million. The shape of the distribution clearly indicates flow separation toward the rear of the boattail. The distribution shown with circular symbols was obtained at a stagnation pressure of 1.2 atm at a stagnation temperature of 117 K while the distribution shown with square symbols was obtained at a stagnation pressure of 5.0 atm at a stagnation temperature of 311 K. These data show nearly identical pressure distributions for ambient and cryogenic conditions and thus provide further substantiation that there are no undesirable effects in using nitrogen at cryogenic temperatures to achieve high Reynolds number data even in the presence of separated flow.

3.5 Future Plans

In addition to being used to verify the validity of the cryogenic wind-tunnel concept and providing more than 1000 hours of experience in the operation of a fan-driven cryogenic tunnel, the 0.3-meter tunnel is being used for aerodynamic research in several areas where either the very high unit Reynolds number ($R/m \approx 3 \times 10^8$ at $M_\infty = 1$) or the 25 to 1 range of Reynolds number is required. Some of the future plans for this unique facility are described in the following sections.

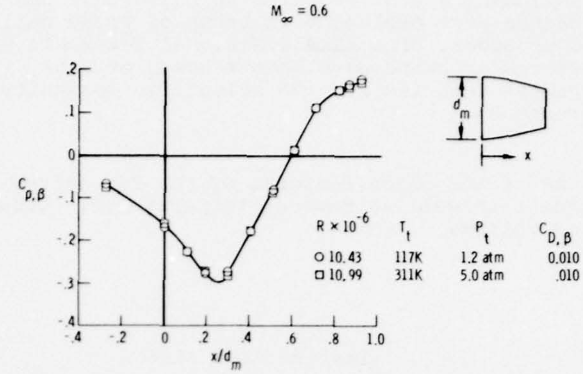


Figure 30. Boattail pressures for a circular arc-conic boattail at cryogenic and ambient conditions.

3.5.1 Self-Streamlining Two-Dimensional Test Section.

A two-dimensional self-streamlining flexible-wall test-section leg has been designed for the 0.3-meter tunnel based on the work by Goodyer at the University of Southampton.[32] Initially, the test section will be used for testing in flows where the Mach number at the walls never exceeds unity. By permitting increased chord length, the flexible-wall test section will allow testing under interference-free conditions at chord Reynolds numbers approaching 100 million.

3.5.2 Magnetic Suspension and Balance System.

The reduction in model loads made possible by the cryogenic wind-tunnel concept and the reduction in the size of the coils used in a magnetic suspension and balance system made possible by superconductor technology makes the combination of these two concepts an attractive means of providing high Reynolds number test capability free from support interference. In such a facility, it will be possible to test free of support interference effects as well as to determine the magnitude of such effects by direct comparison with data obtained by using conventional model support systems. The demonstrated ease and rapidity with which the orientation of the model may be changed with the magnetic suspension system while keeping the model in the center of the test section will facilitate the rapid acquisition of the aerodynamic data which is a desirable feature of any high Reynolds number tunnel. In addition, the retrieval of the model from the test section of a cryogenic tunnel for model configuration changes would be a simple operation with a magnetic-suspension and balance system.

Because of the many advantages offered by a magnetic-suspension and balance system, NASA has supported for several years both in-house and sponsored research in this area. Significant accomplishments resulting from NASA sponsored research include the development of an electromagnetic position sensor at the Aerophysics Laboratory of the Massachusetts Institute of Technology[33] and the development of an all-superconductor magnetic-suspension and balance system for aerodynamic testing at the Research Laboratories for the Engineering Sciences at the University of Virginia.[34]

Additional studies are being made at both Langley, the University of Virginia, and the Massachusetts Institute of Technology with the aim of building a six-component superconducting magnetic suspension and balance system to be used in conjunction with an interchangeable test-section leg for the 0.3-meter Transonic Cryogenic Tunnel.[35] Current plans are for the test section to be square in cross section and to measure approximately 0.45- by 0.45-m. The combination of 5 atmospheres operating pressure and cryogenic temperatures will result in test Reynolds numbers of about 15 million.

4. NATIONAL TRANSONIC FACILITY

Because of the pressing need for increased Reynolds number capability to provide more accurate simulation of full scale viscous effects for both aerodynamic research and aircraft development, both NASA and the U.S. Air Force developed plans during the early 1970's for high Reynolds number transonic tunnels. The approach selected by NASA was a cryogenic fan driven tunnel based on the cryogenic tunnel technology developed at the Langley Research Center. This tunnel, described in Reference 26, was referred to as the Transonic Research Tunnel (TRT) and was to be located at Langley.

The approach selected by the Air Force was an ambient temperature Ludwieg Tube driven high Reynolds number tunnel (HIRT)[7] to be located at AEDC.

Following a U.S. decision to build only one high Reynolds number facility, these two approaches were evaluated in terms of broad national needs. The evolution, from these two approaches, of a single National Transonic Facility (NTF) to meet the high Reynolds number research and development needs of NASA, the DOD, the aerospace industry, other Government agencies and the scientific community has been documented by Baals in Reference 36.

One of the major features of the fan-driven NTF will be its ability to operate at cryogenic as well as ambient temperatures. Other characteristics specified for the NTF are as follows:

Test section size	2.5- by 2.5-m
Pressure range	1 to 9 atmospheres
Mach range	0.2 to 1.2
$R_{\bar{c}}$ at $M_\infty = 1$	120×10^6
Productivity	8,000 polars per year

The process used in selecting the design features to satisfy these requirements as well as the engineering design and anticipated performance of the tunnel that has evolved are discussed in detail in Reference 37 by Howell and McKinney.

The Langley Research Center has been selected as the site of the National Transonic Facility. The design of the NTF has been completed and bids accepted for the major components of the tunnel. The total investment in the NTF will be approximately $\$85 \times 10^6$ when it becomes operational in 1983. To minimize the capital cost of the NTF, it is being constructed on the site of the deactivated 4-foot Supersonic Pressure Tunnel and will utilize the existing office building as well as the existing drive motors and their associated speed control systems and cooling towers. To provide the full tunnel capability an additional drive motor and a two-speed gear box will be added to the drive system.

4.1 Test Section Design

The test section of the NTF will be similar in design to that of the Langley 8-Foot Transonic Pressure Tunnel which is known to be efficient and have good quality flow. A plan view of the test section and the surrounding plenum is shown in Figure 31. The length of the slotted region will be approximately three times the height of the test section. The top and bottom walls will have six slots each and will have adjustable divergence to compensate for boundary layer growth. The parallel side walls will have two slots each. The design of all of the slots is such that both the slot edge shape and the slot width can be easily modified. Reentry flaps at the end of each of the slots will be varied during tunnel operation to control Mach number gradients in the test section and minimize power consumption.

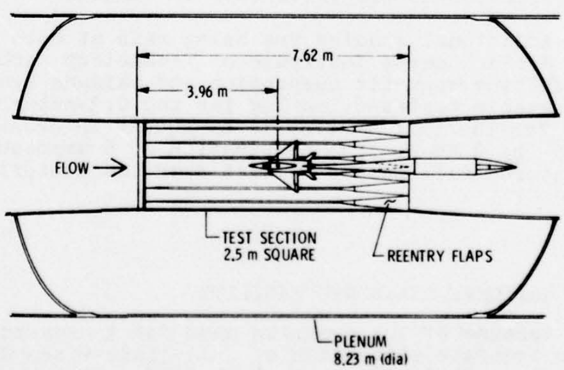


Figure 31. Plan view of the slotted test section.

The model sting support system will be an arc sector having a nominal travel of 30° with additional angle-of-attack range provided by using offset stings. The center of rotation of the arc will be 3.96 meters downstream of the test-section throat. This location places the model well ahead of the aft end of the test section which helps to insure good quality flow over the base of the model. However, as noted in Reference 37, the resulting long sting encounters stress problems for the high model loads that are generated at the higher stagnation pressures. Model pitch rate will be controllable in either a continuous sweep mode or in a pitch-pause mode at rates up to 4° per second.

4.2 Tunnel Circuit

A contraction ratio of 15 to 1 has been selected for the NTF in order to reduce turbulence levels in the test section to 0.1 percent or less while at the same time reducing the pressure losses across the turbulence damping screens to an acceptable level. Up to five screens can be used in order to insure adequate turbulence damping. Details of the analysis used to select the contraction ratio - screen combination are given in Reference 37.

A sketch of the aerodynamic lines of the NTF is shown in Figure 32. A rapid diffuser will be used upstream of the stilling chamber in order to provide the required contraction ratio while keeping the tunnel volume to a minimum. The pressure drop required to make the rapid diffuser work properly will be provided by a water-air heat exchanger which will be used for ambient temperature operation. The tunnel circuit will be completed using a combination of near optimum conical diffuser and constant diameter sections.

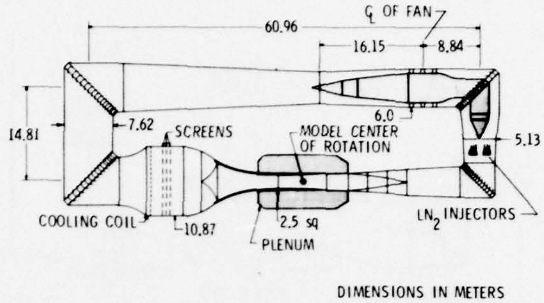


Figure 32. Lines of the aerodynamic circuit of the National Transonic Facility.

In order to conserve time and energy, provision will be made to isolate the test section from the rest of the circuit so that model changes can be made in the test section in a "shirt sleeve" environment while the rest of the circuit remains pressurized and cold. A sketch of the test-section isolation system is shown in Figure 33. Isolation will be accomplished by retracting from the test section portions of the contraction section and the high-speed diffuser and swinging isolation valves into position against the ends of the test section. After reducing the pressure in the test section to atmospheric, the side walls will be lowered and an access tunnel inserted to surround the model and sting. Heaters will be used to warm the model while air is circulated through the access tunnel to provide the required shirt sleeve environment. When work on the model is completed, the access tunnel will be removed, the side walls raised, the test section pressure made equal to the pressure in the rest of the tunnel, the isolation valves removed and, finally, the contraction section and high speed diffuser reconnected to the test section.

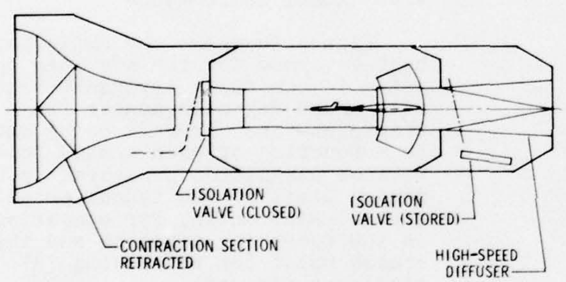


Figure 33. Schematic of the test-section pressure isolation system for the National Transonic Facility.

4.3 Fan Drive System

As noted in Reference 12, the existence of the additional variable of temperature does not affect the aerodynamic behavior of the fan. However, since the speed of sound varies with temperature, it is necessary to vary either fan speed or fan geometry in order to have independent control of test section Mach number over the wide range of operating temperatures available in a cryogenic tunnel. As explained in Reference 37, the desired performance of the NTF will be obtained by using a combination of variable speed and variable geometry.

A single stage fan having variable inlet guide vanes, a fixed geometry rotor, and fixed geometry outlet stators will be used in the NTF. As shown in Figure 34, the fan will be driven by a shaft which is directly coupled to an in-line synchronous motor which is in turn coupled through a two-speed gear box to two variable speed motors. The gear box allows the maximum power available from the existing variable speed motors to be delivered to the fan at both ambient (high fan speed) and cryogenic (low fan speed) conditions. The gear arrangement combined with the variable inlet guide

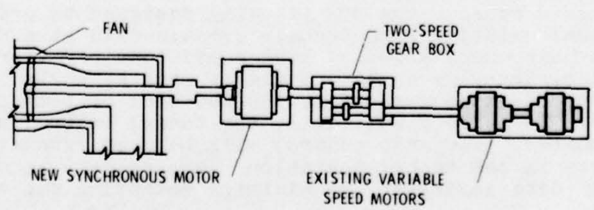


Figure 34. Schematic of National Transonic Facility drive system.

vanes will provide Mach number control at an acceptable level of efficiency over the desired range of test conditions.

4.4 Nitrogen Supply and Vent Systems

As previously mentioned, in the cryogenic tunnel concept developed at Langley, liquid nitrogen is sprayed directly into the tunnel circuit to cool the tunnel and the stream. Venting of the resulting nitrogen gas is then required to control the tunnel pressure. The liquid nitrogen supply and gaseous nitrogen vent systems for the NTF are shown schematically in Figure 35.

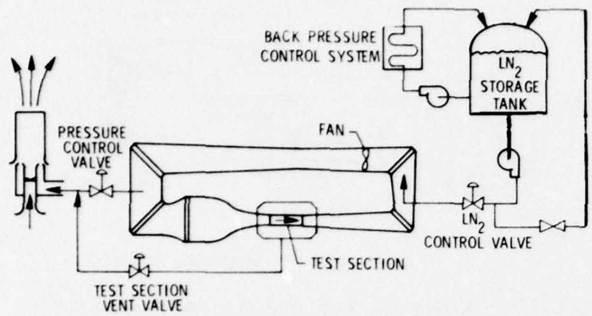


Figure 35. Schematic of the nitrogen supply and vent system for the NTF.

Pumps will be used to transfer LN_2 from the storage tank into the circuit through nozzles located between the first and second set of turning vanes. This location ahead of the drive fan will allow as much length as possible for evaporation of the LN_2 and mixing within the stream in order to insure a uniform temperature distribution in the test section. As noted by Adcock,[38] an additional advantage of injecting ahead of the fan rather than after the fan is a reduction in drive power and, therefore, a reduction in LN_2 usage by from 3 to 5 percent depending on operating pressure.

The vent stack incorporates a fan-driven ejector in order to dilute and disperse the cold nitrogen exhaust over a wide range of flow rates and pressure ratios.

4.5 Tunnel Performance

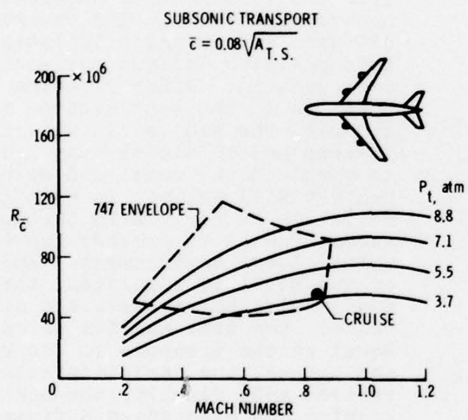


Figure 36. National Transonic Facility operating envelope.

Figure 36 shows the anticipated test envelope for the NTF when operating in the fully cryogenic mode with chord Reynolds number for transport-type aircraft being shown as a function of Mach number for various stagnation pressures up to the design limit of the tunnel pressure shell. Also shown, for comparison, is the operating envelope and the cruise point for the Boeing 747 transport aircraft.

The NTF will provide a maximum Reynolds number of 110 million at near-sonic speeds — about 10 times the current capability in the U.S. and about twice the Boeing 747 cruise Reynolds number. Even at a reduced pressure of 6 atm, the tunnel can simulate full-scale conditions at near-sonic speeds for future aircraft having sizes as much as 40% larger than the 747 with dynamic pressures no greater than 190 kN/m^2.

4.6 Productivity

As explained in Reference 37, as a national facility the NTF must meet the high Reynolds number testing needs of NASA, the DOD, the aerospace industry, other Government agencies and the scientific community. As a consequence of this, as well as the need to conserve energy, the NTF is being designed to produce data at a relatively high rate. Typical existing wind tunnels produce data at about 26,000 specific sets of test conditions in a year where a set of test conditions per year is defined by a combination of Mach number, Reynolds number, angle of attack, angle of yaw, and so forth. The NTF is targeted to produce measurements at 104,000 sets of test conditions or four times the conventional rate. To achieve this goal, the tunnel control and data acquisition system will be highly automated. Computer control will be used extensively to insure optimum procedures and safety in the tunnel operation. Modern data acquisition will be provided with "quick look" data capability to minimize retesting due to improper measurements.

The anticipated high efficiency and productivity of the NTF, combined with the unusual ability to independently control Reynolds number, Mach number, and dynamic pressure, and the availability of run times sufficiently long to allow for all types of tests, together define a tunnel that should uniquely satisfy the high Reynolds number test needs in the U.S. at high-subsonic and transonic speeds for many years.

5. CONCLUDING REMARKS

Based on theoretical studies and experience with a low-speed cryogenic tunnel and with the Langley 0.3-meter Transonic Cryogenic Tunnel, the cryogenic concept has been shown to offer many advantages with respect to the attainment of full-scale Reynolds number at reasonable levels of dynamic pressure in a ground-based facility. The unique modes of operation which are available only in a pressurized cryogenic tunnel make possible for the first time the separation of Mach number, Reynolds number, and aero-elastic effects. By reducing the drive-power requirements to a level where a conventional fan-drive system may be used, the cryogenic concept makes possible a tunnel with high productivity and run times sufficiently long to allow for all types of tests at reduced capital costs, and for equal amounts of testing, reduced total energy consumption in comparison with other tunnel concepts.

The Langley Research Center has been selected as the site of a new fan-driven high Reynolds number transonic tunnel to meet the research and development needs of NASA, the Department of Defense, and industry. The new tunnel will take full advantage of the cryogenic wind-tunnel concept, will have a 2.5- by 2.5-m test section, and will be capable of operating at stagnation pressures up to 8.8 atmospheres.

This new tunnel, to be known as the National Transonic Facility, will provide chord Reynolds numbers, based on a wing chord equal to 0.1 times the square root of the test-section area, of 135 million at transonic speeds which is an order of magnitude increase in Reynolds number capability over existing transonic tunnels in the United States.

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THE CRYOGENIC WINDTUNNEL; ANOTHER OPTION FOR THE EUROPEAN TRANSONIC FACILITY

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SUMMARY

A new option for the proposed European transonic windtunnel is described: a cryogenic facility with test-section dimensions compatible with existing major European transonic facilities. The tunnel performance is to the functional specification of the LaWs Group (Reynolds number based on mean aerodynamic chord variable between 25×10^6 and 40×10^6). The advantages and drawbacks of cryogenic testing as well as fundamental aspects of cryogenic aerodynamics are discussed. Comparative estimates for capital and operating costs are finally presented.

SYMBOLS

a	speed of sound
$\bar{c}$	mean aerodynamic chord
$c_{p_{rms}}$	root-mean-square pressure fluctuation coefficient
i	angle of incidence
l	typical tunnel length-dimension
M	Mach number
p_o	stagnation pressure
P	fan drive power
q	dynamic pressure
Re	Reynolds number
T_o	stagnation temperature
V	velocity
α	isentropic expansion coefficient
γ	ratio of specific heats
ρ	density
μ	viscosity

1 INTRODUCTION

The need for a new transonic windtunnel for Europe for achieving high Reynolds numbers has been well established by the Large Windtunnels Working Group (LaWs) of the Fluid Dynamics Panel of AGARD[1,2] (published in 1972 and 1974).

According to the recommendation of the LaWs Group the tunnel should have a maximum stagnation pressure of 6 bars, 10 seconds running time and the Reynolds number (based on mean aerodynamic chord) should be variable between 25 and 40 million, to enable extrapolation of test results to full-scale conditions.

Four options were proposed for the drive system of the facility : the Ludwieg Tube (LT), the Evans Clean Tunnel (ECT), the Injector Driven Tunnel (IDT) and the Hydraulic Driven Tunnel (HDT), which was not further considered in the course of the studies.

After the LaWs Group completed its work and its conclusions were endorsed by the AEROTEST Group[3], the NATO Defence Research Group's AC/243 (PG.7) continued the work and, in turn, established a Technical Working Group AC/243 (PG.7/WG.1) to prepare the technical case for the selection of the windtunnel drive system. Engineering Studies[4] were done to investigate possible savings on the earlier presented capital and operating costs for the three original options (1975). In the same year an important decision was made in the USA, namely that the National Transonic Facility (NTF) was to be built by NASA, a 2.5 x 2.5 m^2 transonic windtunnel using cold nitrogen as a test gas to achieve high Reynolds numbers. This cryogenic windtunnel concept was selected after numerous other systems (Ludwieg Tube, blowdown, injector tunnel and hydraulic drive tunnel) had been considered[5] since 1967. The feasibility of testing in nitrogen at cryogenic conditions had been demonstrated in a cryogenic transonic pilot facility at the NASA Langley Research Center, which had produced data since 1974.

After a visit to NASA (summer 1975), Technical Working Group AC/243 (PG.7/WG.1) made a short study of a cryogenic transonic windtunnel to the LaWs functional specification and came to the conclusion[6] that substantial savings in capital costs might be reached if a cryogenic facility was built, in comparison to the earlier proposed drive systems. This conclusion prompted a new set of Engineering Studies[7] to determine the feasibility of a cryogenic transonic windtunnel built to the LaWs functional specification and to establish capital and operating costs of this new option comparative to the costs of the earlier proposed options.

This paper serves the purpose of describing the facility concept as a new option for the European transonic windtunnel and to discuss fundamental aspects of cryogenic testing. Relative cost estimates which resulted from the new Engineering Studies[7] will also be presented. Technical Working Group AC/243 (PG.7/WG.1) will make a recommendation[8] on the preferred drive system by the end of 1976.

2 THE CONCEPT OF TESTING AT CRYOGENIC TEMPERATURES

As early as 1945 Smelt[9] indicated the advantages of testing at low cryogenic temperatures which included a considerable gain in Reynolds number and a reduction in drive power.

The idea was taken up in 1971 at NASA, by which time considerable development in low temperature engineering had made it reasonable to consider a windtunnel operating with a test gas at cryogenic temperatures.

The Reynolds number is defined as :

$$Re = \frac{\rho V l}{\mu} \tag{1}$$

where ρ = density

V = speed

l = reference length

μ = viscosity.

There are three ways to obtain high Reynolds numbers :

a. to test on a large model (l), which might result in a large and costly facility

b. to increase pressure (and thus ρ). A limit is here formed because of the high loads on models, balances, stings and sting supports and because of the aeroelastic deformation on the model wings, resulting in a non-representative shape in comparison to the actual aircraft.

The deliberations of the LaWs Group, where only stagnation pressure and size were considered as means to obtain high Reynolds numbers, led to the compromise of a maximum stagnation pressure of 6 bars and a test-section size of 5 x 4.2 m^2.

c. to lower the stagnation temperature (higher ρ and lower μ).

The advantage of testing at lower temperatures than ambient is well illustrated in figure 1, which was taken from reference 10. At a given tunnel size and constant stagnation pressure, the Reynolds number is increased by a factor of around 6 when the stagnation temperature is dropped from ambient to about 100K. The lower limit of stagnation temperature is determined by saturation of the test medium. Nitrogen has a lower saturation boundary than air, and the obtainable Reynolds numbers are higher.

There are other advantages of nitrogen over air, hence nitrogen was chosen for the NTF, and will also be considered here.

Depending on the functional specification for the facility a compromise has to be found between test section size, maximum pressure and stagnation temperature.

3 ADVANTAGES OF THE CRYOGENIC CONCEPT

The advantages of testing at cryogenic temperatures for the European transonic facility are the following :

(a) The size of the facility can be reduced in comparison to earlier options. As explained earlier, the required Reynolds number still necessitate a choice between stagnation pressure, test-section size and operating temperature. As will be shown later, a test-section size similar to existing major European facilities ($\sim 3.2\ m^2$) can be chosen and the reduction in size, compared to an ambient temperature option, decrease the capital costs despite the increased unit costs for cryogenic operation.

(b) Choice of a test-section similar to existing major facilities will also reduce model cost and model lead-times, in comparison to the large and expensive models for a facility with ambient temperature and test section dimensions of $5 \times 4.2\ m^2$. Increases of the order of 25% in cost and time are expected for models for a cryogenic facility in comparison to present-day models of the same size[11].

(c) The potential of testing at lower maximum stagnation pressures than 6 bars. The long recognized advantages of stagnation pressures lower than 6 bars, have been re-emphasized by studies by ONERA[12], which indicated the probability of stagnation pressure limitations to less than 6 bars for a typical fighter model, due to a high stress level in the sting.
Aeroelastic deformation of the model is also reduced by lower maximum stagnation pressure.

(d) The LaWs functional specification ($25 \times 10^6 < Re < 40 \times 10^6$) is based on extrapolation of test results to flight conditions. In the original options for the drive system variable Reynolds number is achieved by varying stagnation pressure. The model wing loads are proportional to the dynamic pressure (and thus to stagnation pressure). In effect variable Reynolds number could only be obtained on a model with variable form (due to aeroelastic distortion). This was, of course, realized by the LaWs Group but accepted as unavoidable. Cryogenic testing gives a solution to this problem; the dynamic pressure can be kept constant for variable Reynolds number through variation in stagnation temperature (at constant stagnation pressure). This method of testing is illustrated as "mode 1" in fig. 2.

(e) Stagnation pressure and stagnation temperature can also be adjusted in such a way that dynamic pressure and Reynolds number are kept constant when Mach number is varied. "Mode 2" in figure 2 illustrates this type of operation. As the only test parameter which then varies is M, this mode of operation may be useful for the determination of "pure" Mach number effects (e.g. buffet boundaries).

(f) "Mode 3" (fig. 2) illustrates yet another way of operation; dynamic pressure can be varied when keeping Mach number and Reynolds number constant. This is of high interest for "pure" aeroelastic studies, to determine the influence of model deformation on the aerodynamic behaviour. The wing deformation can possibly be better matched to that of the real aircraft.

(g) Flutter studies can now be made with variation of dynamic pressure only, without disturbing side effects from variation of Mach number and/or Reynolds number.

(h) For flutter tests complete scaling (Mach and Froude) is possible[13,14] for aircraft components at reduced scale, when stagnation temperature can be varied as in a cryogenic facility. Apart from the gain in Reynolds number, this gives for the first time the possibility of performing flutter tests using the same model to simulate correctly a complete range of altitudes of practical interest[13].

(i) Finally, a continuous transonic facility suitable for operation at cryogenic temperatures is a flexible concept. It can be designed in such a way that operation at ambient temperatures is possible. As such, the facility can be used as a present-day windtunnel for those tests which do not need high-Reynolds-number simulation. Depending on the drive power installed the facility can be operated at ambient temperatures and give a performance comparable to the best present-day windtunnels (about 10×10^6 for complete models).
Moreover, the continuous facility has no inherent limitation on testing time, which can be favourable, for instance, for flutter tests[13].

4 DRAWBACKS OF TESTING AT CRYOGENIC TEMPERATURES

Beside the numerous advantages of cryogenic testing some drawbacks exist which can be indicated and commented upon as follows :

(a) As mentioned in Section 2, testing at cryogenic temperatures results in a considerable reduction in fan drive power. For a selected test medium the following relationships are valid[15] for Reynolds number (Re) and fan drive power (P) :

$$Re \sim p_o\, l\, T_o^{-1.4} \qquad (2)$$

$$P \sim p_o\, l^2\, T_o^{0.5} \qquad (3)$$

where l = typical tunnel dimension

p_o = stagnation pressure

T_o = stagnation temperature.

For given tunneldimension and stagnation pressure the following relationship holds :

$$P \sim Re\, T_o^{1.9} \qquad (4)$$

Thus, for this example, if stagnation temperature is lowered from ambient to say, 100K the required fan drive power is reduced by a factor 0.12 for the same Reynolds number, resulting in considerable required energy reductions for the fan.

However, it has to be realized that cryogenic conditions can only be obtained by means of cooling with large quantities of liquid nitrogen. This required LN_2 is not only determined by the time that test data are gathered, it is mainly needed[7] to cope with other factors as facility transients from test condition to test condition, fan starting and stopping, and losses (insulation; distribution and facility cool-down). Thus, for testing at cryogenic temperatures, there are large energy requirements for providing LN_2, although fan drive power is considerably reduced. This will be further illustrated in Section 7.5 of this report.

(b) Model design and manufacture will be more complicated (and thus more costly) than for present-day models in existing facilities. However, discussions between experts in the USA on models for NTF[14] indicate that the main concern for those models is the high aerodynamic loads at high Reynolds numbers (maximum Reynolds number of 120 million at a stagnation pressure of 8.9 bars) and not the complications of the cryogenic environment. The same conclusions are reached in reference 11. Nevertheless, the cryogenic environment will necessitate the use of steels which are existing but which are new to the model designer, thermal stresses may give problems and probably the same, or nearly equivalent, materials have to be used throughout the model.
General opinion[11,15,16] appears to be that these problems can be overcome in due time, through development. For a European cryogenic facility the model loads are less severe than in the NTF due to the much lower maximum stagnation pressure; but sophisticated stress calculations and design, more quality control and inspection of models will probably be necessary. This would make the models more expensive, but with experience gained after the fabrication of a few models the cost are expected not to be more than about 25% higher than apresent-day models[11] of the same size.

(c) Model handling is more complicated in a cryogenic facility than in present-day tunnels. It is clear that it is impossible to make neither the smallest modification nor any adjustment which cannot controlled from a distance to a model in the surroundings of cryogenic nitrogen gas which is hostile to the human being. So,special measures have to be taken to bring the model in an acceptable environment before work can be started. Elaborate systems have been conceived, both for NTF as well as in the DSMA Engineering Studies[7] to gain access to the model. Such systems are expensive and the time to gain access to the models are expected to be long : the sequencing of events will take something like an hour for model warming up and cooling down. The system as conceived by DSMA[7] will be discussed in Section 7.2 of this report, its feasibility has been indicated by preliminary engineering design.

Notwithstanding the long model sequencing times required, a productivity can be foreseen for the cryogenic facility which is at least equal to that of the other options of LEHRT which were considered earlier[7].

Checking of models and on-board instrumentation will be required before installation in the tunnel, hence additional cooling chambers will be needed[7,18], with consequent effects on model process time.

(d) The design of balances, transducers, scanners etc. for operation in a cryogenic facility will require development. Experts agree however[16,17,18], that no insoluable problems for performing accurate measurements are expected. Substantial experience both in research laboratories and industry with cryogenic technology is existing and could be tapped.
Validation tests[19] with a balance for the 1/3-m cryogenic facility at Langley Field have shown it to be feasible to design and use fruitfully a heated balance for force measurements. Insulation for highly loaded balances in slender bodies might not prove feasible however[18]. It is therefore comforting that tests with the mentioned balance produced, when the temperature was not controlled on purpose, surprisingly good results although no special means were taken either to design the balance for low temperature or to prevent temperature gradients[19].

5 FUNDAMENTAL ASPECTS OF CRYOGENIC AERODYNAMICS

5.1 Minimum operating temperatures

Due to the sensitivity of both Reynolds number and fan-drive power to stagnation temperature is it extremely important to know the minimum temperatures for operation of a cryogenic facility. At low temperatures condensation effects could have adverse effects on the measurements. Condensation is expected first in the regions on the model with lowest pressure, however time and length considerations are also of importance.

To determine the effects of liquefaction of nitrogen on pressure measurements a NACA 0012-61 airfoil was tested in the NASA 1/3-m cryogenic facility[20,21,22]. The model was tested at zero-degree angle of incidence, under variable stagnation temperature and stagnation pressure conditions, keeping the Reynolds number constant ($M = 0.75$, 0.85 and 0.95).
The analysis of these measurements shows that for this airfoil, stagnation temperature can be lowered a few degrees below free-stream saturation temperature before any adverse effects on measurements are detected. Strictly this conclusion is valid only for the profile tested (which has relatively low maximum local Mach numbers), with its chord length of 13.7 cm. Studies of length effects (airfoils with various chords) and measurements on other profiles are planned for 1977 at Langley Research Center, NASA.

A greater caution in the assignment of a minimum operating temperature is appropriate for the European transonic facility than for NTF. In the event of it proving necessary to use higher minimum temperatures than originally supposed, a shortfall in maximum Reynolds number will occur. The NTF is aimed at providing Reynolds numbers of 120×10^6. A small reduction in maximum Reynolds number from this value will only mean that modest extrapolation of wind-tunnel data will be necessary in some cases.
On the other hand the European transonic facility has a maximum Reynolds number of 40×10^6 so that extrapolation to full scale will be required in virtually all cases.
Moreover, this maximum was fixed as the minimum value that would ensure reliable extrapolation to full scale. Hence, a shortfall of maximum useable Reynolds numbers below 40×10^6 will materially degrade the effectiveness of the whole facility.

For this reason, and pending reliable data, the minimum stagnation temperature for the European transonic facility has been conservatively taken as that corresponding to saturation at a local Mach number of 1.7.

5.2 Real-gas effects

Cryogenic nitrogen departs from ideal-gas behaviour because of thermal ($p\rho^{-1} \neq RT$) and caloric ($\gamma \neq 1.4$) imperfections; γ can be as high as 1.5 under the pressure and temperature conditions of interest to windtunnel testing.

To obtain an impression of the importance of these departures from ideal-gas behaviour, NASA has calculated[23] isentropic expansions and normal-shock relations in cryogenic nitrogen. Pressure, density, temperature and area ratios were calculated for the real gas, and subsequently compared to the ideal-gas solutions. The differences found were insignificant in the region of interest[23]. It was shown that the gas behaves almost like an ideal gas, and that in the pressure-density relationship, which can be written as :

$$p = C\rho^{\alpha} \tag{5}$$

α is very near to 1.4, the value for an ideal diatomic gas.

The equation of state used in these evaluations was that of Jacobsen[24], where 32 parameters are used to describe the relationship between pressure, density and temperature. 32 parameters are needed for the temperature and pressure ranges considered (63K to 2000K and pressures up to 10000 bar). Because of the lack of sufficient experimental data in the region of interest for windtunnel testing, and because of the use of a 32-parameter equation in future calculations might prove time-consuming for computer use, the calculations of isentropic expansions and shock-waves were done at NLR[25] using the Beattie-Bridgeman equation of state (only 4 parameters). The results of these calculations were identical with the results obtained by NASA, they confirmed negligible real-gas effects in expansions and shock-waves at cryogenic conditions (p_o up to 10 bars, T_o down to 100K).

Viviand[26] showed that, using the transonic small perturbation assumption, the perturbation potential for a real gas obeys the same equation as that for a perfect gas with a constant "effective γ" which he defined and which appeared to be very near to the ideal gas value of 1.4.

Albone[27] has indicated that if the pressure-density relationship in a gas can be presented by equation(5), with α very near to the ideal-gas value of 1.4, the Bernouilli equation used in flow calculations takes an identical form as that for an ideal diatomic gas (γ is replaced by $\alpha \cong 1.4$). As the potential equation is not influenced by real-gas behaviour, Albone concludes that the solution of the potential and Bernouilli equation in cryogenic nitrogen is the same as for an ideal gas.

Hartzuiker and Kooi reached the same conclusion[28], they have calculated the influence of "effective γ" on the pressure distribution on a supercritical airfoil. Although real-gas effects on isentropic expansions were found to be insignificant, the question might be posed whether integration of these effects in the flow field might not give rise to appreciable changes in sonic line and pressure distributions. The "effective γ" (or α) was shown to vary in the flow field from 1.39 to 1.40 under the cryogenic conditions of interest (T_o = 120K and pressure up to 10 bar). As it was impossible to introduce variable real-gas effects into available computing programs at short notice, a value of 1.39 was chosen: the real-gas effects should thus be less than those calculated in this simplified way.
A 16.5%-thick supercritical airfoil was selected at design point condition (M = 0.72), which was known to be very sensitive to Mach number and incidence variations. The calculations showed that there are slight differences in sonic-line position and pressure distribution between γ = 1.39 and 1.40. The changes were however insignificant, similar deviations were obtained by variation in incidence or Mach number ($\Delta i < 0.02$ degree, $\Delta M < 0.0016$), well within the measuring accuracy. Variable γ in the flow field would have introduced even smaller effects.

It is interesting to note here that the real-gas effects which can be expected due to the fact that the "effective γ" varies between 1.39 and 1.40 in the flow field, are of the same order but opposite to those under ambient temperature conditions and a pressure of 6 bars ($\gamma_{eff} = 1.408$)[23,28].

Pressure distribution calculations on an 12%-thick supercritical airfoil were also done by DFVLR[29], resulting in the same answer : real-gas effects in cryogenic nitrogen on two-dimensional airfoils are insignificant.

Viviand[30] used a time dependent method, with complete Euler equations and exact boundary conditions, to calculate the flow field over an NACA 0012 airfoil at M = 0.8 with different γ. Pressure, Mach number and temperature distributions appear to be insignificantly influenced by γ variations in the region of interest.

Wagner and Schmidt[31] used the Beattie-Bridgeman equation of state and the transonic small-perturbation theory to calculate the pressure distributions at M = 0.9 on a swept transonic wing for ideal- and for real-gas conditions which are of interest for a cryogenic windtunnel. The pressure distributions produced for different spanwise wing locations show a minor influence of real-gas effects on pressure distributions and shock locations, far within measuring accuracy. This is also the case for the normal-force distribution over the wing span. In the same report calculations will be presented on NACA-0012 pressure distributions and on laminar shock-wave boundary-layer interaction (Hakinen experiment). Unfortunately, no data are available to date.

Summarizing, it can be concluded that real-gas effects in cryogenic nitrogen are of the same order of magnitude as those in air at ambient temperature and at a stagnation pressure of 6 bars, but of opposite sign. Such real-gas effects are insignificant as was illustrated by means of isentropic-expansions and normal shock-wave calculations and by pressure-distribution calculations on airfoils and on a transonic wing.

6 EXPECTED FLOW QUALITY

The flow quality in a cryogenic windtunnel is mainly dependent on the design of the tunnel circuit and the test section, and will not be dependent on the temperature level provided that a good temperature distribution can be obtained.

Only very limited data on pressure fluctuations in a continuous cryogenic facility are available at present. Figure 3 presents the available results from measurements of broadband pressure fluctuations in the test section of the Langley pilot transonic cryogenic tunnel, which are derived from data in ref. 32. The measurements were made with a 2-dimensional model under angle of incidence mounted in the test section, and thus do not necessarily represent empty test-section noise. Comparison with available data at M = 0.8 from conventional continuous facilities[33] indicates a fairly low level ($C_{p_{rms}}$-values from 1 to 2.5% are found in the existing tunnels). Figure 3 indicates that there are no adverse effects of cryogenic operation on broadband pressure fluctuations.

The criteria[34] developed for pressure-fluctuation spectra for the earlier options of LEHRT (Ludwieg Tube, Evans Clean Tunnel and Injector Driven Tunnel) are based on the philosophy of obtaining test results in 10 seconds running time with the same or better accuracy as in present-day continuous tunnels. As such, the flow quality requirements could be somewhat relaxed for a continuous cryogenic facility. On the other hand the high LN_2 consumption rate during testing will necessitate measurements to be done as quick as possible. Hence, it will only be prudent to aim at the same level of allowable pressure fluctuations for a continuous cryogenic facility as for the other options or, at most, a factor of 2 higher. The latter requirement corresponds to taking full advantage of the smaller size (and, hence, higher natural frequencies) in terms of relaxed standards of flow quality without degradation of data acquisition rates. Comparison of these requirements with results from existing continuous facilities indicate that in the region of reduced frequencies of interest for the various types of tests there exist several tunnels that have fluctuation spectra similar or approaching the requirements[33]. This indicates that by careful design of a new facility the requirements for allowable pressure fluctuations can be met in the reduced frequency domain of interest.

As explained in reference 33, the contribution to the broadband noise level at the higher reduced frequencies primarily originates from the test sections, fans, model supports and diffusors. The contribution from the test section is inherent to the wall configuration that will finally be chosen. Contributions from model supports and diffusors can be minimized again by careful design (e.g. operation with a sonic throat downstream of the test section). As far as fan noise is concerned, attention will have to be paid during the fan design to the requirement of low noise level[7].

In conclusion, it can be expected that the noise from the circuit and the fan in a cryogenic continuous facility can be reduced to the desired levels through careful design and, possibly, through careful study and optimization in a pilot facility. The same conclusion is reached by Mabey[34] in a recent paper where it is concluded : "Careful attention to the design of every element of the circuit is required to achieve low levels of flow unsteadiness in the working section of a transonic wind tunnel. It is possible that the levels of flow unsteadiness in future transonic tunnels will be determined primarily by the working section design, rather than the drive system adopted".

As noted in reference 33, very little is known about the turbulence level in transonic test sections, due to measurements and interpretation difficulties (concomittant pressure fluctuations). It can be expected however that the turbulence level in a cryogenic facility can be made low (of the desired order of 0.1%) through anti-turbulence devices (screens) and a sufficient large contraction ratio from settling chamber to test section.

When cooling by means of injection of LN_2 is applied for a cryogenic facility some concern might exist about the temperature distribution and fluctuation in the test section. Tests at Langley in the pilot transonic cryogenic tunnel[32] have shown that the temperature distribution obtained with an unsophisticated injection system is comparable with present-day good continuous facilities.

Measurements of temperature fluctuations are further necessary to show that concern expressed about thermal turbulence is unwarranted in the cryogenic concept.

7 A CRYOGENIC FACILITY TO THE LAWS FUNCTIONAL SPECIFICATION

7.1 The LaWs functional specification

The LaWs Group[1,2] has applied the philosophy of extrapolation of data obtained in the windtunnel to full-scale Reynolds numbers in flight. Reliable extrapolation is possible only when the flow over the model is representative of that of flight, it was concluded that the Reynolds number based on model mean aerodynamic chord should be varied between 25×10^6 and 40×10^6.

The LaWs Group only considered ambient stagnation temperature facilities and it defined a maximum stagnation pressure of 6 bars, determined by model and model support considerations. These Reynolds number and stagnation pressure requirements led to a test-section size of 5×4.2 m^2.

The cryogenic concept introduces a new parameter into the consideration : the stagnation temperature. Keeping the Reynolds-number requirements as defined by the LaWs Group (variable from 25×10^6 to 40×10^6 at design point M = 0.9), a choice has to be made either on test-section size or on maximum stagnation pressure.

For the present studies of a cryogenic option of the European transonic facility the test-section dimensions were fixed, the maximum stagnation pressure is then determined by the minimum-operating temperature envisaged.

The test-section dimensions chosen were 1.95×1.65 m^2, this being the minimum size and, hence, lowest cost compatible with the representation of adequate detail in models for project development tests.

7.2 Description of the facility

The Engineering Studies by DSMA[7] were based on a fan-driven tunnel concept. An artist's conception of the facility studied is presented in figure 4. The airline consists of a conventional circuit, with a sonic throat aft of the test section to regulate Mach number and to prevent disturbances from downstream to enter the test section (this optional arrangement is also kept in to allow comparative estimates with earlier LEHRT options).
Cooling of the facility can be done in two ways, either by liquid nitrogen or by a conventional cooler to facilitate running at ambient temperatures, and thus lower Reynolds numbers, in an economic way (in comparison to cooling with LN_2).

Liquid nitrogen is introduced into the circuit through the first-corner turning vanes. The contraction ratio from settling chamber to test section is 23. This large contraction ratio was chosen to minimize ambient-temperature cooler losses and thus fan power and LN_2 consumption. Gate valves are situated for and aft of the test section to allow depressurization for model handling. A flexible nozzle assures good Mach number distribution at supersonic Mach numbers.

The tunnel shell is insulated from the environment by means of a cold box structure, carrying a 0.30 m-thick layer of expanded polystyrene. The space between the tunnel shell and the insulation is purged and continuously filled with nitrogen gas, maintained at a pressure slightly above atmospheric to prevent any air or moisture leakage in the space and insulation. Excessive gaseous nitrogen is blown off to the atmosphere through a 20 m high stack.
High transport costs of LN_2 necessitate the cooling system to include an air separation plant and a liquefier, to produce the required nitrogen at the site. Such systems can be bought at modest costs. (Of course, this requirement would not be there if the facility would be built next to an existing air separation plant, when LN_2 could be transported by means of pipes). The air-separation plants can run continuously day and night and require only modest power. The LN_2, which is produced at a rate of approximately 3 tonnes per hour, is stored in a LN_2-storage tank. Based on the consumption during the most demanding 2-week program that was thought likely, the tank capacity is chosen to be about 1500 tonnes of LN_2, or 2400 m^3. Maximum demand during a test run requires some auxiliary LN_2 tanks, including gas accumulators, thus assuring quick LN_2 transport to the facility.

Access to the model is gained in the way illustrated in figure 5. After a run the gate valves up- and downstream of the test section are closed and the test section is blown off. Inside a cool room, at slightly higher pressure than atmosphere, the test section and the model cart are moved side ways on rails. Subsequently the model cart is rolled backwards and the test section is put in place again. The model cart is then moved forwards, until the model penetrates the wall of the cool room into a chamber with controllable environment.

Once the model is at a workable temperature level (and in an air environment) part of the environmental room is removed, and model modification can start.

A second cool room is provided for model check-out and calibration under cryogenic conditions.

7.3 Performance

In figure 6 an operating envelope (at $M = 0.9$) is presented with the assumption that the minimum operating temperature is determined by saturation at a local Mach number on the model of $M = 1.7$.
As can be seen from figure 6 the LaWs functional specification can be met in a 1.95 x 1.65 m^2 facility at a stagnation pressure of 4.4 bars. The Reynolds number can be varied between 25 and 40 x 10^6 at constant dynamic pressure (and thus constant model deformation), enabling extrapolation of data without disturbing model deformation effects).
From model and model-support point of view the relatively low stagnation pressure is extremely favourable. The maximum power is determined by $Re = 25 \times 10^6$ at 4.4 bars (27 MW). This power enables ambient temperature operation up to 3.3 bars and up to a Reynolds number of approximately 10 x 10^6.
During the Preliminary Design Phase of the facility further studies will have to be done with regard to the minimum temperature levels and power to install. A drop in stagnation temperature from the present considered one (120K) to, say, 95K would allow the maximum pressure to be dropped still further (to approximately 3.3 bars); the installed power could be lowered to less than 20 MW. Careful weighting of ambient versus cryogenic operation will be necessary in further development stages, especially because not only capital costs are influenced, but the dynamics and control and the LN_2 consumption are penalized somewhat by the introduction of the ambient temperature mode[7].

Off-design calculations for the facility considered result in maximum Reynolds-numbers of approximately 18 x 10^6 at $M = 0.3$, 35 x 10^6 at $M = 0.7$ and 30 x 10^6 at $M = 1.1$.

For determination of the productivity, for LN_2 consumption and for cost estimates, the way of operation of the facility has carefully been considered[7]. It was assumed, for the purpose of comparative studies, that the number of test periods and their lengths would be the same as for the other options of LEHRT[2] (5000 test periods of 10s per annum). Due to the continuous fan drive 10 periods of 10 seconds are thought to be made within one test run. After these 10 periods the tunnel has to stop for model changes or modifications. In such a test run several modes of operation are possible: constant M, constant q or constant Re (see Section 3). Consideration of the dynamics and control of the facility led to the conclusion that considerable time will be needed to proceed from one test condition to another. Including starting and stopping of the facility, it is estimated that the test-runs will take approximately 6 minutes for 10 periods of data taking.

When the tunnel has been stopped, an elaborate procedure will be necessary to gain access to the model. Based on a conceptual design (see also Section 7.2) a time schedule was estimated, showing that approximately 30 minutes would be needed to gain access to the model and 25 minutes for model re-installation. Allowing 45 minutes for model modification (and a 10 minutes contingency) results in a total time between runs of approximately 2 hours.

So a typical test program day is expected to consist of 4 start-ups per day, for a period of 6 minutes, every two hours. 40 test periods of each 10 seconds per testing day are thus thought to be feasible, as in the case of the earlier options (one test period every 12 minutes).

7.4 Feasibility

The working components of the facility are large and complicated in comparison with present-day cryogenic engineering. However, it was concluded[7] that such components are feasible. The structures and mechanisms should be kept as simple as possible and actuating devices placed at ambient temperatures where possible. The conclusion was[7] that "it can be generally concluded that from a design and construction point of view the basic concept developed for the cryogenic facility is feasible".

Some problems are identified, such as the dynamics and control of the tunnel system (with its impact on LN_2 consumption and operating costs). Also fan design warrants further study.

7.5 Estimated comparative costs

The Engineering Studies by DSMA[7] have produced capital and operating costs for a cryogenic facility, comparative to the cost estimates for earlier versions of LEHRT[4]. The numbers from such studies cannot be used for budgetary purposes, therefore, to avoid confusion, the capital costs of the cryogenic facility will beconsidered here relative to the earlier LEHRT options costs. Budgetary cost estimates will be produced in a separate report[35].

The mean of the base-cost estimates for Ludwieg Tube, Evans Clean Tunnel and Injector Driven Tunnel is defined as 100% and comparative costs are presented in figure 7. As the minimum possible operating temperature was unknown at the time of the start of the Engineering Studies, two versions of a 1.95 x 1.65 m^2 cryogenic facility were studied, one with a stagnation temperature T_o = 149K and the other with T_o = 120K. It can be concluded from figure 7 that the capital investments for a cryogenic windtunnel with a test-section of 1.95 x 1.65 m^2 (including an air separation plant) will be approximately 50% of the capital costs of the earlier options.

Annual operating costs for the facility consist of labour cost, maintenance and power cost.

For operation and maintenance of the air separation plant the windtunnel staff will have to be increased by 5 persons, resulting in a slight increase in labour costs, in comparison to earlier options.

Maintenance of the facility is expected[7] to show a drastic decrease (approximately 35%) in comparison with earlier versions of LEHRT. This reduction is caused by the much smaller size of the facility. Power costs are difficult to estimate. The installed power of the cryogenic facility under consideration will be approximately 27 MW (compare 54 MW for LT, 20 MW for ECT and 52 MW for IDT). The costs depend on power consumption and on maximum demand. The power demand for the earlier options is more or less continuous, whereas the demand for the cryogenic facility will have an intermittent character, which might impose further cost penalties. This will depend on the site chosen and the power demand already existing at that site. The determination of power costs will be the subject of further studies to determine realistic operating costs.

A comparison of the energy consumption between earlier options and a cryogenic facility leads to interesting results, however (fig. 8). Here the annual energy consumption was determined for each option on the basis of a simplified testing program (all tests at the facility design point of M = 0.9 and Re = 40 x 10^6 as in the studies[1,2] of the LaWs Group). Although in this particular case the cryogenic tunnel has not to be reset between a test period of 10 secs, it was assumed that similar running times are required as determined in ref. 7 (6 minutes for 10 test periods of 10 seconds). It is shown clearly that the energy consumed by the fan drive (during testing, cooling and circulating) is a small part only of the total energy demand. The energy consumption by the LN_2 production plant (with a power of the order of 3 MW) is dominant. The dependence of energy demand on stagnation temperature is apparent for a cryogenic tunnel. The figure demonstrates that this energy demand can be expected to be in the order of twice the ECT demand, but less than that of the Ludwieg Tube and the Injector Driven Tunnel versions.

8 CONCLUSIONS

(1) Cryogenic nitrogen can be used as a test medium without detrimental effects on test results (negligible real-gas effects).
At present, there is not sufficient evidence available to determine the minimum operating temperature (condensation effects). For the time being, the conservative assumption has been made that the minimum stagnation temperature corresponds to saturation at a local Mach number of 1.7.

(2) The introduction of stagnation temperature as a new variable offers the following advantages for a transonic facility to the LaWs functional specification (25 x 10^6 < Re < 40 x 10^6) :
- test-section size can be made compatible with existing major European facilities.
- the cost of models and their lead time is reduced in comparison to the ambient temperature drive system options.
- maximum stagnation pressure can be lowered to 4.4 bars, in comparison to 6 bars according to the LaWs specification. This has the much wanted effects of lowering model and model-support loads. Model elastic deformation will be less.
- it will be possible, for the first time, to separate Mach number, Reynolds-number and dynamic-pressure (aeroelastic deformation) effects. This quality is of extreme importance, especially when prediction of flight behaviour will be based on extrapolation of windtunnel results.
- new possibilities emerge in the field of aeroelastic testing. Apart from the possibility to vary dynamic pressure keeping all other parameters constant, flutter tests can now be performed on one model simulating correctly a complete range of altitudes of practical interest, including sea level.

(3) Engineering Studies have shown the feasibility of a cryogenic transonic facility, with test-section dimensions of 1.95 x 1.65 m^2, built to the LaWs functional specification. Comparative cost estimates have shown that such a facility would have approximately 50% of the basic capital costs envisaged for earlier options.
Yearly operating cost would be influenced in such a way that labour costs would be slightly higher, maintenance would go down by one third and power costs would not be excessive, even comparable to the other options.
The productivity of a cryogenic facility will be at least the same as earlier considered options.
A fan-driven cryogenic facility has no inherent limitations on testing-period length. Ambient-temperature operation, at low running cost, is feasible.

(4) Windtunnel testing at cryogenic temperatures will introduce new challenges in model design-and-construction, balances and instrumentation. According to experts in the field, existing experience in cryogenic engineering can be tapped and problems arising can be overcome through development. It is expected that models for a cryogenic tunnel, once the first models have been built, will be only slightly more expensive than present-day models.

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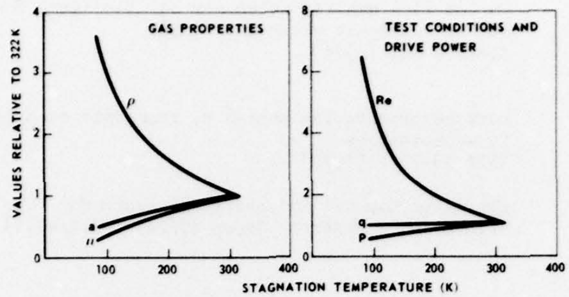


FIG. 1 EFFECT OF TEMPERATURE REDUCTION ON GAS PROPERTIES, TEST CONDITIONS AND DRIVE POWER (FROM REF. 10, FOR M = 1 AND CONSTANT STAGNATION PRESSURE AND TUNNEL SIZE)

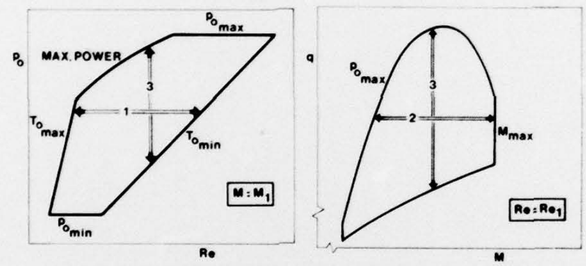


FIG. 2 THE DIFFERENT MODES OF OPERATION OF A CRYOGENIC WINDTUNNEL:
MODE 1 : REYNOLDS-NUMBER EFFECTS (CONSTANT q AND M)
MODE 2 : MACH-NUMBER EFFECTS (CONSTANT q AND Re)
MODE 3 : AEROELASTIC (q) EFFECTS AND FLUTTER TESTS (CONSTANT M AND Re)

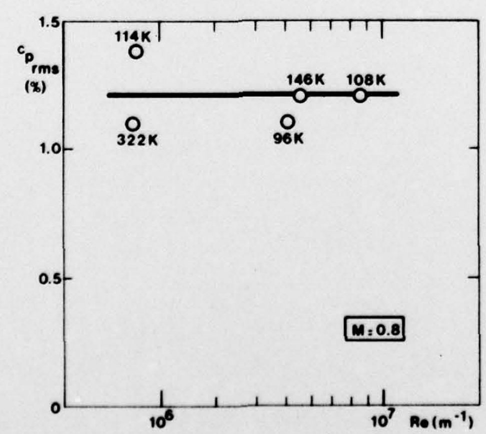


FIG. 3 BROADBAND NOISE LEVELS AS MEASURED IN LANGLEY PILOT CRYOGENIC TRANSONIC TUNNEL (CALCULATED FROM DATA FROM REFERENCE 32)

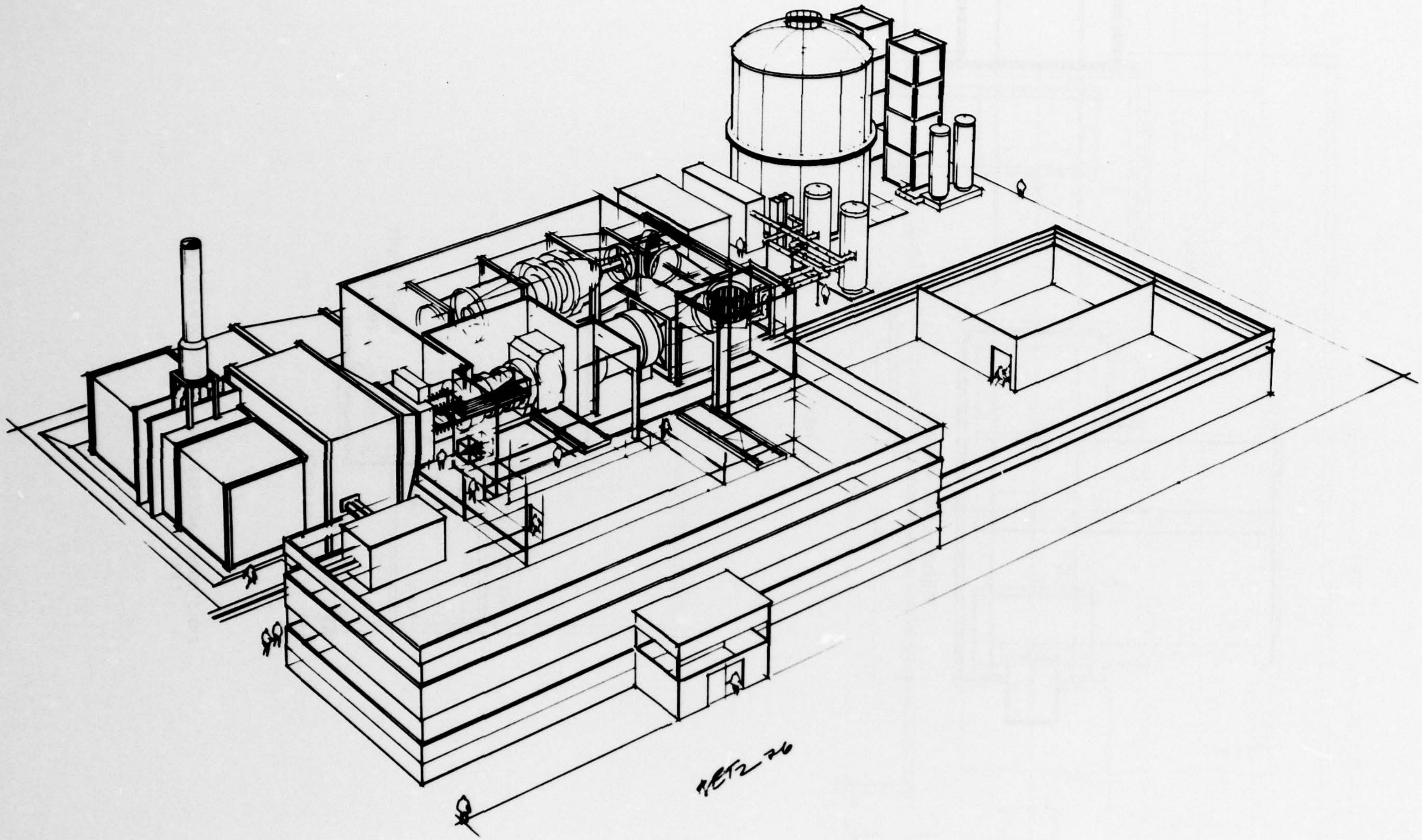
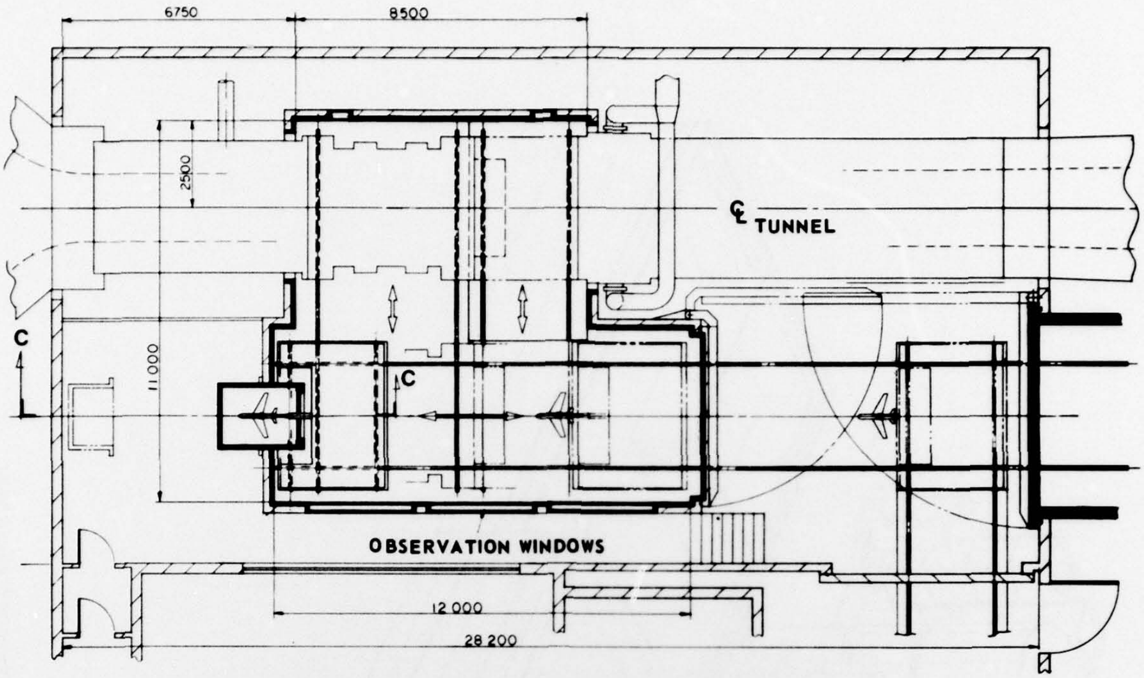
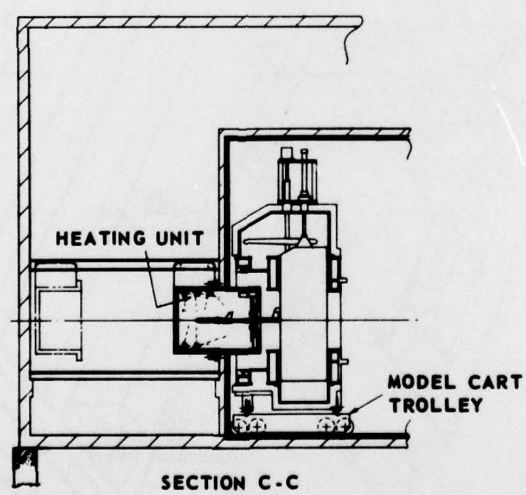


FIG. 4 ARTISTS' IMPRESSION OF A 1.95 x 1.65 m^2 CRYOGENIC WINDTUNNEL



PLAN VIEW



SECTION C-C

FIG. 5 MODEL–ACCESS SYSTEM AS DEVELOPED BY DSMA
(REF. 7)

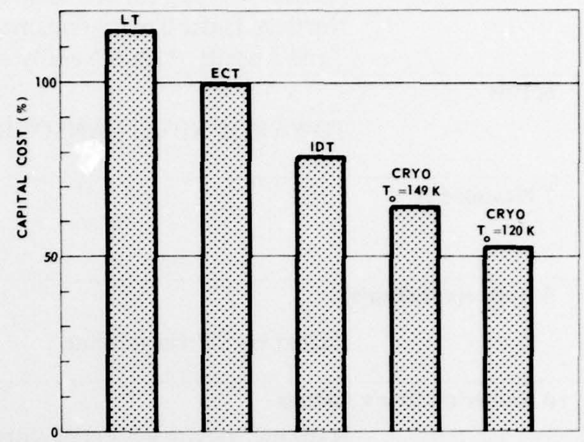


FIG. 7 COMPARITIVE CAPITAL COSTS FOR EARLIER OPTIONS OF L E H R T AND 1.95 x 1.65 m^2 CRYOGENIC FACILITIES

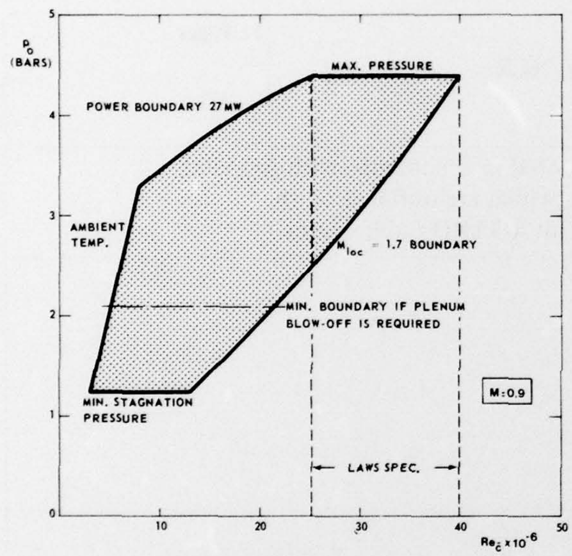


FIG. 6 OPERATING ENVELOPE OF A 1.95 x 1.65 m^2 CRYOGENIC WINDTUNNEL (CONSERVATIVE ASSUMPTION THAT LIQUEFACTION EFFECTS WILL BE FELT AT LOCAL MACH NUMBER M_{loc} = 1.7

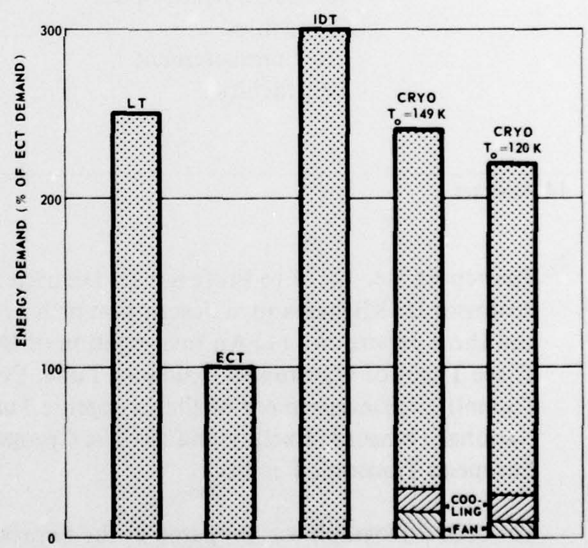


FIG. 8 COMPARISON OF ENERGY DEMAND FOR LT, ECT, IDT AND SEVERAL VERSIONS OF A 1.95 x 1.65 m^2 CRYOGENIC FACILITY (FICTIVE PROGRAM : ALL TESTS AT M = 0.9 AND Re = 40 x 10^6)

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14. Abstract

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The AGARDograph was prepared at the request of the Fluid Dynamics Panel of AGARD.

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