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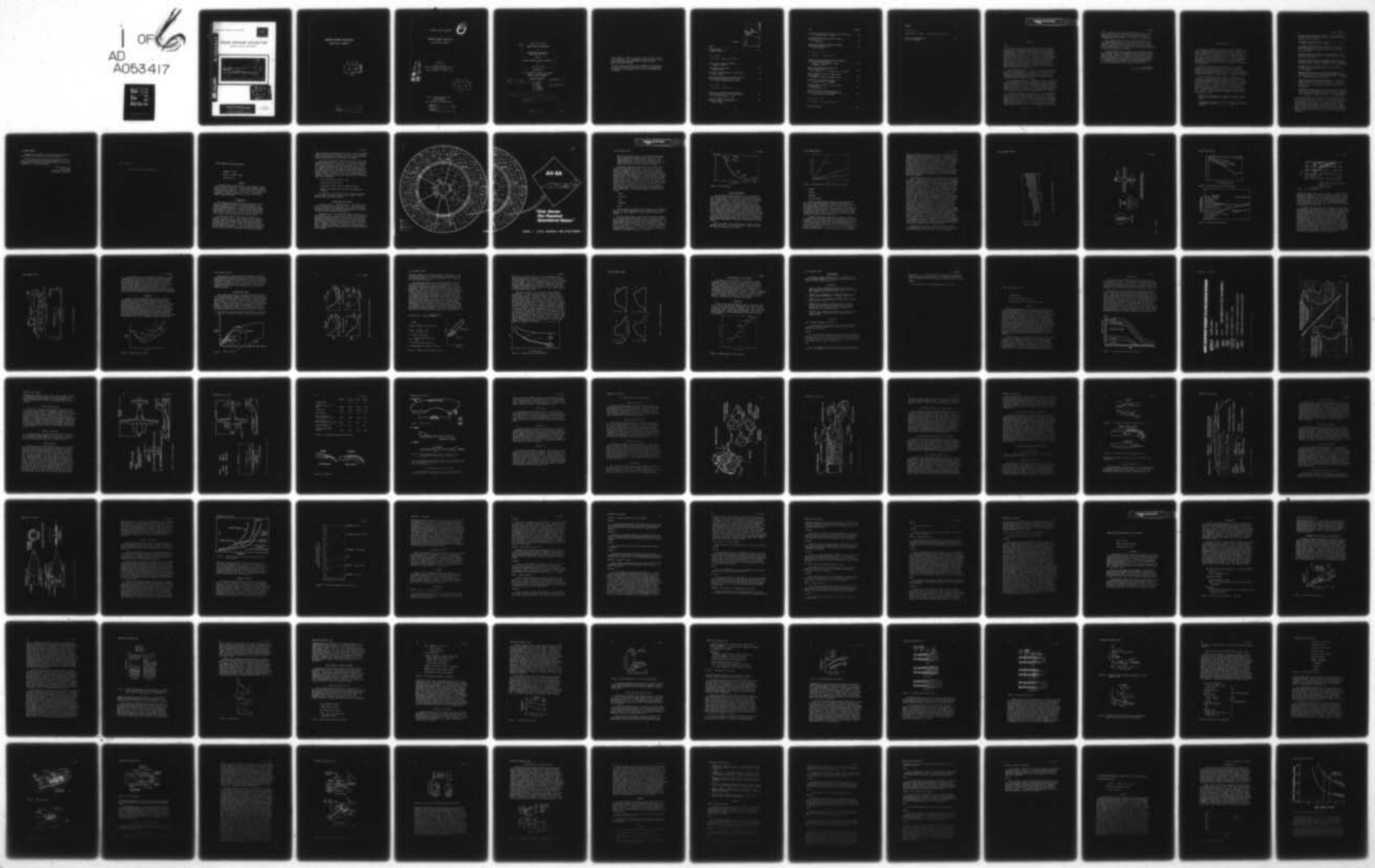

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A PROJECT SQUID WORKSHOP

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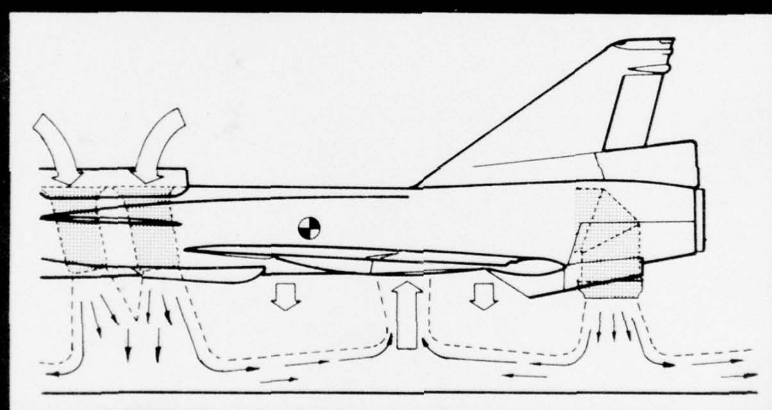
ENGINE-AIRFRAME INTEGRATION

SHORT-HAUL AIRCRAFT



Edited by
S. N. B. Murthy

ENGINE-AIRFRAME INTEGRATION

SHORT-HAUL AIRCRAFT

(12)

A PROJECT SQUID WORKSHOP

ENGINE-AIRFRAME INTEGRATION

SHORT-HAUL AIRCRAFT

Edited by

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Purdue University
West Lafayette, Indiana 47907

Project SQUID Headquarters
Chaffee Hall
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West Lafayette, Indiana 47907

Report No. PU-R1-78

6

PROCEEDINGS OF THE WORKSHOP

on

ENGINE-AIRFRAME INTEGRATION
(Short-Haul Aircraft)

held at

The Naval Academy, Annapolis Maryland

on

May 11-12, 1977,

SPONSORED BY

Air Force Office of Scientific Research
Naval Air Systems Command
Office of Naval Research

ADVISORY COMMITTEE

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SECTION V

SUMMARY AND RECOMMENDATIONS

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PREFACE

The principal concerns in the development of air-breathing propulsion, it is generally agreed, are fuel availability, engine-airframe-control integration, life-cycle reliability and environmental problems. In several respects integration presents the most challenging problems from the point of view of future airplanes, especially those in which multi-mission capability, maneuverability and special flight characteristics are demanded. Within the framework of military aircraft technology, short-haul carriers with V/STOL capability constitute one of the important classes of airplanes requiring urgent development. Integration problems are especially severe in this class of airplanes, and the solution of those problems largely determines the success of the airplanes to meet the projected mission objectives.

This Workshop was devoted to a discussion of the current status, the scope for developments and the needs in research in the general area of integration problems in short-haul, V/STOL problems. The Workshop was attended by about 63 persons who came from various institutions such as universities, industries, government laboratories and research funding agencies.

The Workshop was held under the auspices of Project SQUID (Office of Naval Research) which is devoted to basic and applied research of long range relevance to propulsion technology. It was cosponsored by the Air Force Office of Scientific Research, the Naval Air Systems Command and the ONR. This fact is of considerable significance both from the point of view of commonality of interests and from that of the close collaboration that exists between the different agencies.

In organizing a Workshop on a subject that is as comprehensive and as variously defined as integration, it was felt important to secure direct advice from those who have been involved in research and development in integration problems. An advisory committee was formed that included Dr. W. H. Heiser (Arnold Engineering Development Center), Mr. W. Koven (Naval Air Systems Command), Dr. H. W. Mark (then NASA Ames Research Center), Dr. B. A. Reese (Army Scientific Advisory Panel), Dr. Abe Silverstein (National Academy of Sciences),

and Dr. H. von Ohain (Air Force Aero-Propulsion Laboratory). The program and the format of the Workshop were discussed continuously with the Advisory Committee and it is the greatest pleasure to acknowledge here our gratitude to them for their valuable guidance.

The Proceedings have been typed by Miss Cynthia Hoffman and Mrs. Amanda Niemantsverdriet; the latter has also effectively contributed in a number of ways in the editorial work. Mr. Stanley Timmons has been responsible for a great part of the art work which had to be done skillfully. We appreciate very much their contributions in the final evolution of this volume.

Functional integration of airplanes is an evolutionary process. There are many aspects of the problem that need considerable research and development. Advances in computers and in instrumentation are having a strong impact in such studies, particularly in the area of hybrid experimentation which ultimately is the most effective means of conducting research in aeronautics including integrated control. It is hoped that this Workshop and its recorded proceedings will provide encouragement for creative activity in this field.

S.N.B. Murthy
Editor and Workshop Chairman

WELCOMING REMARKS

First of all, I wish to say how much we appreciate the very cordial welcome of Admiral Kinnaird McKee to the U.S. Naval Academy. There are representatives here today from the three Military services, NASA, other government agencies, industry, and from abroad and I know all of us are impressed with the new buildings and facilities at the Naval Academy; this lecture room and the arrangements seem ideal for a workshop such as this. In this connection, we appreciate the very great assistance provided by Professor Andrew Pouring, Department of Engineering Sciences with arrangements for the Workshop.

On behalf of the Office of Naval Research, it is a great pleasure and a privilege for me to welcome this group today and especially to see the large number of people with interest in engine-airframe integration problems. The Office of Naval Research is particularly happy to join with the Air Force Office of Scientific Research and the Naval Air Systems Command in sponsoring this Workshop.

We have planned this Workshop, as well as another one to be held this year in September on the subject of "Alternative Hydrocarbon Fuels-Combustion and Chemical Kinetics", as part of a series of workshops initiated in 1969 to emphasize problem areas in selected subjects relating to air breathing engines for aircraft and missile applications. Due to increasing demands for higher performance, smaller and lighter weight power plants, and for operation over a wider range of operating conditions, there is a need to develop deeper, fundamental understanding of the physical phenomena involved in all aspects of engine design and development. These workshops have been held to date:

- Research in Gas Dynamics of Jet Engines, ONR/Chicago, December 4-5, 1969. R. Goulard and M. L'Ecuyer, eds. Project SQUID Report.

- Fluid Dynamics of Unsteady 3-D Separated Flows, Georgia Institute of Technology, June 10-11, 1971. F. J. Marshall, ed. NTIS AD736248.

- The Use of the Laser Doppler Velocimeter for Flow Measurements, Purdue University, March 9-10, 1972; W. H. Stevenson and H. D. Thompson, eds. NTIS AD753243.

- Aeroelasticity in Turbomachines, Detroit Diesel Allison, June 1-2, 1972; S. Fleeter, ed. NTIS AD749680.

- Laser Raman Diagnostics, General Electric Research & Development Center, May 10-11, 1973; M. Lapp and C. M. Penney, eds. 1974. Laser Raman Gas Diagnostics, New York Plenum Press. Also NTIS AD782652.

- Second International Workshop on Laser Velocimetry, Purdue University, March 27-29, 1974; H. D. Thompson and W. H. Stevenson, eds. NTIS AD010223.

- Turbulent Mixing: Nonreactive and Reactive Flows, May 20-21, 1974; S.N.B. Murthy, ed. 1975. Turbulent Mixing in Nonreactive and Reactive Flows. New York Plenum Press. Also NTIS AD006322.

- Unsteady Flows in Jet Engines, United Aircraft Research Laboratory, (UARL) now (UTRC) July 11-12, 1974; F. O. Carta, ed. NTIS AD003853.

- Combustion Measurements in Jet Propulsion Systems, Purdue University, May 22-23, 1975; R. Goulard, ed. Combustion Measurements: Modern Techniques and Instrumentation, 1976. Washington, D.C. Hemisphere Publishing Corporation.

- Transonic Flow Problems in Turbomachinery, Naval Postgraduate School, February 11-13, 1976; T. C. Adamson, Jr., and M. F. Platzer, eds. 1977. Transonic Flow Problems in Turbomachinery. Washington, D.C. Hemisphere Publishing Corporation. Also NTIS ADA043317.

- Turbulence in Internal Flows, Airlie House, Warrenton, VA, June 14-15, 1976. S.N.B. Murthy, ed. 1977. Turbulence in Internal Flows, Turbomachinery and other Applications. 1977, Washington, D.C. Hemisphere Publishing Corporation. Also NTIS ADA040966.

It is not necessary to explain to this group the importance of Engine-Airframe Integration. Suffice it to say that the U.S. Navy has plans to utilize V/STOL aircraft and that we recognize integration of the vehicle with the power source is a very important aspect of the development of such systems. The subject is obviously more applied in nature than covered in other SQUID workshops. However, the technology is of particular concern to the Navy and we recognize there is a need to be concerned with research areas that would be helpful in advancing one of the key areas in this technology, namely integration.

On behalf of the sponsors, I wish to thank the organizer, Dr. S.N.B. Murthy, for arranging and planning the Workshop.

We look forward to participating in the Workshop with all of you and I wish to say that we are deeply appreciative for the contributions of the group and the time and energies of each of you, so essential to a successful workshop.

James R. Patton, Jr.
Power Program
Office of Naval Research
U.S. Department of the Navy

SECTION I

SYSTEM REQUIREMENTS

V/STOL AIRCRAFT DESIGN CONSIDERATIONS

Raymond F. Siewert

Naval Air Systems Command

Washington, D.C.

ABSTRACT

The major technical considerations in the conceptual design of V/STOL aircraft are discussed. Areas of concern are hover and low speed, transition, cruise and high speed flight, and the environment of V/STOL aircraft operations. A wide range of V/STOL concepts from helicopters to supersonic designs are included. The technical compromises necessary to achieve vertical flight and efficient forward flight are stressed.

INTRODUCTION

The design of any aircraft to do a particular job or mission represents a compromise between the conflicting demands of the various technologies striving for maximum performance, reduced weight and enhanced capability. These conflicting demands become even greater for V/STOL aircraft because of added complexity and increased weight sensitivity. It is the intent of this paper to examine some of the basic characteristics of V/STOL aircraft and their influence on an emerging aircraft conceptual design.

The term V/STOL has been widely applied to broad classes of aircraft in the past, and as a result conveys different images to different people, depending on their background and experiences. The term V/STOL (Vertical/Short Take-Off and Landing) aircraft is sometimes construed to include aircraft only capable of Short Take-Off and Landing (STOL) as well. In this paper, the term applies to aircraft capable of accomplishing a vertical take-off and landing and efficient cruise flight. Experience has shown that aircraft

capable of Vertical Take-Off and Landing (VTOL) flight will generally exhibit excellent STOL performance. Helicopters, while considered unique in many respects, are included, as they logically form one end of the spectrum of the aircraft under consideration.

The design considerations addressed in this paper reflect the conceptual design phase of an aircraft. While there are many other aspects of V/STOL aircraft design such as design-to-cost, mechanical systems integration, structural considerations, etc., which are interesting and important, they are obviously beyond the scope of a paper such as this. The conceptual design considerations to be addressed will attempt to answer a question that is often asked, "What is the best way to go V?" That is, given the wide variety of concepts available to achieve vertical flight, which one should be selected. In attempting to answer the above question, the following areas will be considered:

° Vertical Take-Off and Landing - Hover

° Transition to Forward Flight

° Conventional Flight - Impact of V/STOL Constraints

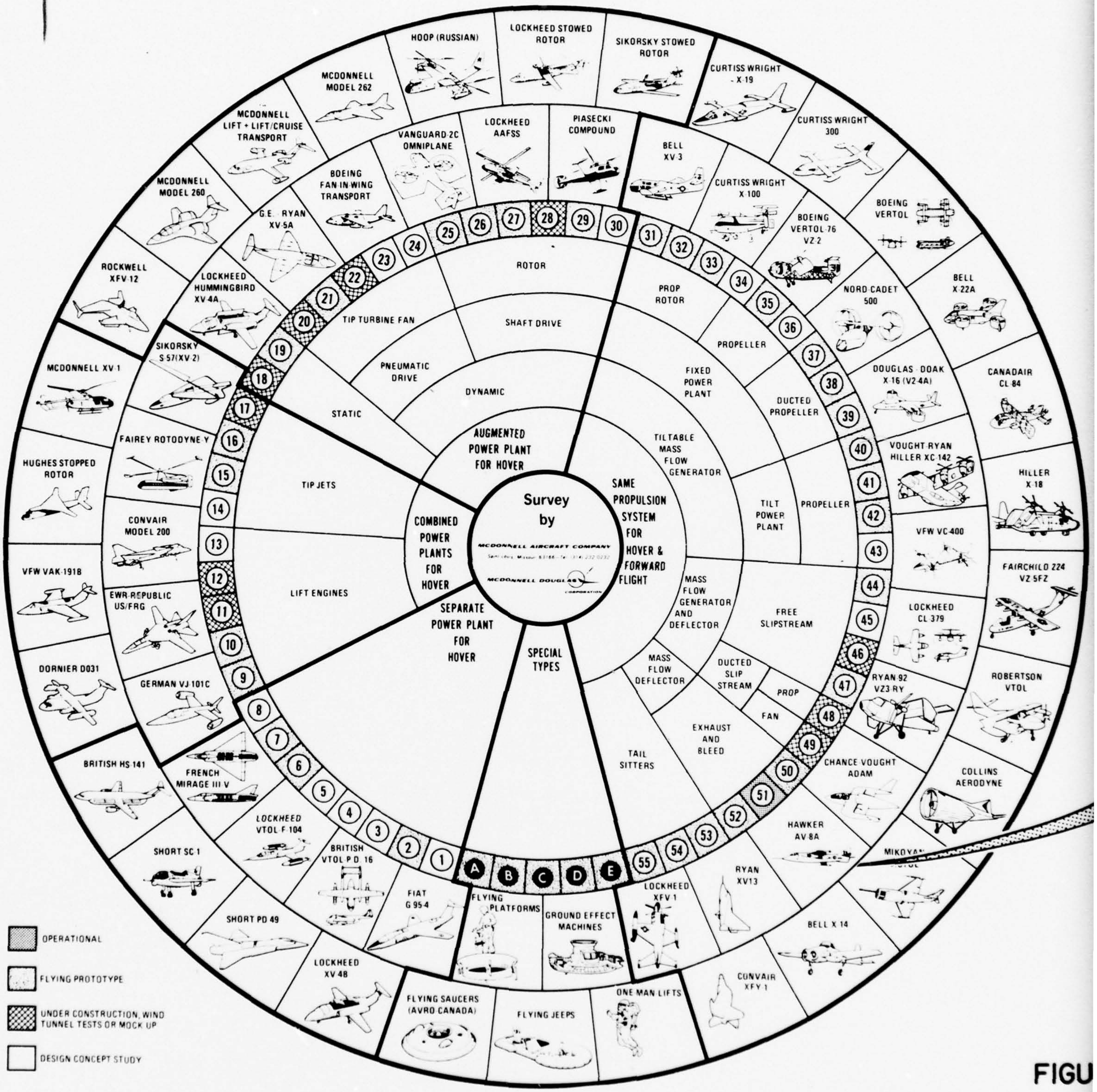
° Effect of VTOL Operations on Surrounding Equipment and Personnel.

This paper will first review the types of propulsion systems used in V/STOL, followed by discussion of the impact of these systems on various aircraft concepts.

V/STOL Propulsion Systems

As shown in Figure 1 from Reference (a), there appears to be no end of schemes to achieve vertical flight. Interestingly, only one of these concepts has reached operational status. Perhaps the reasons for this will become apparent after considering the aspects of V/STOL aircraft design as discussed herein.

The heart of any of the V/STOL schemes shown in Figure 1, is the propulsion system. For any practical V/STOL aircraft application, the propulsion system must have sufficient thrust (lift) directed downward to overcome the weight of the vehicle in vertical flight. The propulsion system must be also capable of providing sufficient thrust in forward flight to overcome the vehicle drag. This latter point is sometimes overlooked as posing any real technical challenges. However as indicated in reference (b), the requirements for vertical flight capability manifest themselves throughout the design flight envelope.



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MCDONNELL AIRCRAFT COMPANY
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AUGMENTED POWER PLANT FOR HOVER
SAME PROPULSION SYSTEM FOR HOVER & FORWARD FLIGHT
COMBINED POWER PLANTS FOR HOVER
SEPARATE POWER PLANT FOR HOVER
SPECIAL TYPES
ROTOR
SHAFT DRIVE
TIP TURBINE FAN
PNEUMATIC DRIVE
DYNAMIC
STATIC
TIP JETS
LIFT ENGINES
PROP ROTOR
PROPELLER
FIXED POWER PLANT
DUCTED PROPELLER
TILTABLE MASS FLOW GENERATOR
TILT POWER PLANT
PROPELLER
MASS FLOW GENERATOR AND DEFLECTOR
FREE SLIPSTREAM
MASS FLOW DEFLECTOR
DUCTED SLIP STREAM
PROP
FAN
EXHAUST AND BLEED
TAIL SITTERS
HOOP (RUSSIAN)
LOCKHEED STOWED ROTOR
SIKORSKY STOWED ROTOR
CURTISS WRIGHT - X 19
CURTISS WRIGHT 300
BOEING VERTOL
BELL X 22A
CANADAIR CL 84
HILLER X 18
FAIRCHILD 224 VZ 5FZ
ROBERTSON VTOL
COLLINS AERODYNE
MIKOYAN
BELL X 14
CONVAIR XFY 1
ONE MAN LIFTS
FLYING JEEPS
FLYING SAUCERS (AVRO CANADA)
LOCKHEED XV 4B
SHORT PD 49
SHORT SC 1
BRITISH HS 141
DORNIER D031
VFW VAK 191B
HUGHES STOPPED ROTOR
MCDONNELL XV 1
ROCKWELL XFV 12
MCDONNELL MODEL 260
MCDONNELL LIFT + LIFT/CRUISE TRANSPORT
MCDONNELL MODEL 262
VANGUARD 2C OMNIPLANE
LOCKHEED AAFSS
PIASECKI COMPOUND
BELL XV 3
CURTISS WRIGHT X 100
BOEING VERTOL 76 VZ 2
NORD CADET 500
DOUGLAS DOAK X 16 (VZ 4A)
VOUGHT RYAN HILLER XC 142
VFW VC 400
LOCKHEED CL 379
RYAN 92 VZ3 RY
CHANCE VOUGHT ADAM
HAWKER AV 8A
RYAN XV13
LOCKHEED XFV 1
GROUND EFFECT MACHINES
FLYING PLATFORMS
FIAT G 95 4
BRITISH VTOL P D 16
LOCKHEED VTOL F 104
FRENCH MIRAGE III V
GERMAN VJ 101C
EWR REPUBLIC US/FRG
CONVAIR MODEL 200
FAIREY ROTODYNE Y
SIKORSKY S 57(XV 2)
LOCKHEED HUMMINGBIRD XV 4A
G.E. RYAN XV 5A
BOEING FAN IN WING TRANSPORT
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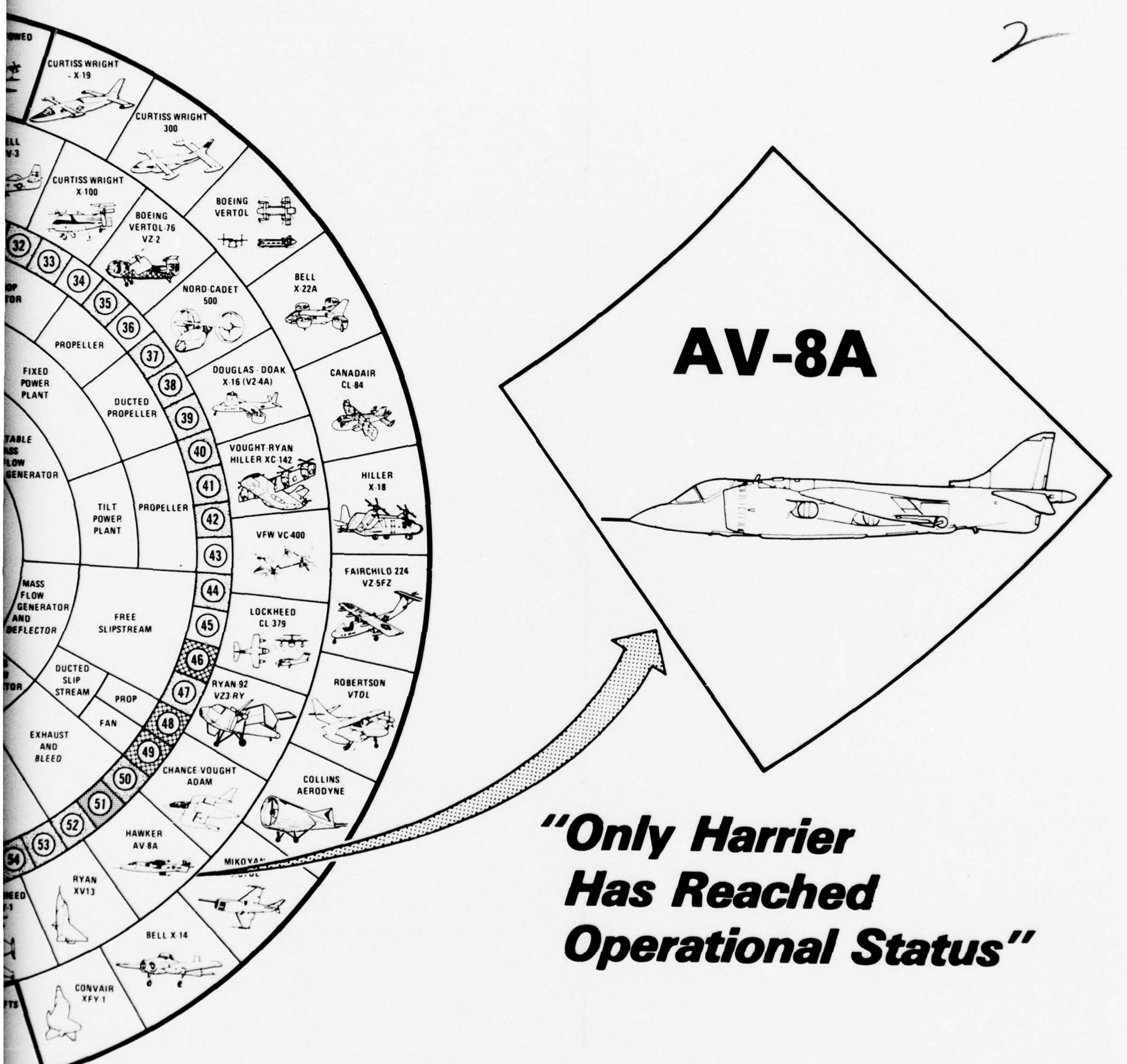

FIGURE 1. V/STOL RESEARCH AND DEVELOPMENT

Even in cruise flight, V/STOL aircraft have more complicated inlet integration requirements, lower fineness ratio, more difficult propulsion system selection requirements, more complex nozzles when compared with conventional aircraft. These characteristics can lead to significant penalties in the ability of a V/STOL aircraft to perform its mission.

Because the propulsion systems are the heart of any V/STOL aircraft, it has become common practice to classify the aircraft according to propulsion system type. However, no one seems to agree on a standard classification for these aircraft. Reference (c) utilizes thirteen classes by employing aircraft type as well as propulsion system characteristics, i.e., several different propeller classifications. The Soviets have also tried their hand in V/STOL aircraft classification Reference (c) proposes quite an elaborate scheme, nearly resulting in only one aircraft in each class. A more simple approach is used in Reference (e) wherein the author employs only two classifications, which are shaft driven and jet lift. This system results in some anomalies when discussing the characteristics of a particular class of aircraft. Not to be outdone, another V/STOL aircraft classification scheme is proposed to be used in the present paper. The propulsion concepts are classified as follows:

° Rotors

° Propellers

° Fans

° Augmenters

° Jets

These are obviously subsets of each of the major classifications such as tilt rotors, ducted propellers, deflected thrust or Harrier-type jet lift and the composite lift plus lift/cruise jet lift concept.

The above classification lists the various propulsion concepts in order of increasing "disc loading." that is, the maximum vertical thrust produced divided by an appropriate cross sectional area of the thrust producing device (rotor disc area in helicopters; total propeller disc area in tilt wings or tile propellers; total augmenter thrust area for augmenter configurations; total exhaust nozzle area for all engines for jet lift configuration). The disc loading is a fundamental design parameter. Aircraft performance can be directly related to disc loading, at least to first order.

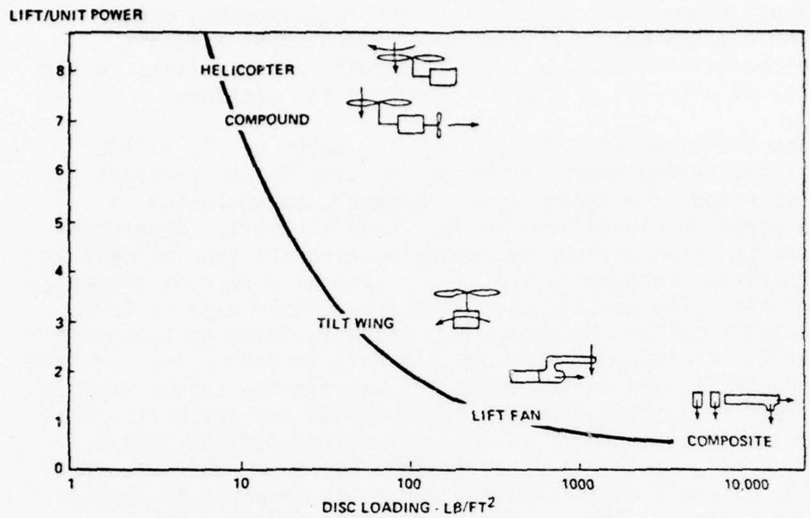

Figure 2 Lift Efficiency

Hover Considerations

The classification scheme shown above lists the concepts in order of decreasing lifting efficiency. As shown in Figure 2, as disc loading increases, the lift per unit power decreases markedly. This of course translates directly into impact on mission performance, as shown in Figure 3. Here the fuel required to hover (expressed as a fraction of the gross weight) is plotted versus hover time. Considering that operational aircraft normally only have .25 to .30 fuel fraction available, the hover requirements of the intended mission use of the aircraft are an important consideration in propulsion concept selection. A general rule of thumb is that if more than fifteen minutes of hover are required, then rotors are the only practical scheme for obtaining vertical flight. Figure 3 also indicates that even at low hover times, the fuel used by direct jet lift concepts is significant and that it is important to develop operational techniques and pilot aids to minimize the time spent in the subaerodynamic flight mode of the higher disc loading concepts.

While disc loading is of prime importance in V/STOL aircraft operation in hover, it is not the whole story. Additional factors that should be considered are as follow.

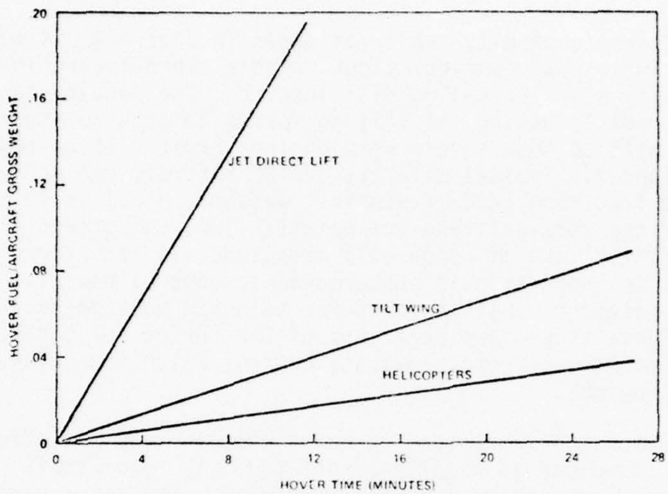

Figure 3 Fuel Required for Hover--Various VTOL Concepts

- Balance
- Control
- Download
- Ground Proximity

The first of these factors, balance, may seem so elementary that it is not worth mentioning, yet achieving balanced flight in the vertical mode is fundamental to V/STOL aircraft design and will have profound impact on all aspects of the vehicle performance. The larger the desired center of gravity travel for various loadings, the greater the balance problem. Consideration must also be given to the upset of the aircraft if one or more of the main lift system elements should fail. Rapid upset from balanced flight could well preclude safe ejection by the flight crew.

A companion installation consideration to balance is control. In flight at very low forward speeds, the conventional aircraft control surfaces are not effective, and some sort of powered control scheme is necessary. These generally fall into two broad categories. The first is through manipulation of the thrust/lift device itself, i.e., cyclic pitch of rotors and differential thrust of lift jet engines. The second is through the use of auxiliary devices, i.e., tail rotors and reaction control systems. In either case, the control requirements

impose a performance penalty which, as shown in Figure 4, is quite dependent on individual configurations, with a trend toward increasing penalty with increasing disc loading. The penalty can be minimized by redistributing the lifting forces to provide control moments and will be most severe when engine thrust must be reversed for trim (balance). Typical military V/STOL aircraft can achieve military load fractions (crew, avionics, weapons, etc.) of 10 to 20 percent of the vertical take-off weight; since the potential control "penalties" are of comparable magnitude, it is extremely important not to overdesign in subaerodynamic control power. As indicated in Reference (e), the criteria selected must be carefully chosen to balance these considerations of the "price for control" against the consequences of inadequate control which have plagued many V/STOL programs.

Installation considerations in hover also include the effects of download. Download is usually associated with rotor craft, where the rotors are placed above a fuselage or wing. The rotor downwash produces a download (or drag) on the body, hence creating an effective increase in vehicle weight. Primary parameters in determining the magnitude of the download are the relative affected area (affected area divided by the total disc area) and the shape of that portion of the vehicle which is affected by the downwash, as shown in Figure 5. A wing surface oriented normal to the flow will experience more severe download compared with that on a fuselage, whereas a wing in profile will have only a slight download. In Figure 6, these considerations are quantified in terms of the net vertical thrust (rotor thrust has download) over the isolated rotor thrust, shown as a function of the relative size of the wing to the rotor area. A conventional helicopter, i.e., no wing, experiences download on the fuselage which can vary considerably for various fuselage designs and between single and tandem rotor configurations. The addition of a wing to a helicopter has less impact than a comparable wing on a tilt rotor, since much of the helicopter wing area under the disc is also buried in the fuselage and the wing span is seldom large enough to put it under the maximum downwash velocities near the rotor tip. As can be seen from Figure 6, tilt rotor configurations are more susceptible to download penalties. Currently, tilt rotor and compound helicopter configurations usually optimize at wing chord/rotor diameter ratios of 0.1 to 0.2, resulting in 5 to 15 percent download in vertical flight. This phenomenon emphasizes the tradeoff between forward and vertical flight capability; adding or increasing the area of a wing expands the available speed-altitude envelope and improves maneuverability, but at a direct cost in vertical flight efficiency.

Generally, all V/STOL aircraft experience ground proximity effects when hovering near the ground. As shown in Figure 7, these are usually positive (increased lift) for rotor systems and

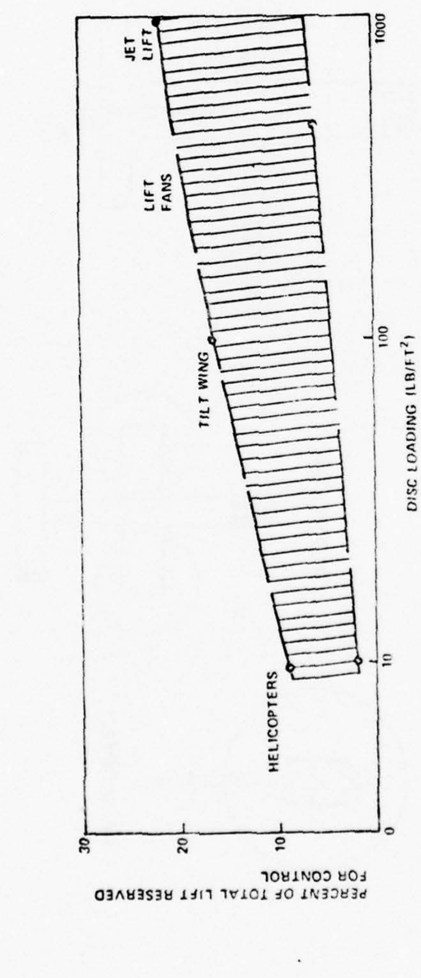

Figure 4 Impact of Control Requirements

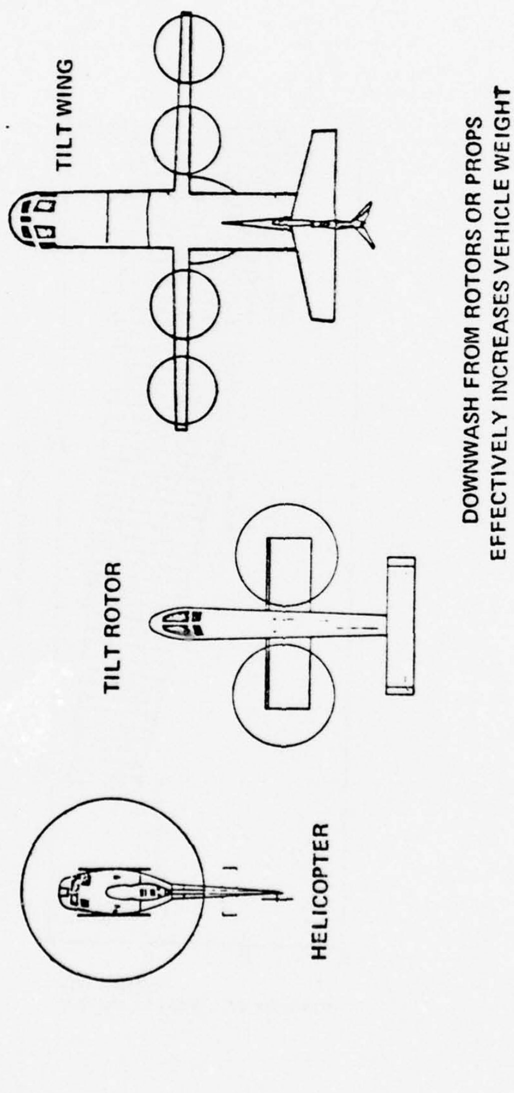

Figure 5. Download

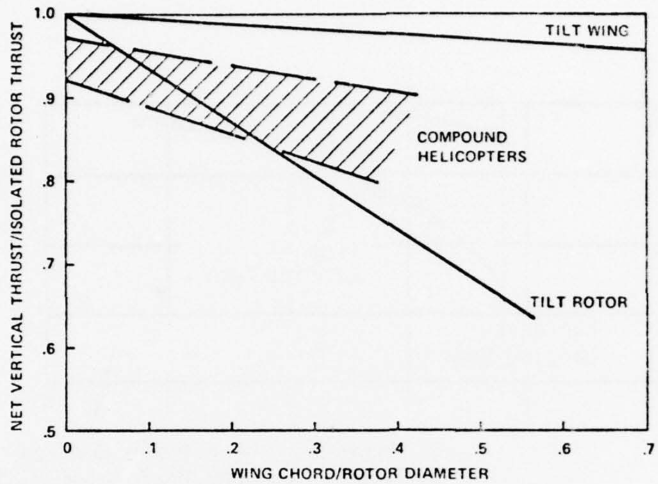

Figure 6 Effect of Download on Hover Performance

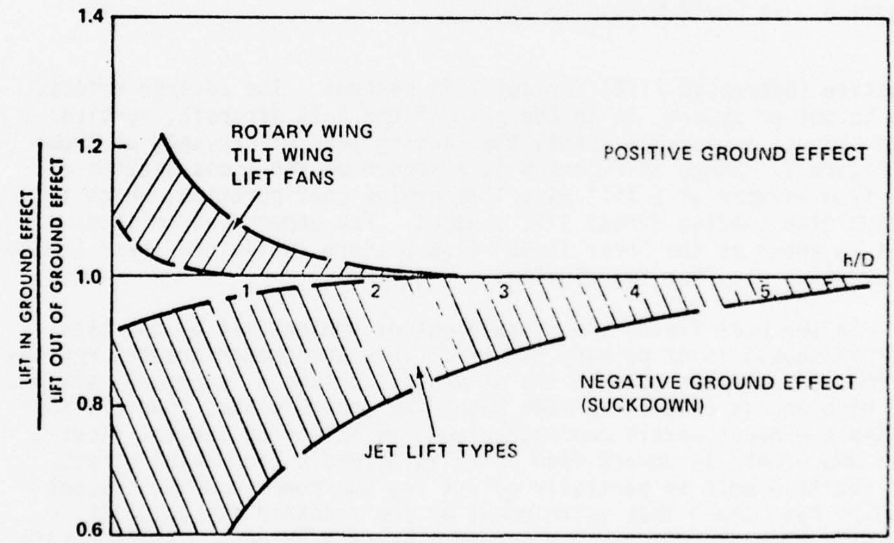

Figure 7 Effect of Ground Proximity

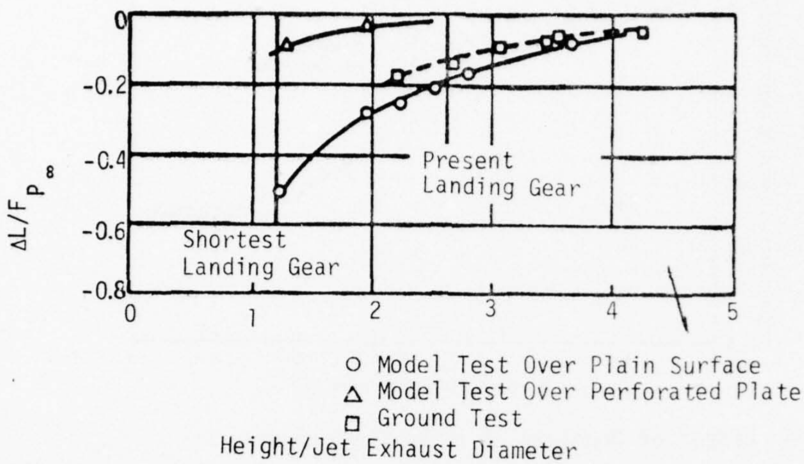

Figure 8. It Could Happen to You

negative (decreased lift) for jet lift systems. The adverse effects can become so severe, as in the case of the X-14 aircraft, vertical take-off was impossible until the landing gear was raised, as shown in Figure 8. Shown in Figure 9 is a sketch of the cross section of the flow pattern of a lift plus lift cruise configuration, which is a high disc loading direct lift concept. The upper diagram shows what is known as the "near field" flow pattern, while the lower sketch illustrates the "far field" flow.

In the near field, the entrainment of ambient air by the high energy exhaust is of primary concern. This phenomenon creates regions of reduced pressure beneath the aircraft, causing a "suckdown" effect. The high energy exhaust spreads along the ground plane; however, in multiple exhaust nozzle configurations, the spreading exhaust flows meet and create an upward flow which is termed a "fountain" effect. The fountain acts to partially offset the suckdown; however, recent studies have shown that entrainment by the fountain may actually increase suckdown at very low ground clearance heights. The fountain also carries the hot exhaust gases upward from the ground plane. If a fountain is formed in the area of the engine inlets or if the hot gases can spread toward the inlets, the engine inlet air temperature will rise with a consequent reduction in thrust or, in severe cases, actual stalling of the engine. Considerable model testing and analysis have been and are continuing to be pursued to permit reliable predictions of suckdown and/or inlet temperature rise.

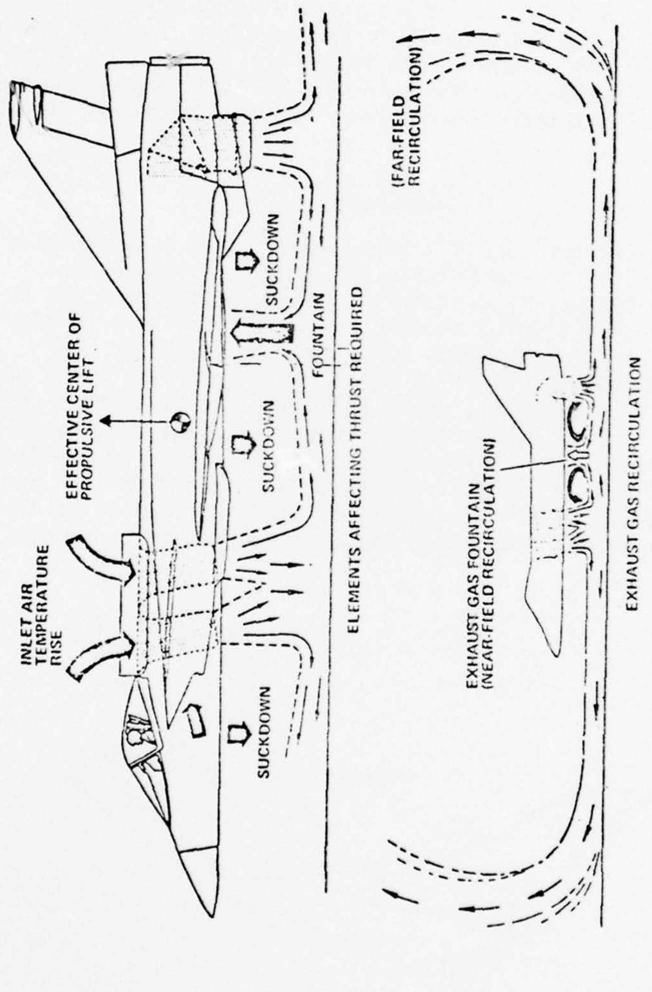

Figure 9. VTOL Exhaust Gas Effects

If a configuration can be developed with acceptable near field characteristics, then the far field phenomenon must be considered. As indicated in the lower half of figure 9, when the hot exhaust gases lose energy, they separate from the ground plane and rise, mixing with the ambient air and eventually returning to the vicinity of the aircraft. Normally by this point, mixing is essentially complete and the gases are at ambient temperature. However, if a surface wind blows these gases back toward the aircraft or if the aircraft rolls forward through them before sufficient mixing takes place, there can be an inlet temperature rise from this far field circulation.

Transition

The basic aircraft design problem encountered in transition from vertical flight to conventional flight is to ensure that sufficient excess power is available and that this excess can be directed to effect a smooth, trimmed conversion. Figure 10 displays power required at forward speed, normalized to the hover power. It can be seen that concepts such as the lift fan experience the more "narrow" conversion corridor, while tilt wing and tilt rotor configurations exhibit "wide" conversion corridors. These configurations also tend to possess the characteristic of reversible transition, i.e., the pilot can terminate transition in either direction and return to the original flight condition. As disc loading is increased, irreversibility of the transition maneuver becomes more dominant.

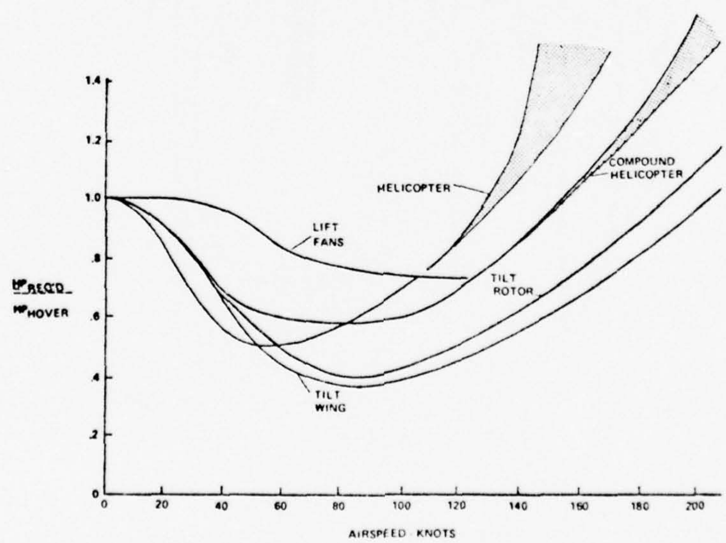

Figure 10. Power Required Vs. Speed

Other aspects of transition, while not amenable to graphical representation, are nevertheless important and should be considered. Among these are the degree of vehicle instability, particularly due to propulsion effects, the number of cockpit controls or manipulators the pilot is expected to use, and the degree and severity of cross coupling effects which may lead to abrupt upsets. These items are fundamental to any V/STOL aircraft design and must be considered from its inception.

Conventional Flight

Most of the discussion up to this point seems to auger well for low disc loaded designs. However, when forward flight performance is considered, the choice becomes less clear cut. Figure 11 presents typical forward flight speed--altitude envelopes for several of the basic concepts. It can be seen that the low disc loading vehicles have very limited altitude capability as well as low maximum forward speeds. It should be noted that these envelopes are not meant to be definitive and that each concept can be designed for a range of performance around that of Figure 11; for example, certain of the jet driven configurations can be designed for supersonic capability.

The reason for the limited forward flight velocity can be seen in Figure 12. In all cases the parasite drag of the configuration

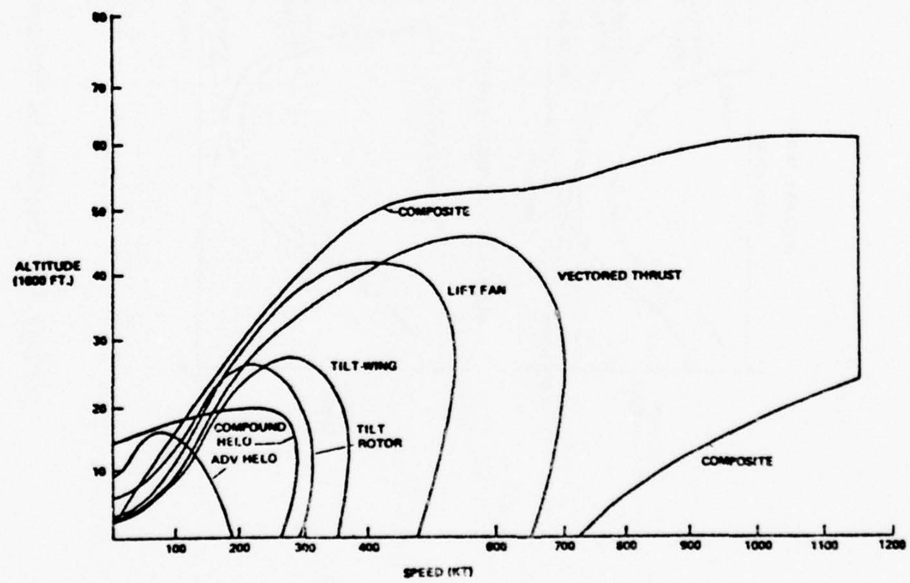

Figure 11. Speed - Altitude

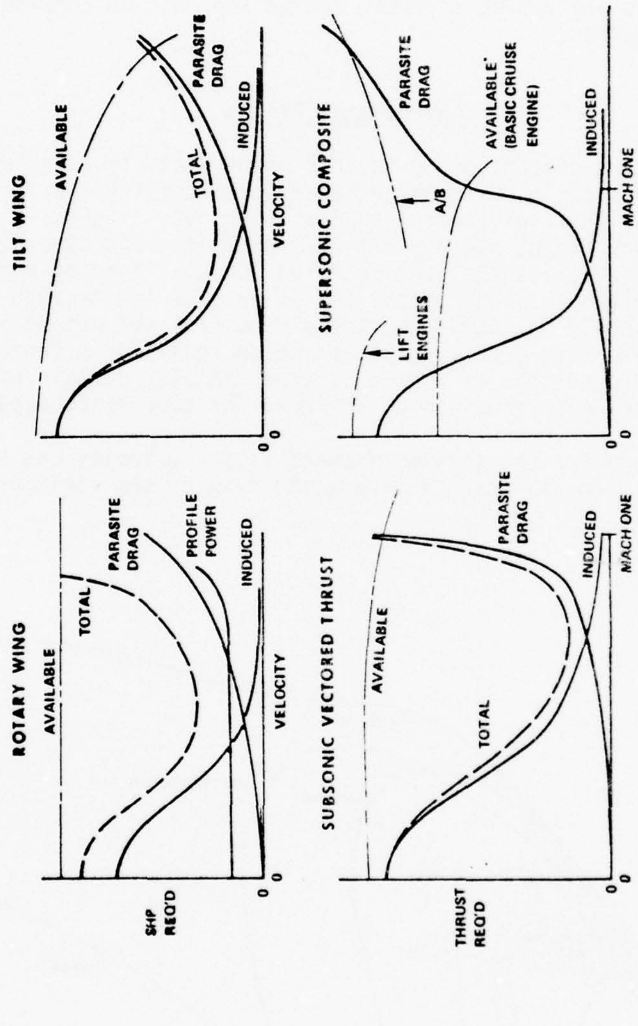

Figure 12. Factors in Forward Flight Performance

increases rapidly while the thrust available is decreasing. In the case of the helicopter, the abrupt increase in parasite drag is due to retreating blade stall. Since this phenomenon is well known, it will not be discussed further.

The large drag increase of the other configurations is due to the impact of designing for V/STOL on cruise performance. In the Breguet Equation for specific range the maximum lift to drag ratio (L/D max) is identified as the primary aerodynamic factor in cruise performance. (L/D) max is dependent on certain geometric and aerodynamic parameters as described in Figure 13. On the right hand side is shown a state-of-the-art comparison of (L/D) max versus the principal geometric parameter, span squared over wetted area (b^2/Swett). It can be seen that V/STOL configurations tend toward lower values of (b^2/Swett) and consequently experience lower values of L/D max than conventional designs. This trend is due to the fact that V/STOL aircraft will have greater wetted area because the total volume of these aircraft will be greater, since they must include additional components for propulsion and/or exhaust gas ducting, nozzle deflecting mechanisms, etc., and also since the aircraft designer will have less freedom in placing components due to packaging constraints such as meeting the balance requirements and limited inlet location placement to avoid recirculation/reingestion problems. Because of the

$$\text{Specific Range} = (L/D)_{Max} \frac{V_{cruise}(1/TSFC)}{\text{Weight}}$$

WHERE:

$(L/D)_{Max}$ = Maximum Lift to Drag Ratio

$$(L/D)_{Max} = f\left(\frac{b^2}{S_{wett}}, \frac{e}{C_f}\right)$$

b^2 = Square of Wing Span

S_{wett} = Total "Wetted" Surface Area of A/C

e = Efficiency Parameter for Wing

C_f = Measurement of Aerodynamic "Cleanness"

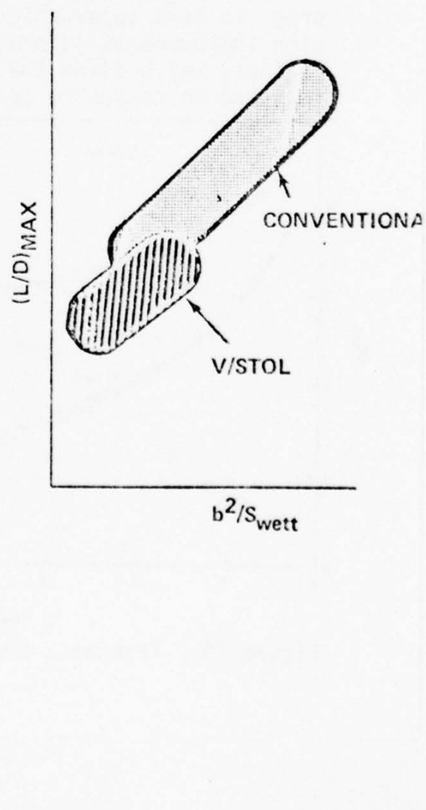

Figure 13. Impact of V/STOL Design on Cruise

propulsive lift available, V/STOL aircraft configurations do not require large amounts of aerodynamic lift for slow speed flight; the resultant designs tend toward lower wing spans in order to reduce structural weight. However, the attendant penalty of reduced (L/D)max must be accepted. Further, the mechanical complications involved in providing subaerodynamic control and either additional propulsion components or the capability to reorient the thrust axis will result in V/STOL aircraft that are less clean, i.e., have a higher parasite drag than conventional aircraft.

The design of supersonic V/STOL aircraft requires special consideration. In order for any aircraft to fly even reasonably efficiently at supersonic speeds, the transonic wave drag must be within acceptable limits. As shown in Figure 14, the wave drag of existing aircraft can be expressed as a first order function of the equivalent body fineness ratio. However, the theory also states that the cross sectional area distribution should be smooth, with no discontinuous second derivatives. Figure 15, presents the normalized cross-sectional area distribution of several contemporary aircraft, as well as a proposed lift plus lift/cruise configuration (CV-200A). As can be seen from the figure, configurations such as the AV-8A tend to experience area distribution indicative of high wave drag. This is borne out from inspection of Figure 14. At the present time, the area distributions and fineness ratios achievable with the lift plus lift/cruise configurations appear to offer low enough values of wave drag to make supersonic V/STOL aircraft achievable. However, continuing improvements in propulsion technology, such as Variable Cycle Engines, may provide the capability for supersonic flight without reliance on composite configurations.

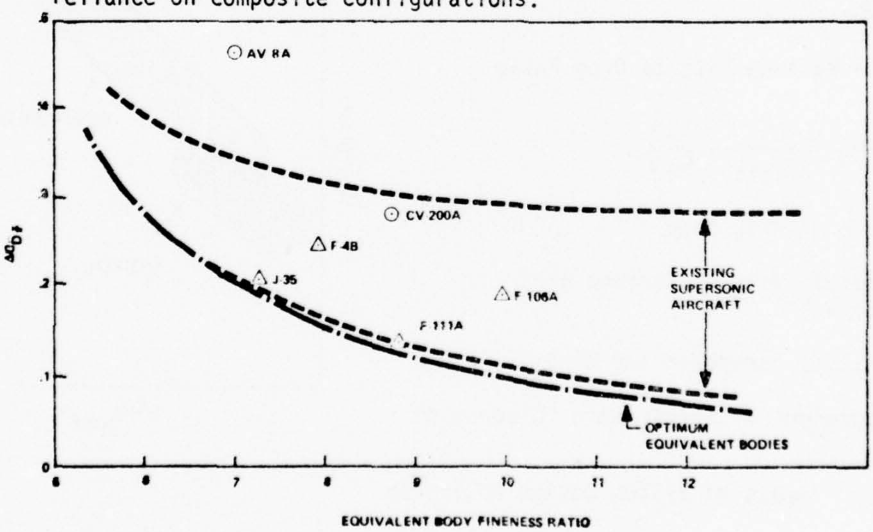

Figure 14. Transonic Drag Increment Comparison

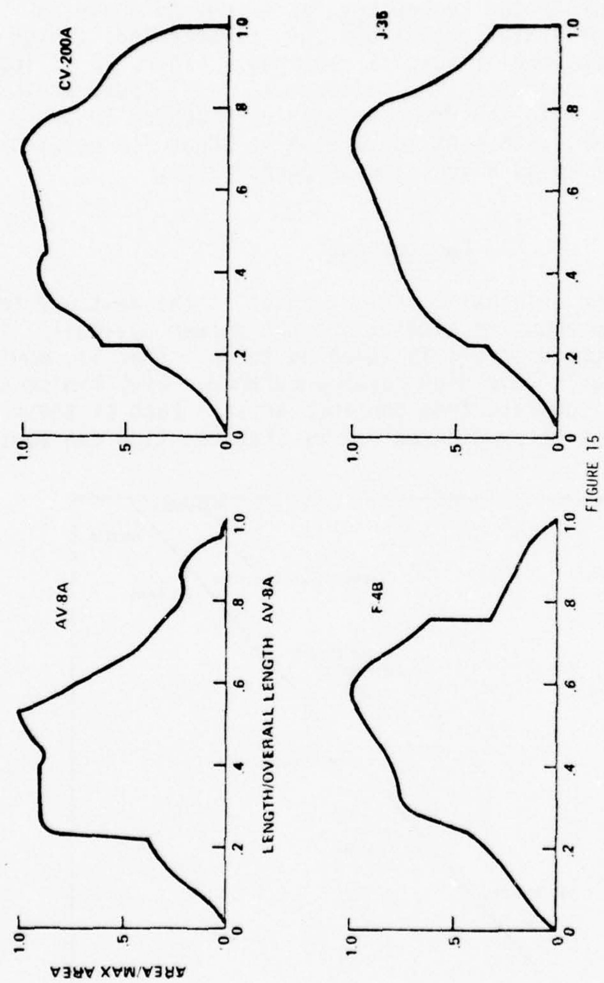

Figure 15. Comparison of Equivalent Bodies

Environmental Effects of V/STOL

The effects of V/STOL operations on surrounding personnel, equipment and facilities are noise, high temperature, and high velocity flow fields. In general the lower disc loading concepts have a more benign effect on the environment, but within the constraints of the demonstrated technology, as discussed above, a significant impact of V/STOL operations must be accepted if high performance in conventional flight is required. Figure 16 illustrates this relationship by comparing the maximum sea level speed achievable for various concepts with the efflux velocities produced in a VTO maneuver. Similar relationships could also be shown for noise or exhaust gas temperature as a function of performance.

Conclusions

We return to the original question, "What is the best way to go V?" In light of the above considerations, the answer obviously lies in what it is that the aircraft is intended to do. That is, what is the mission? Should it have high hover endurance, high cruise speed, fly supersonically, operate from confined areas? Each of these aspects and more must be considered before that question can begin to be answered.

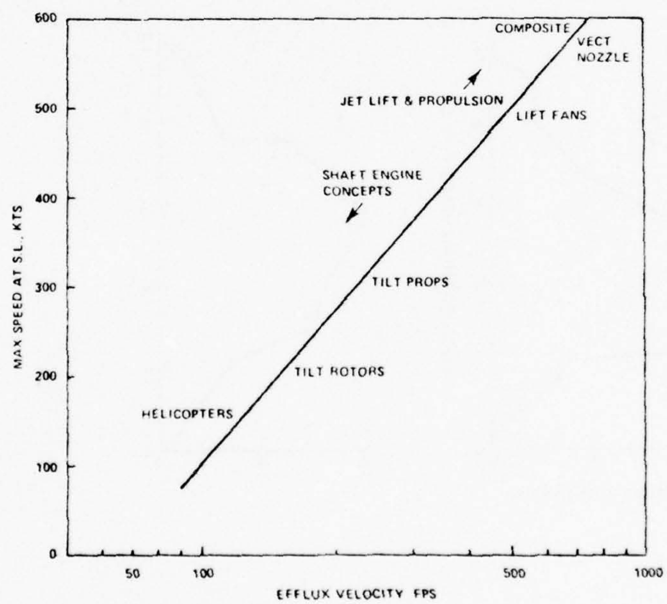

Figure 16. Maximum Speed Vs. Efflux Velocity

ACKNOWLEDGMENT

The author is deeply indebted to Mr. F. J. O'Brimski of the Naval Air Systems Command for his efforts in providing most of the data that forms the basis of this paper.

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DISCUSSION

HILL: (Grumman Aerospace Corporation)

I noticed that on a whole series of your curves that one configuration in the middle was left out. Was that for a specific reason or was it just that ...

SIEWERT:

No, I would not place any implications on the little figures that were put in or not put in. Basically it is the trend with increasing disc loading that is of significance and the little figures were only for illustration, we could not put them all in. You notice, augmentors are left out on quite a few of the pictures too.

HILL:

Yes. We personally feel that the high bypass fan tends to be the right kind of compromise, right in the middle of where many specific

paths cross. It is right in the middle of that curve in between the propeller and pure jet kind of thing. I was wondering if you had any specific reasons for having that left out on the whole series of curves.

SIEWERT:

No, we do not like to play favorites, that is for sure!

FROM THE AMST TO THE FUTURE

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Aeronautical Systems Division

Wright-Patterson Air Force Base, Ohio 45433

ABSTRACT

The unique requirements of an Air Force STOL aircraft require careful consideration of engine-airframe integration problems. Two airframe manufacturers are under contract to design and build prototype aircraft aimed at meeting the goals set by the Air Force for the modernization of tactical airlift. The paper will describe briefly the requirements for such STOL transport and the integration concepts related to performance, functional operability, and engine physical life. Thus, the principal features of YC-14 will include the super critical wing, upper surface blown flap, thrust-reversers and triply redundant digital flight control. The integration of the flight control system with engine throttle control will be discussed with particular emphasis on loss of engine. Similarly, various features of the YC-15 will be discussed including the super critical wing, externally blown flap, thrust-reverser operation, and exhaust nozzle mixer, as well as digital thrust management system. Some unexpected problems that arose during prototype testing will also be described.

In developing such aircraft, there is obvious need for initiative on the part of management to reduce cost in particular operations and support costs for both airframe and engines. The management approach to AMST is therefore of interest. A schedule of program activities as well as potential technological spin-offs as a result of developments in this program are presented to illustrate the management approach. Finally, a brief description is given of monetary constraints in undertaking future developments in this area.

INTRODUCTION

The C-130 has been a workhorse for the USAF tactical airlift inventory for over 20 years. The question may be asked, "Why replace it?" As shown in Figure 1, the C-130 fleet is aging, and many of the aircraft will be phased out of the inventory in the mid- to late-1980 time frame as a result of having used up the structural fatigue life; therefore, these aircraft will have to be replaced in this time frame.

Why not buy more C-130s? A major reason is that, over the past 10 to 15 years, the Army's main fire power has grown heavier and wider. In addition, the increased mobility provided by the mechanized infantry combat vehicle (MICV) and self-propelled artillery, such as the 155mm self-propelled Howitzer and 8" self-propelled Howitzer, are all too large for the C-130. This is also true of the M-60 main battle tank and the XM-1 replacement for the M-60. Thus, the AMST will provide a tactical airlift capability of combat vehicles never before provided to ground forces. In addition, the AMST will provide a Short Takeoff and Landing (STOL) capability which will allow delivery of twice the payload in less than half the field length (see Figure 2). As shown on Figure 3, there are significantly more air fields in West Germany available with a length of 2,000 feet or longer, which is the minimum design capability for AMST, than there are at 3,500 feet, the

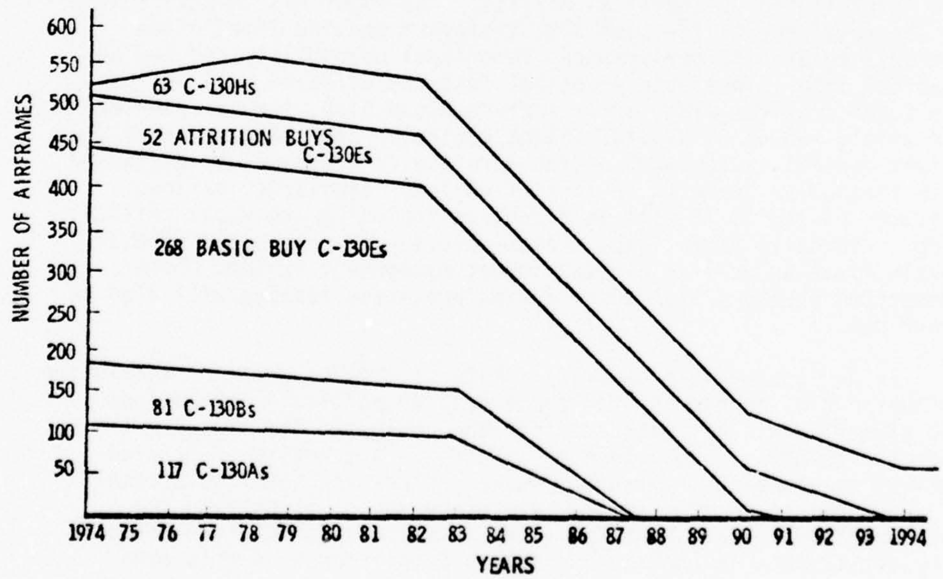

Figure 1. C-130 Airframe Fatigue Life Projection

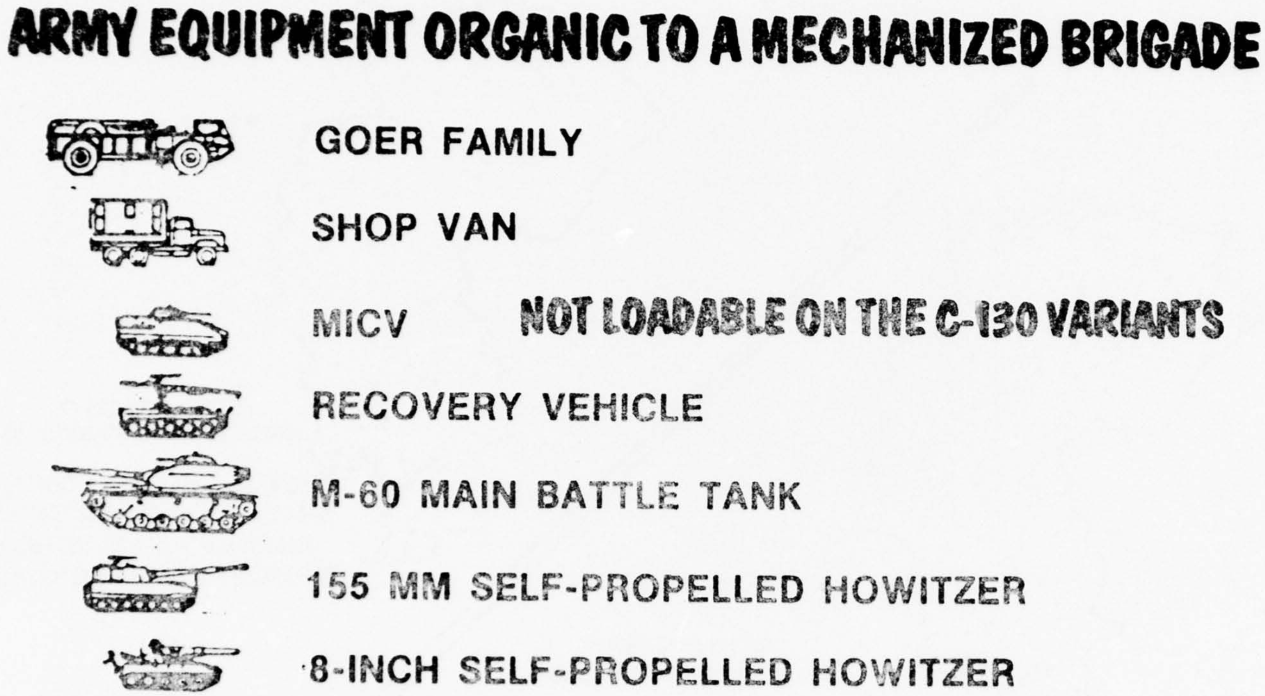

Figure 2. Outsize Payloads and STOL Field Lengths

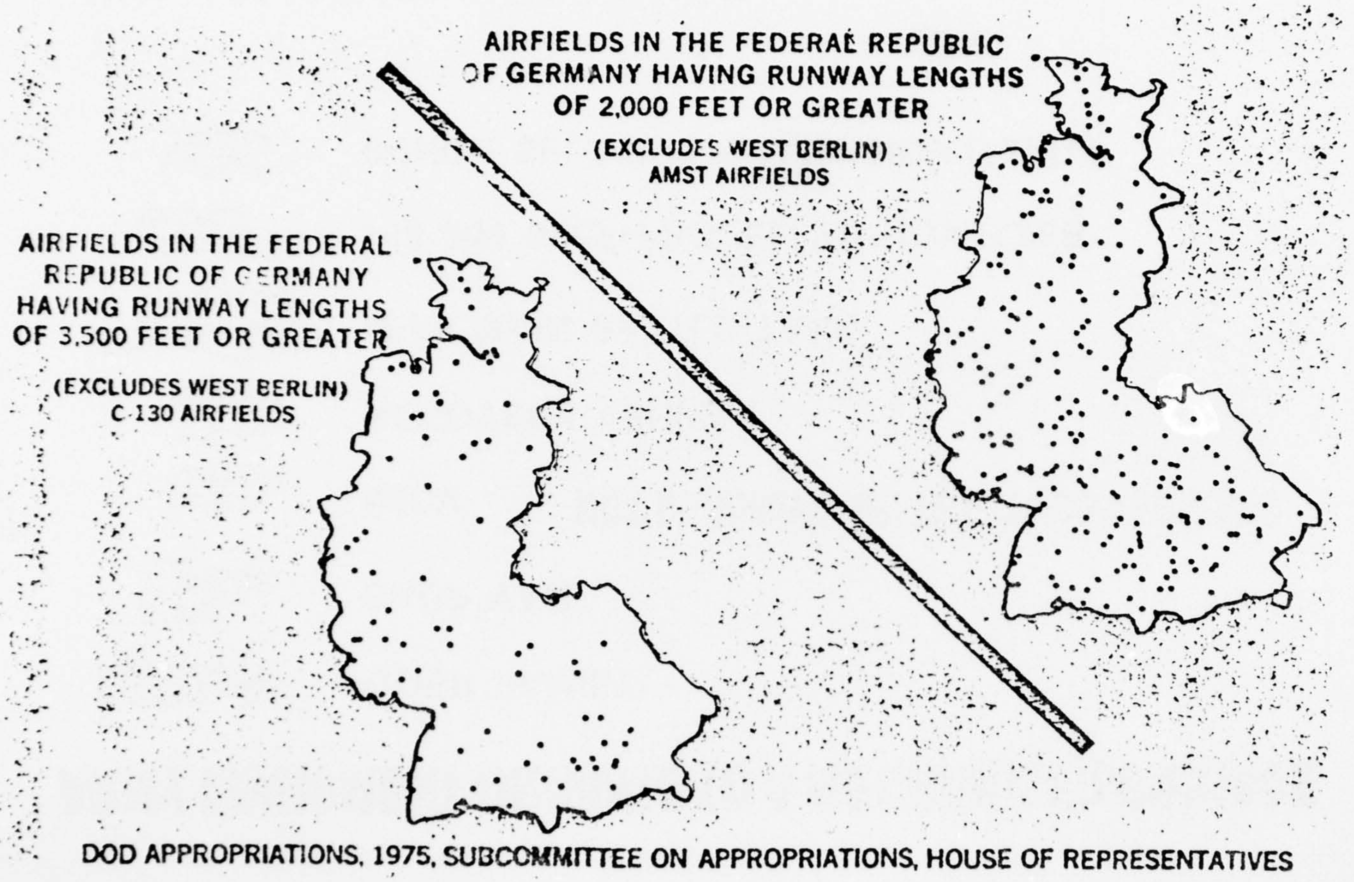

Figure 3. Need for STOL

minimum design capability for the C-130. Thus, the AMST will provide a replacement for the aging tactical airlift force capable of transporting the key fire power elements of the ground forces with a tactical flexibility far in excess of the current tactical airlift inventory.

AMST Prototype Features

The YC-14 is a two-engine design powered by CF6-50D GE engines at a rated sea level thrust of 50,000 pounds. Dimensions of this aircraft are shown in Figure 4. The McDonnell Douglas YC-15 is powered by four Pratt & Whitney JTID-17 engines at a rated sea level thrust of 16,000 pounds each. Dimensions of this aircraft are shown in Figure 5. In conjunction with a CFM-56 engine, McDonnell Douglas also incorporated a 2,100 square foot wing on one of the YC-15 prototypes. The YC-15 is currently flying three different engines: JT8D-17, the JT8D-209, and the CFM-56. The CFM-56 and the JT8D-209 are installed in the number one engine position on each prototype. Characteristics of these engines are shown in Figure 6 along with the CF6-50D engine used on the YC-14.

Powered Lift Concepts

To achieve the STOL capability desired in the AMST without prohibitive weight increases, powered lift was utilized. The Boeing YC-14 incorporates an upper surface blown flap, and the Douglas YC-15 incorporates an externally blown flap, as shown in Figure 7.

AMST Performance

The design goals for the AMST prototype were: a ferry range of 2600 miles without payload; the capability to take off with 27,000 pounds of payload, fly 400 nautical miles, and land at an austere 2,000 foot strip; off-load 27,000 pounds and on-load 27,000 pounds; take off without refueling and return to the main operating base (see Figure 8). The box size for the aircraft is 11.3 x 11.7 x 47 feet. Conventional takeoff and landing capability was established as a 53,000 pound payload. The runway surface for landing at midpoint on the 400 mile radius mission was a California Bearing Ratio (CBR) 6 surface, which is about the equivalent of a soft golf course fairway. In addition, the aircraft must land on this runway over a 50-foot obstacle at the end of a 2,000 foot runway under sea level on a 103°F day with idle reverse thrust (see Figure 9). The aircraft must be able to take off at this 2,000 foot field, lose an engine, and be able to either successfully continue takeoff or stop on the remaining runway. This is the definition of critical field length. The aircraft must also be able to land in 2,000 feet with an engine

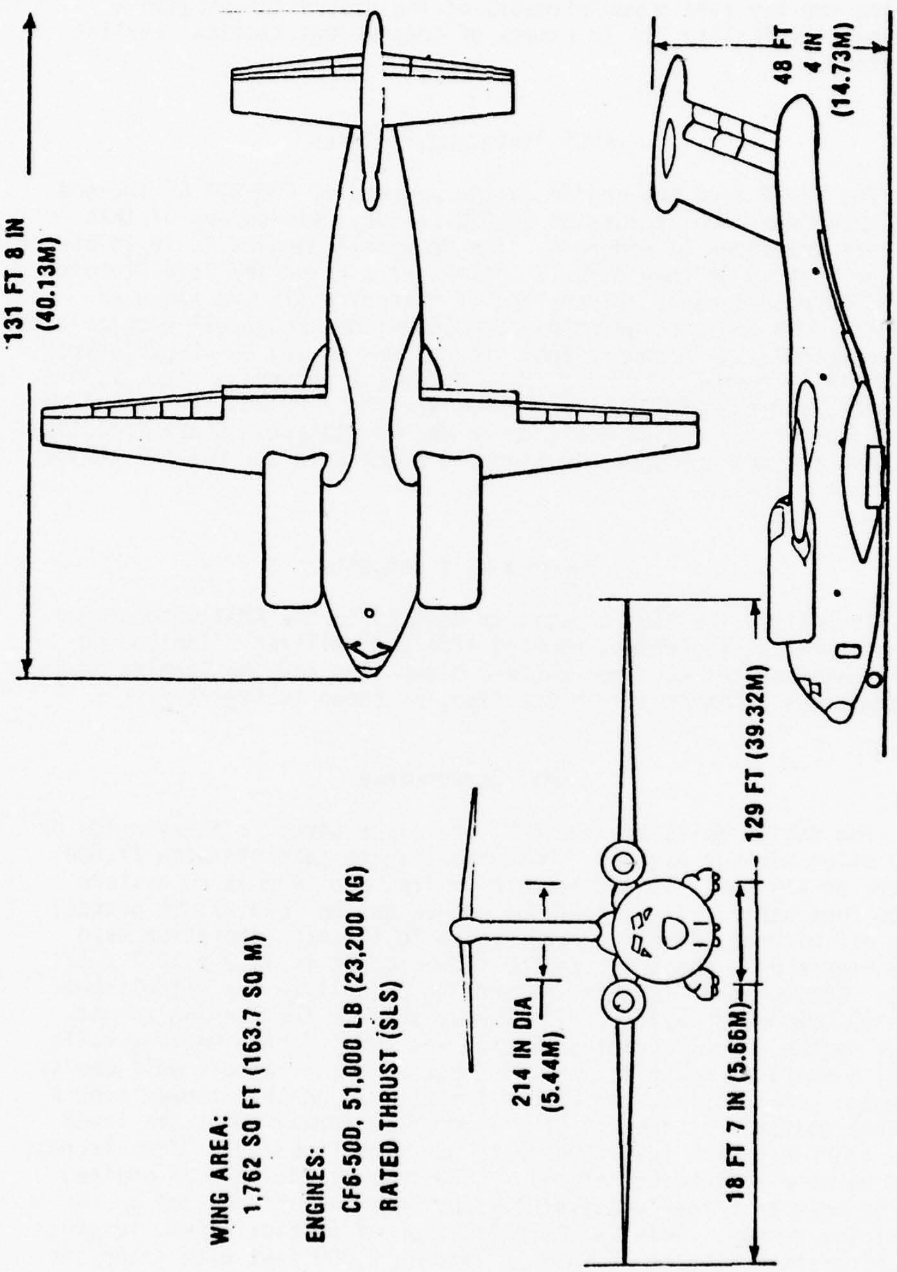

Figure 4. YC-14 General Arrangement

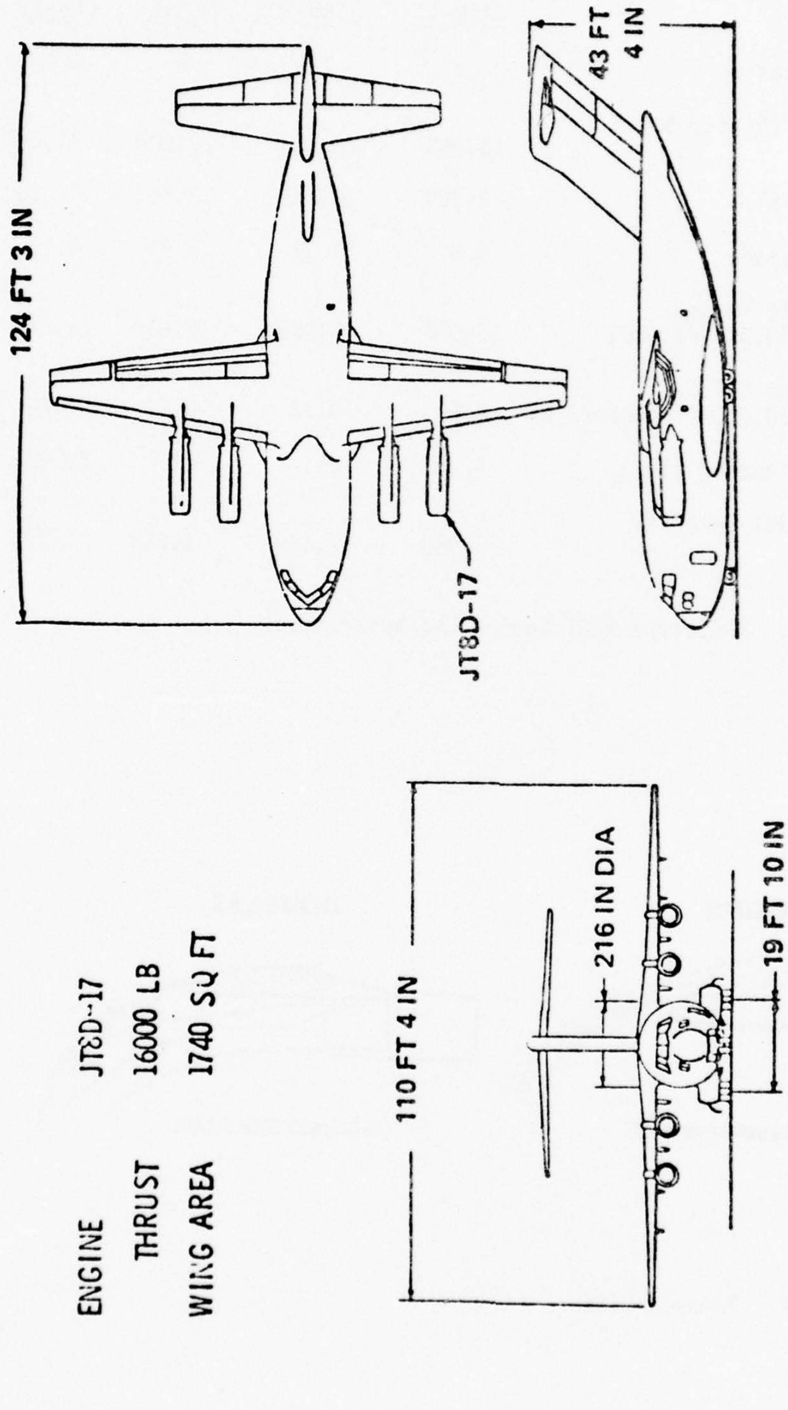

Figure 5. McDonnel Douglas AMST Prototype YC-15

	JT8D-17	JT8D-209	CFM56	CF6-50
By-Pass Ratio	1	1.68	6	4.4
Take-Off Thrust, S.L.S., (lb)	16,000	18,000	22,000	51,000
Weight (lb)	3,300	4,135	3,700	8,325
Thrust Weight	4.8	4.35	5.95	6.1
Max Cruise Thrust M 0.75, 30,000 Ft (lb)	5,040	5,350	5,810	13,375
Max Cruise S.F.C. M 0.75, 30,000 Ft (lb/hr/lb)	0.82	0.74	0.64	.64
Engine Diameter (in.)	42.5	54	72	86.4
Primary Jet Velocity (ft/sec)	1,950	1,445	1,255	1,500

Figure 6. Prototype AMST Engine Characteristics

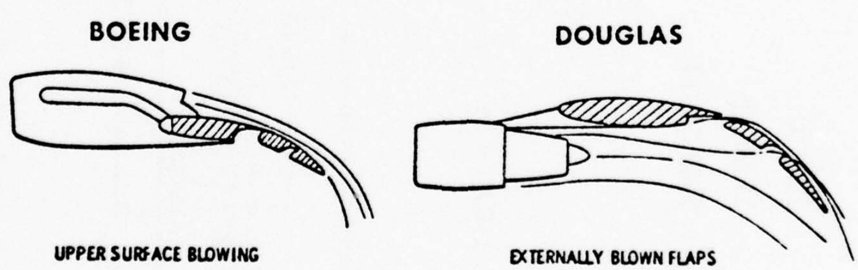

Figure 7. Powered Lift

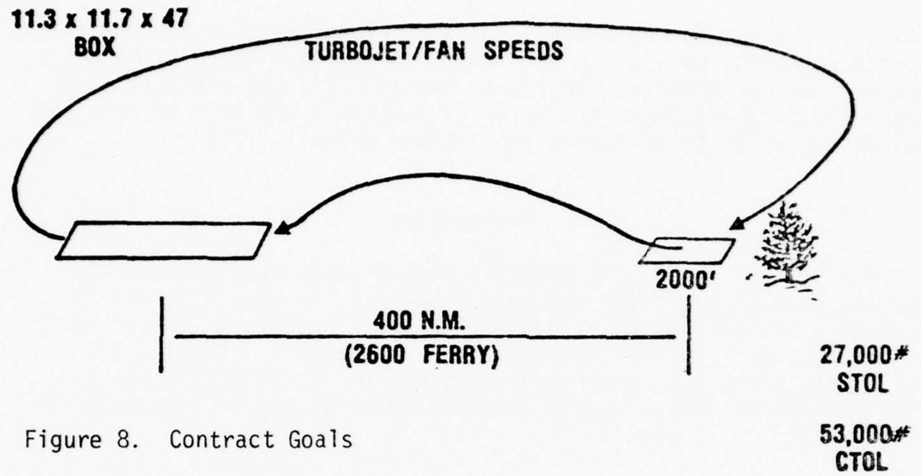

Figure 8. Contract Goals

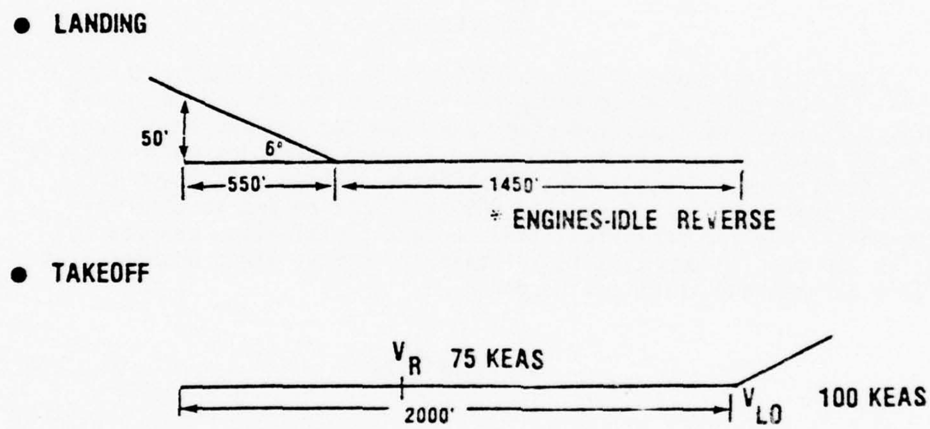

Figure 9. AMST Landing/Takeoff Distances, 27,000 Lb. Payload, 400 Nautical Mile Radius, Sea Level/103°F Day

out. This provides the capability of landing on an unprepared field with 27,000 pounds of payload under assault rules in less than 1,000 feet on a standard day.

ENGINE/AIRFRAME INTEGRATION ASPECTS

The integration of an engine into an aircraft system is very complex and requires the consideration of many technical aspects to

arrive at a balanced engineering design. Among the more important aspects requiring careful consideration during this integration process are the performance, functional operability, and physical life areas. These three areas are mutually dependent and must be balanced against tradeoffs to arrive at the optimum overall system.

Performance

Performance covers the thrust and fuel usage requirements to meet the conditions of the aircraft mission. The engine thrust required for the AMST aircraft was dictated by the mission requirements for a 2,000 foot critical field length for takeoff. The aircraft cruise conditions are of particular importance to determine the aircraft fuel tank sizing to meet the mission range and payload requirements.

Operability

Operability concerns the ability of the engine to provide the required performance while installed in the aircraft and under all mission flight envelopes and under any expected ambient conditions. Special emphasis was placed on the nozzle and inlet design to satisfy the requirement for stall-free transient operation. Although the engines being considered for the AMST application are to be "off-the-shelf" and FAA certified, their unique installation aspects require the special emphasis to maintain or improve their demonstrated stall margins and transient response.

Physical Life

This includes the mechanical aspects of the installation of the engine and the manner in which the engine is used and maintained in operation for the optimum parts usage life. Power management, or the judicious use of engine thrust to the level required, can provide a significant increase in engine life. With the high thrust to weight ratios of the AMST, power management can play an important role in life cycle costs. Initial studies have indicated that reduced power takeoffs and other power management techniques, coupled with the high thrust to weight capability, will provide engine usage less severe than encountered on commercial engines today.

C-14 ENGINE/AIRFRAME INTEGRATION ASPECTS

Supercritical Wing

The YC-14 wing incorporates an advanced technology airfoil designed specifically for the AMST application. By taking advantage of this advanced airfoil technology, the YC-14 is able to achieve cruise Mach numbers of 0.70 and higher with a relatively thick, straight wing. Because of the interaction of the jet exhaust and wing flow fields, due to the unique engine installation, the wing design had to be done parallel with the nozzle development.

Upper Surface Blowing (USB)

For an airplane using powered lift, the engine is an integral part of the system required to generate high lift for STOL operation. In the YC-14 USB system, the flow from the nozzle has been tailored in conjunction with the USB flap behind it so that the engine exhaust flow is turned efficiently and remains attached to the flap contour by the Coanda effect to generate powered lift (see Figure 10). The major task of the aerodynamicist has been to integrate the engine installation to be able to utilize the exceptional powered lift potential of the USB concept in the low speed regime while maintaining high cruise efficiency in the high speed regime.

Confluent-Flow Exhaust

A confluent-flow exhaust nozzle is used--that is, the cool fan exhaust is ducted aft where it joins with the hot core engine exhaust. then further aft to a final nozzle located on the upper surface of the wing. No forced mixing occurs between the core nozzle plane and the final nozzle exit (see Figure 11). Use of a high bypass turbofan engine along with the core nozzle canted upwards tends to significantly reduce the surface temperatures on the USB flap. This nozzle system is, obviously, aerodynamically different from the separated flow system and must be sized to achieve a proper rotational speed match between the two engine rotors.

Thrust Reverser

The exhaust system incorporates a thrust reverser that discharges both the primary and fan exhaust gas through a single opening located at the top of the nacelle. The engine exhaust gas is directed upward and forward, putting a downward force on the landing gear. The system can be used to back up the airplane with part power and

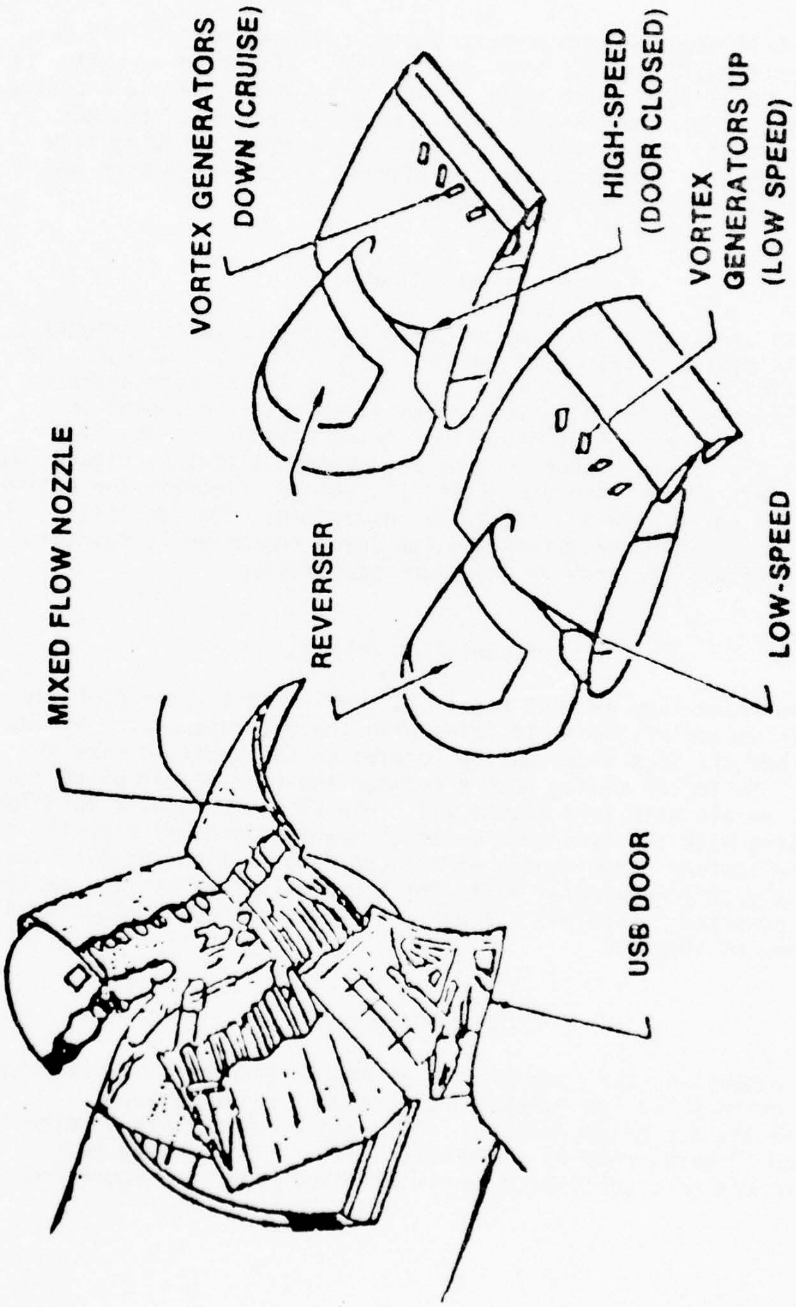

Figure 10. YC-14 Engine Nozzle and Vortex Generators

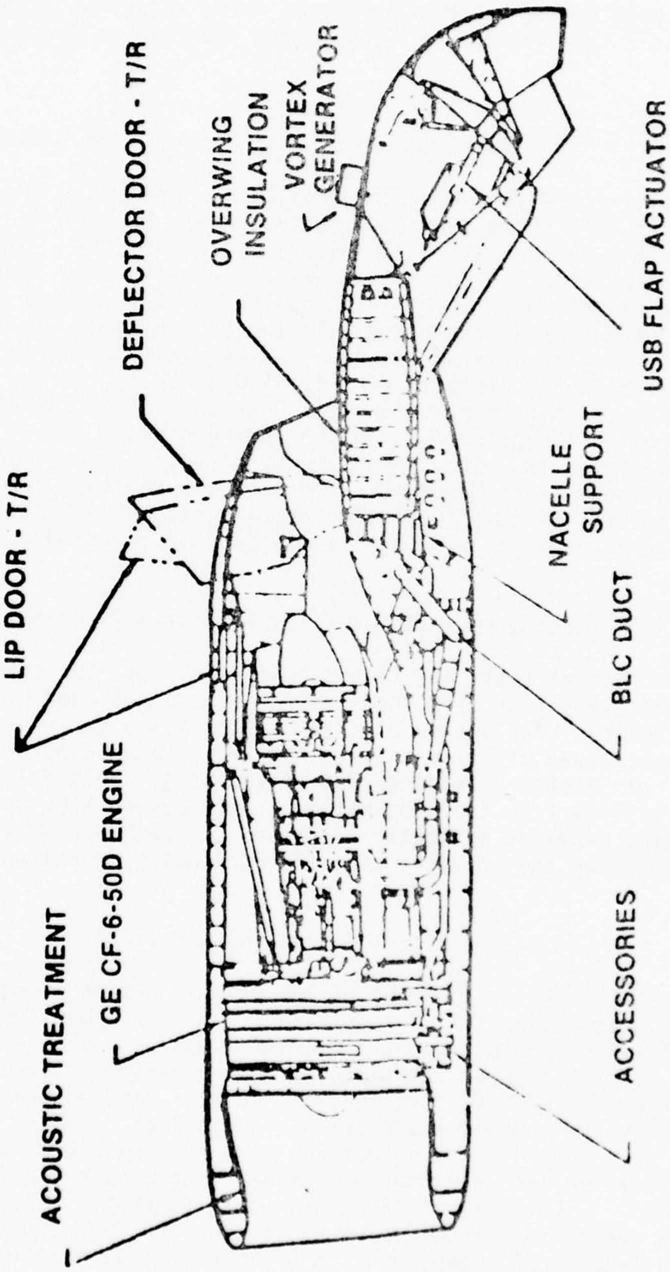

Figure 11. YC-14 Engine and Nacelle Cross Section

to direct the exhaust gases of an idling engine away from the cargo loading area during loading operation. The actuation is performed by a dual concentric hydraulic actuator attached to the deflector door.

Low Speed Nozzle Door and Vortex Generators

One of the keys to obtaining optimum performance from the nozzle in both the low speed powered lift and cruise regimes is the inclusion of a door on the outboard side of the nozzle. The door is opened to provide maximum exit area for takeoff thrust and to spread the exhaust jet over the USB flap, which results in more efficient Coanda turning characteristics. In the cruise mode, the door is closed to provide optimum exit area for minimum cruise specific fuel consumption and to minimize exhaust flow scrubbing of the wing surface. At very high turning angles, Coanda turning is enhanced by the use of vortex generators in the engine exhaust stream which essentially energize the boundary layer to delay jet flow separation on the USB flap. These generators are also two-position actuated: raised during low speed operation and lowered to be flat on the exterior surface of the wing in the cruise configuration.

Leading Edge Boundary Layer Control (BLC) System

Wind tunnel tests and flight demonstrations have successfully shown that very high lift coefficients can be achieved by boundary layer control. The system utilized on the YC-14 takes high pressure compressor bleed air from the engines and, via an ejector system, "blows" air over the leading edge of the wing. This delays separation of the airflow from the wing at low speeds during STOL operations. The system provides automatic operation to keep crew workload to a minimum during the critical takeoff and landing operations.

Flight Control System (FCS)

The YC-14 Flight Control System employs 3-axis mechanical powered controls with electrical augmentation. Through the use of a primary mechanical FCS, pilot inputs are transmitted by cables and pushrods to the hydraulic power actuators on the elevators, ailerons/spoilers, and rudders to provide longitudinal, lateral, and directional control respectively. Command augmentation is provided in pitch and roll with triple redundant digital computers. The triplex electrical flight control system also provides engine out, yaw damping, and autopilot functions. Fiber optic interchannel data links are employed in the triplex system to transmit synchronization and redundancy management signals between the computers, thus assuring that electrical isolation

is maintained between the redundant channels. The high lift system, which is composed of both leading and trailing edge flaps, is controlled by a combination of mechanical and electrical means. The leading edge Krueger flaps and the double slotted outer and middle trailing edge flap control inputs are transmitted through cables and torque tubes to hydraulic actuators. The trailing edge upper surface blown (USB) flaps are commanded through the triplex electrical flight control system only.

Propulsion/Flight Control Integration

Propulsion/FCS integration is used on the YC-14 to provide for automatic speed hold control and automatic reconfiguration after an engine failure occurs during STOL operation. The electrical FCS provides speed hold control through coordinated adjustments of the USB flaps and the autothrottle, which results in short term and long term variations in flight path velocity. In order to control flight path during STOL approach, the YC-14 pilot adjusts only the aircraft attitude with control column inputs. The FCS automatically repositions control and high lift surfaces after an engine failure during STOL operation to improve reaction time for landing or go-around. After an engine failure, the FCS automatically positions the engine out USB flap to match the outboard flaps (see Figure 12). The engine out USB slots are opened to increase lift effectiveness of that flap. Outboard flaps on the live engine side are partially retracted to reduce adverse rolling moment. The FCS augmentation provides aileron/spoiler and rudder inputs to stabilize the aircraft in roll and yaw. The live engine USB and autothrottle continue to modulate for speed hold control.

YC-15 ENGINE/AIRFRAME INTEGRATION ASPECTS

Supercritical Wing

The YC-15 has a supercritical wing design with the same advantages as mentioned on the YC-14.

Externally Blown Flap (EBF)

The key to the short field performance of the YC-15 is the externally blown flap system (see Figure 13). The pylon is arranged to position the exhaust nozzle ahead of and close to the under surface of the wing such that a large proportion of the jet reacts on each of the two slotted flap segments and makes for maximized thrust deflection and wing circulation. The exhaust gases impinge on the flaps and are deflected downward at approximately the same angle as the flap deflection. The jet leaving the flap trailing edge acts

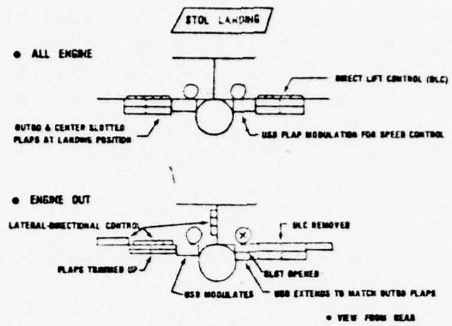

Figure 12. YC-14 Engine Out Automatic Reconfiguration

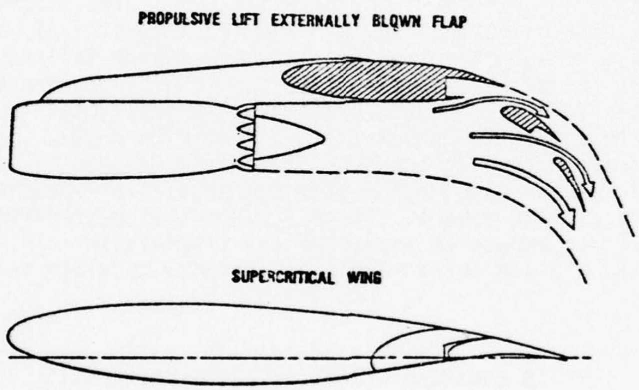

Figure 13. YC-15 Propulsive Lift Externally Blown Flap

as a physical extension of the flap as well as providing direct reaction lift.

Thick Inlet Lower Lip

The increased wing circulation due to a blown flap results in high angles of attack at the engine inlet. A thick inlet lower lip is featured to prevent inlet flow separation at high inlet air attack angles during high lift modes of flight (see Figure 14).

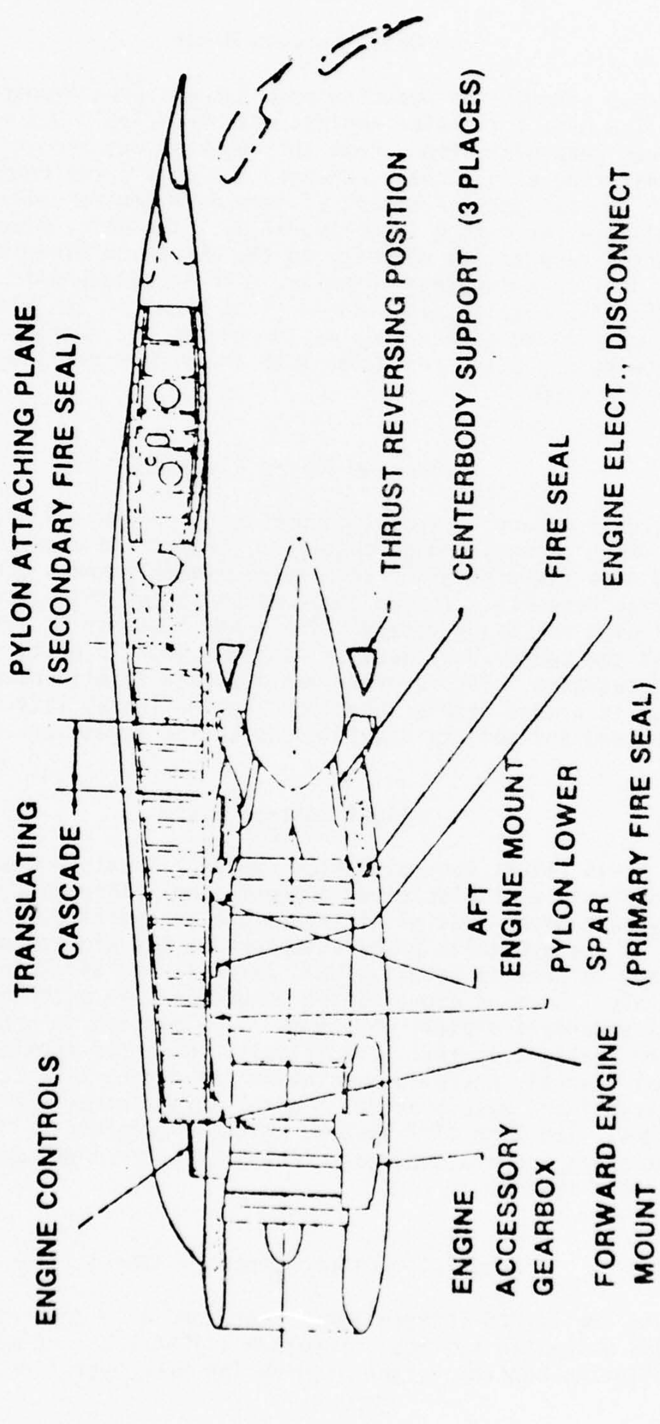

Figure 14. YC-15 Engine and Nacelle

Daisy Exhaust Mixer

Numerous schemes for reducing peak temperatures imposed on the EBF using low bypass turbofan engines with relatively hot exhaust temperatures were evaluated. From this work it was recognized that a mixer featuring daisy lobes arranged around a large center body would tend to maximize the amount of secondary ambient air between lobes available for mixing (see Figure 15). The daisy mixer also significantly reduces jet velocity on the EBF, which minimizes noise generated by flap scrubbing. Finally, with an aerodynamic constraint that the nozzle exit plane must be ahead of the wing leading edge, use of a large center body would provide the shortest nozzle, hence allowing for a shorter pylon with the engine positioned closer to the wing.

Cascade Thrust Reverser

The YC-15 exhaust system incorporates a ground and inflight reverser which directs the exhaust flow forward and upward and is comprised of a fixed reverser structural assembly and a translating cascade ring assembly. In the deployed position, the cascades are moved aft with the daisy nozzle. The inner flowpath is effectively blocked at the center body maximum diameter, and air exits through the uncovered cascades. The reverser was designed to eliminate engine damage due to ground debris ingestion, engine instability due to self-ingestion, and instability due to cross-engine ingestion.

Flight Control System

The YC-15 Flight Control System employs 3-axis mechanical powered controls with electrical augmentation. Through the use of a primary mechanical FCS, pilot inputs are transmitted by cables and pushrods to the hydraulic power actuators on the elevators, ailerons, and rudders to provide longitudinal, lateral, and directional control respectively. Command augmentation is provided in pitch and roll with dual redundant digital computers. Yaw damping is provided with a dualized analog computer. Electrically commanded spoilers provide additional lateral control augmentation and direct lift control. The electrical FCS also provides digital thrust management and autopilot modes. The high lift system, which is composed of leading edge flaps and slats and trailing edge flaps, is controlled mechanically through cable systems.

Propulsion/Flight Control Integration

YC-15 engine/FCS integration is composed of a combination of manual and automatic systems. Throttle adjustments are made manually by the pilot to control flight path during STOL operations. The

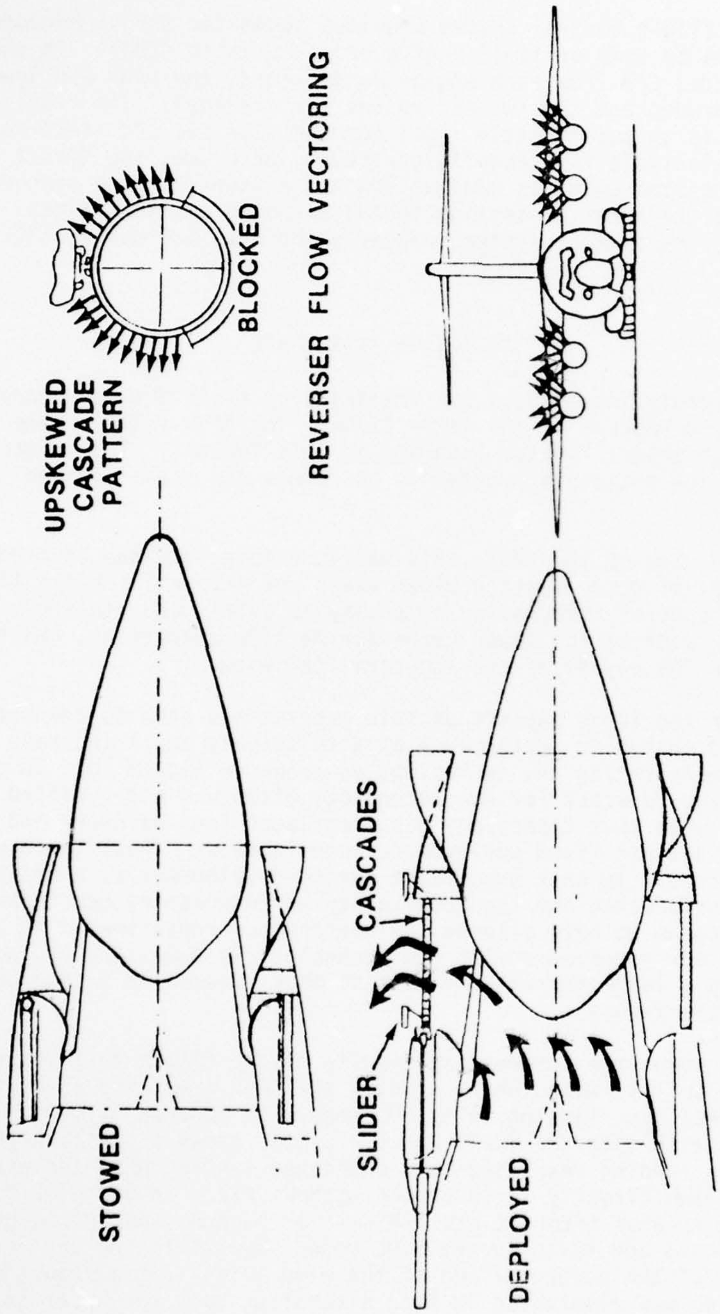

Figure 15. Thrust Reverser Daisy Exhaust Assembly

electrical flight control system provides automatic thrust management capabilities to trim or limit engine pressure ratio (EPR). In the EPR trim mode, EPR from each engine is detected, the high and low value discarded, and the two mid-values are averaged. The resultant EPR signal is an autothrottle input command. In the EPR limit mode, which is selectable for takeoff, go-around, or climb, the thrust management system computes maximum EPR and prevents engine overboost by limiting the throttle setting to the computed maximum values. The YC-15 thrust management system reduces pilot workload during STOL operation.

PROTOTYPE EXPERIENCE

The prototyping concept was initiated by Assistant Secretary of Defense David Packard in the early 1970s. The AMST followed the AX and the Lightweight Fighter Programs in this concept. The primary purpose of the prototype program is to reduce the areas of high technical risk.

In the case of the AMST, this was to demonstrate the powered lift concept of upper surface blown flaps and externally blown flaps, the flight control integration necessary to safely and routinely fly on the back side of the power curve during STOL operations, and to demonstrate the payoff of the supercritical wing.

One of the large payoffs of this program has been to demonstrate a projected change in performance at a relatively small increase in cost by demonstrating new technology, as shown in Figure 16. In past programs, the forecast for new technology often was not verified by experience when that technology was translated into hardware and demonstrated against fixed performance requirements. Thus, there was a large increase in cost experience during development as a result of fixed performance and the uncertainty of forecasting new technology. In this program we have allowed the performance requirements to vary so that as our experience with new technology is demonstrated, we do not pick up a large increase in cost if this advance in technology is less than the forecast.

There were some interesting results of the flight test program. First, the initial wind tunnel studies for NASA predicted a negative ground effect upon landing, in a STOL mode with powered lift. Flight experience with both the YC-14 and YC-15 has shown a positive ground effect upon landing resulting in a cushioned landing when the aircraft is flown directly to touchdown without flare in the STOL mode. The second area of interest was the skip or bounce experienced by both the YC-14 and YC-15 in the STOL mode. By initiating spoiler deployment of the upper surface of the wing after initial touchdown, this problem was eliminated in both aircraft. Both the upper surface blown flap on the YC-14 and the externally blown flap on the YC-15

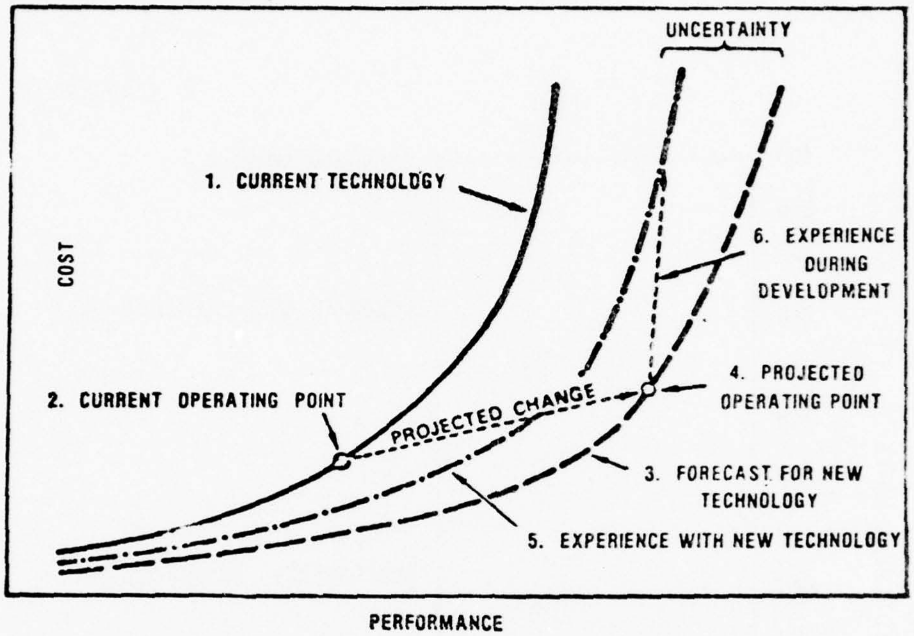

Figure 16. Impact of Uncertainty on Systems Acquisition Costs and Performance

have proven to be very successful means of generating high lift coefficients with powered lift. Lift coefficients in the order of four to five are common in STOL operations. In addition, the successful demonstration of engine/airframe integration, flight control systems capable of safe and routine operation during STOL operations, along with the supercritical wing have proven that potentionally high risk areas forecasted for the prototype program have been demonstrated to be of low technical risk.

MANAGEMENT APPROACH

The recent history of spending by GNP sectors in constant year dollars is shown in Figure 17. The reduction in defense spending through FY76 is quite striking in comparison to the other GNP sectors. This reduction, as well as defense requirements for other high priority programs, has resulted in many of the innovative and austere approaches taken in development of the AMST. Under these circumstances, the development dollars required for a traditional type of development program are not available. Therefore, an innovative program has to be developed to reduce both the acquisition and support costs comprising the total life cycle cost on the aircraft. To reduce

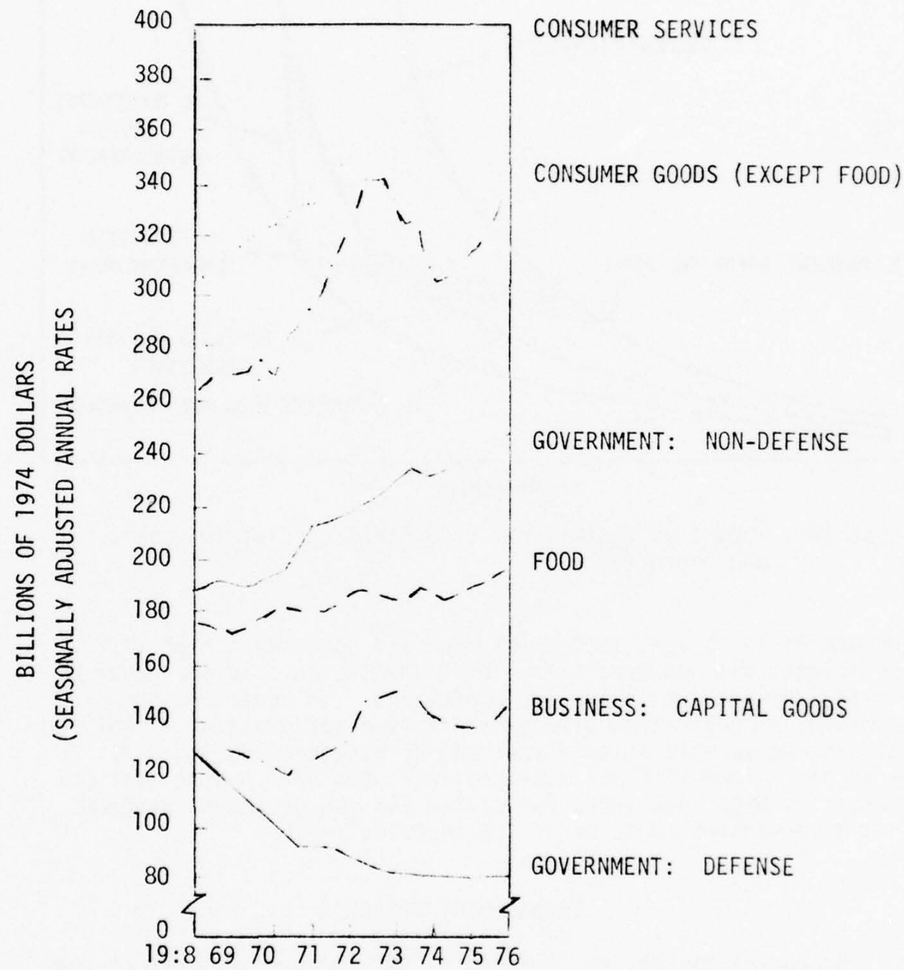

Figure 17. The Course of Spending by G.N.P. Sectors

development dollars, we have a goal of the use of 70% common (existing) equipments on the aircraft. These common equipments are proven reliable Government furnished or commercially available equipments. In addition, it is planned to use existing facilities to reduce both development and acquisition costs and a major effort has been made to reduce the part count on the aircraft by the simplicity of design and maximum use of interchangeability The commercial approach to maintenance manuals has been employed to reduce the skill level required on the flight line as well as to expedite fault isolation. The state-of-the-art improvements in corrosion prevention have been utilized, capitalizing on the AF lessons learned on transport aircraft. The AMST will be designed for an extended service life of at least 25,000 hours on this aircraft. Military specifications were tailored for the AMST and reduced to the minimum required for this procurement. An additional major payoff to the prototype program is to examine supportability of the aircraft and allow feedback of this information to the contractors for the follow-on design.

TECHNOLOGICAL SPINOFFS

Some of the technological spinoffs of this program are as follows: the proven STOL concepts of the AMST will provide a significantly greater capability as a replacement aircraft for many hundreds of C-130s sold to foreign governments. The engine/airframe integrated powered lift flight control systems in both of these aircraft have proven to be highly successful. The concept of the integrated control system and digital systems application, as well as the power management techniques developed in this program, may well have direct application to future commercial aircraft design.

SUMMARY

There is a well-reasoned requirement for the tactical airlift improvements offered by the AMST. The engine/airframe integration during STOL operations has proven to be very successful in this program. Dollar constraints in this program have resulted in a unique management approach for engineering development. There has been significant technology developed in this program that will have applications in other programs in the future.

DISCUSSION

WEINRAUB: (Naval Air Stystems Command)

It interested me, the fact that you put the propulsion airframe integration off on the contractor. By doing this,did you lose some communication and some control over this, and how do you feel about this area if you did lose this?

KISHLINE:

Well, that is something that does concern us. The primary reason for laying it off on the prime contractor is that I could not find anybody in the Air Force smart enough to stand between them and do the technical integration arbitration, to be quite honest. We are trying to ensure that we keep a commercial configuration on the engine rather than deviating off to a military configuration so that we can take advantage of the rate at which a commercial engine matures, and we can take advantage of the lessons learned on a commercial engine. Our usage rate is not near what it is on the commercial inventory, so we are trying hard to keep the commercial configuration and not allow the prime manufacturer to drive the engine to solve his problems. Now that is a concern. One of the ideas I am toying with is to freeze the configuration. In other words, the engine has to come off the commercial line and, maybe, give configuration control of the engine to the engine manufacturer.

EMERSON: (Pratt and Whitney Aircraft)

Remembering that the automatic configuration control is part of the aircraft system and that these are commercial engines with relatively simple hydromechanical controls, how does the flight control detect a failed or failing engine? Does that also have automatic modulation of the flight control surfaces to compensate for a failed engine?

KISHLINE:

There is a logic network in the computer. When you get a given set of conditions or family of conditions reached, the computer software is programmed to assume that that is an engine out. So, if there is a thrust differential or a given set of conditions between both engines, it will then reconfigure the aircraft. I do not recall exactly what the various conditions are, but thrust is one of them.

CRAGIN: (General Dynamics)

On the over-the-wing blowing system where you deployed those spoilers for, I guess, helping the flow stay attached to the flap, what sort of increase in efficiency do those spoilers really have as far as increasing the time the flow can stay on the flap and the actual flap deflection angle? In other words, what would be the limit of the flap deflection with and without the spoilers?

KISHLINE:

I do not believe I can answer that question. I have flown the F-15 and you can use the spoilers in any flap setting and in the STOL mode. When you deploy the spoilers, it is just like going down in an

elevator. It really increases your rate of descent.

CRAGIN:

I am talking about the little vortex generators on the flap. How much do they help the flow to stay on the flap? In other words, what sort of flap deflection could you get with or without the vortex generators?

KISHLINE:

Essentially it increases the energy in the boundary layer which allows it to remain attached, increasing the turning angle 10° - 15°. They are rotating the air over 75° rotation, and it is still remaining attached.

CRAGIN:

Does it stay attached all the way through the take-off and landing sequence?

KISHLINE:

You normally do not take-off with the full USB flaps down, because you have sufficient power to do without that, but the USB flaps are deployed for landing. Yes, it does stay attached through landing.

BERNSTEIN: (Canadair Limited)

Are you aware of any consideration to the application of the AMST technology to commercial short-haul operation? Would it make a sensible proposition?

KISHLINE:

I think some studies have been done along that line. In terms of United States commercial operation, we have quite a few airfields around this country, and you are going to have a higher thrust to weight in the AMST than is really necessary for these runway lengths. The major air terminals will have to become saturated, probably out in the 1990 time frame, before you will see a commercial application in this country. However, looking at commercial application in foreign military sales does look quite attractive. We have already been approached by several countries that are quite interested in using this capability. For countries that do not have a modern transportation network, it may be more expensive to have to put in a rail or road network, particularly in jungle or mountain terrain, than it would be to put in a 2000 ft unprepared strip and operate out of those on strategic locations. The interest is further heightened by the fact that once you have that capability, you can now deploy your security

forces in times of national unrest at the same key locations, and it is easier to secure short field boundaries and strategic locations with fewer ground forces than it would be to secure an entire road or rail network. So there is a great deal of interest by developing nations in using a dual purpose aircraft, hauling large equipment, passengers, that sort of thing, more so than in this country. I did get an interesting call a week ago from the people up in Alaska. They wanted to find out if they could make some contractual arrangements with the U.S. geological survey up there, so they would have the capability of using this aircraft in summer in case we had a blow out in some of the deep oil wells they are going to be drilling between 1977 and 1980, allowing them to bring in a much larger payload th the C-130. This would allow construction of a much shorter runway at significant cost savings to them. So there has been some interest expressed by other areas of the government.

KOVEN: (Naval Air Systems Command)

Could you comment on the need for direct lift control spoilers in landing?

KISHLINE:

They essentially provide you a capability to go down or to make a much larger correction down to glide slope from above. They really provide a significant capability to get back down a glide slope if you come out high. If you did not have those, you would end up coming down a glide slope of say 9°, trying to get back on 6° at a very reduced power setting. This way, you can keep your power up and spill your lift and come down rather quickly if you are high on the glide slope. Does that answer your question?

KOVEN:

Well, I was wondering if you actually needed them or just added them to further improve glide slope control.

KISHLINE:

I think you need them for that situation. You do not always come out right where you want to be, especially when you are flying in low weather conditions and come out right where you want to be on a glide path, and it is nice to have the capability to come high and descend instead of having to fly it up to the glide path and then come down the glide path.

GOETHERT: (The University of Tennessee Space Institute)

I understand that one of the prime requirements for this type of aircraft is to be able to land on very short strips, maybe of 2000 ft

length, and if possible also on unprepared fields. I wonder, did you give any consideration to the air cushion landing gear, or do you see a future for that device with which you could land safely on any reasonably smooth surface?

KISHLINE:

No, we did not. The only thing they looked at in the design of the rear on this aircraft was a truck type gear where the rear wheels touch down first and cushion the landing, but that is the extent to which they went, with no advanced techniques on cushion landing.

GOETHERT:

I think it would be a great advancement if this type of aircraft could be equipped with an air cushion gear, since then you open up a large number of additional strips on which you could land without any extensive preparation.

KISHLINE:

Clearly if you could develop that capability, it would significantly improve your STOL capability on the aircraft. It is a matter of dollar constraint. We do not have the kinds of dollars to look at many more techniques than what we have already looked at in the prototype aircraft.

WU: (The University of Tennessee Space Institute)

YC-14 versus YC-15, as you explained, are quite different in generating high lift. Could you give a comment, aerodynamically speaking, on which one is better?

KISHLINE:

They are both very good! I have a competitive prototype program going right, now and we are going to be selecting a contractor here very shortly. You will see the results in the newspaper.

BRADLEY: (General Dynamics)

I would like to ask a question on your approach speed. I believe you quoted an approach speed of about 86 knots on the landing you showed. What do you consider to be the minimum controllable approach speed without any kind of reaction control for these transports?

KISHLINE:

We have gone down to 70 knots and in one case, I think, I saw 68 knots on file.

BRADLEY:

With no control problems?

KISHLINE:

No sir, in fact one of those prototypes landed in a 35 knot cross wind, on STOL configuration.

KEMPER: (Vought Corporation)

I have two widely different questions. One, what is the sink rate when you are landing? There does not seem to be any flare or anything. You just fly right into the ground? And second, what is the flyaway cost of one of these airplanes going to be compared to, say, a C130?

KISHLINE:

The first question, the gear is designed for 15 or 16 ft/second. The normal STOL operation is a result of the cushioning affect you get, rate of descent at touchdown between 7 and 10 ft/second. The C130 has about a 56,000 lb. dcpr weight; this aircraft will be in the neighborhood of 100 to 110 thousand lb. dcpr weight upon the configuration. We are going through a configuration review at the present time to determine exactly what the Air Force wants in its aircraft. As you are well aware, flyaway costs are pretty well dependent upon the dcpr weight, about twice. But it provides us with significantly greater capability than we have with the C130. That is a very good airplane. It has been a very successful airplane in the Air Force inventory. The problem is quite straightforward, you know. We are like the guy what had three little boys and his wife, and they had a Volkswagon sedan; now they are 200 lb. kids and they need a station wagon. You only need a tape measure to see what your problem is when you look at the Army equipment.

KOVEN:

You indicated that you had zero based all the design requirements. Could you comment on those which you found were totally inappropriate for this design?

KISHLINE:

We had the Air Force flight control spec that did not even address STOL operations. We threw that one out and wrote a new one. It was a mil spec on flying quality. It took a year, but we wrote a new specification specifically tailored for STOL operations, a good example for this one. I threw out mil specs that said I had to put all my data in military format. We have commercial engines. We took the commercial engine maintenance manual and commercial tech orders and

sent them down to San Antonio Air Logistic Center and said, "Why can't you use those?" You know, airlines are using them. They came back and said they could, so I decided not to follow that mil spec. I would have had to pay a lot of money to take those commercial manuals and put them back into military format. Those are two that come to my mind right off the top of my head.

KAILOS: (U.S. Army Mobility Research & Development Lab)

Are you getting a commercial engine warranty?

KISHLINE:

That is one we are still working with. I am inclined not to, but I am talking out of school here. The reason is because I do not think our usage is going to be the same as commercial usage. The problem is this, you pay all that money for that warranty and then do not use the engine the way you said you would. This makes the warranty invalid, and that is throwing money away. We have another unique approach. We are requiring the prime contractor to have system level reliability, maintainability, and availability goals on the aircraft. I want so many maintenance man hours per flying hours as a requirement and as a goal, the engine being part of this. We are going to perform an operational test with six airplanes and fly them for thirty days very intensively, and everything that fails counts against him. And if he does not meet his system level goal, then he is going to have to select some system on that airplane. I am not going to tell him which one, but he is going to have to select some system on that airplane and bring the total system to reliability goals he has guaranteed. Now because I am constructing a contract like that with the prime contractor, I think he is possibly going to lean on his engine subcontractor to give him some reliability guarantees on equipment, subequipment. He is going to have to sign up for a total system level reliability, a total system level maintainability. We are going to fly the airplane exactly as we told him we were going to do during this test. We have thirty days of missions and several missions for each day and will tell him every system that I am turning on and turning off, when I am turning it on, when I am turning it off, so he can calculate and sign up to both a system level reliability and maintainability goals on the aircraft. Then we will provide (that was the stick, now the carrot) a performance incentive that if he exceeds the requirements and goes towards a goal, then we have structured about eight million dollars of performance incentives as an award if he does better than he said he was going to do. If he does not do as well, he is going to have to fix it at no change in contract price. And the best reliability and maintainability for the total system will be part of the basis of award during source selection.

PROPULSION TECHNOLOGY NEEDS FOR VSTOL AIRCRAFT

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ABSTRACT

Achievement of efficient flight for high-density short-haul air travel and military logistics, sensor, or attack missions is important for sustained U.S. aviation leadership. The role of propulsion technology in attaining this goal will be reviewed. Advances in propulsion technology can have a greater impact on takeoff gross weight of VTOL compared with STOL or CTOL aircraft because of the higher propulsion system weight fraction required for vertical lift-off. An example of the sensitivity for both VSTOL and STOL short-haul aircraft is given.

Many of the long-range research programs of the NASA Lewis Research Center have goals that offer benefits to CTOL, STOL, or VSTOL propulsion systems. Several programs are reviewed, such as the SCAR Variable Cycle Engine program, fuel conservative engines, and the Quiet, Clean, Short-haul Experimental Engine program (QCSEE).

Some propulsion related studies are discussed using results for a Navy Anti-Submarine Warfare multi-purpose VSTOL aircraft, such as operation at low throttle setting, the use of water injection to increase thrust at engine-out-conditions or maximizing vertical lift-off weight. In addition, the application of turboprop engines for fuel conservation on this long loiter time mission will be shown.

INTRODUCTION

In 1975 a study group composed of NASA, DOD, and FAA representatives surveyed government agencies, industry and the universities as an aid in planning NASA's future program in aeronautics (Ref. 1). Some of the survey findings applicable to this workshop are shown in Figure 1. Quiet vertical takeoff and landing aircraft were identified as one of two critically important developments for U.S. leadership in aviation. (The second item pertaining to efficient supersonic flight will not be addressed herein.) Potential applications include VSTOL aircraft as transports on short-haul high density routes and for military logistics, sensor platforms, and attack missions.

The history of VSTOL aircraft activities spans at least 25 years and considerable research, study, and testing of experimental aircraft has been accomplished (Refs. 2 to 4). The purpose of this paper is to selectively focus on some aspects of VSTOL propulsion in order to illustrate: (1) the contribution of propulsion advances toward a useful aircraft, (2) to show how propulsion research and development directed at other types of aircraft can impact VSTOL systems. The VSTOL propulsion system has the more difficult task, since it has several unique functional requirements. These are: (1) to provide thrust greater than the lift-off weight, (2) to convert to an efficient cruise power mode, (3) be able to contribute attitude control during takeoff, transition, and landing, and, (4) to endure a loss of power from an engine core. For commercial use, quiet operation is also necessary. Thus, these requirements cause the ratio of propulsion system to gross weight to be many

°IDENTIFIED AS CRITICALLY IMPORTANT TO U.S. AVIATION LEADERSHIP:

QUIET VTOL

EFFICIENT SUPERSONIC FLIGHT

°SHORT-HAUL AIRTRAVEL

HIGH DENSITY ROUTES - VSTOL

VARIETY OF SHORT RANGE MISSIONS FOR ARMY & NAVY LOGISTICS AND SUPPORT

°PROPULSION

CYCLES FOR EFFICIENT OPERATIONS OVER A WIDER RANGE OF FLIGHT SPEEDS AND ALTITUDES

Figure 1. NASA outlook for aeronautics - 1980-2000

times greater than for CTOL or STOL aircraft, and the reward for technology advancements is also much richer. The general requirement of different modes of efficient operation over a range of flight speeds and altitudes suggests that perhaps the concept of Variable Cycle Engines, VCE, could be applied to this subsonic problem of VSTOL aircraft. One way of achieving this might be by providing wider operating ranges with less power dependent installation penalties.

IMPORTANCE OF PROPULSION SYSTEMS IMPROVEMENTS

The impact of propulsion technology on future VSTOL aircraft can be illustrated first by observing the historical improvement in turbine engines and second by parametrically examining a typical short-haul aircraft using different state-of-the-art engines. One commonly used index of propulsion improvement of particular importance to VSTOL in the increase in engine thrust/weight ratio with time as shown on Figure 2. The upward trend of the curve indicates a twenty year improvement in uninstalled engine thrust to weight ratio of approximately 2. The year of introduction shown is only approximate. As an example, consider the F 106 interceptor airplane. In 1959 it used the J 75 engine, but today could use an engine with a thrust to weight factor twice that of the original engine, as well as significant reductions in specific fuel consumption and frontal area. These advances are the result of improvement of many factors,

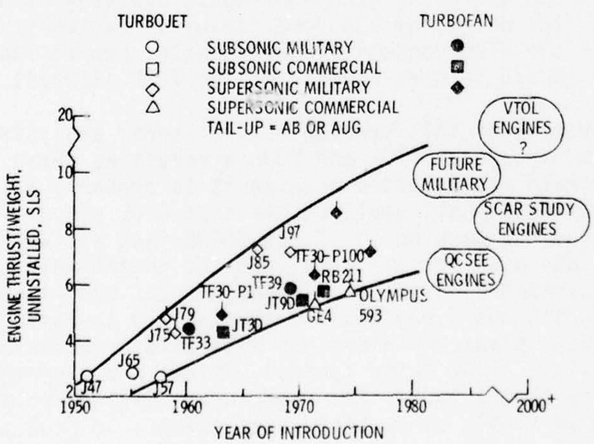


Figure 2. Engine thrust/weight trends

such as material and fabrication, mechanical, aerodynamic, and matching design procedures, turbine cooling, and related thermodynamic cycle changes. Other indexes of propulsion systems include thrust per unit of frontal area procedures, turbine cooling, etc. Other indexes of propulsion systems include thrust per unit of frontal area and specific fuel consumption, but these indexes are difficult to plot in this manner for bypass engines. Of course, engine complexity and cost are additional factors to be judged. Today's technology gives uninstalled thrust to weight ratios between 5 and 8. What can be anticipated in the future? Cycle and material improvements should continue, and contributing NASA programs will be discussed subsequently. Perhaps one thought for future engines is to observe that current short-life, restricted-cycle lift-engines offer thrust/weight ratios as high as 15 to 20. If some of these approaches to engine design can be adapted to conventional engines to drastically improve thrust/weight ratio, a compromise on engine lift might well be acceptable.

For CTOL aircraft the ratio of installed thrust to airplane gross weight is usually in the range of 0.2-0.3. To achieve the short-field capability of STOL, the ratio rises to the order of 0.6. For VTOL aircraft the ratio of thrust to gross weight must obviously be greater than 1.0 - usually 1.1-1.15 - which invests a greater portion of the total gross weight in propulsion system items. A VTOL aircraft has the added requirement of a power transfer system for driving all fans in the event of a core failure during vertical operations, since the plane otherwise would probably be uncontrollable. Thus, the weight of the power transfer system is an important item for the VTOL concept. Consequently, propulsion technology advancements should be more beneficial for VTOL aircraft.

As an example of this premise a first-order analysis was made for short-haul, 80 passenger STOL and VTOL aircraft as shown in Figure 3. The STOL aircraft was selected because it is probably a more direct competitor for short-haul applications than CTOL planes. Range was 500 miles, cruising Mach no. = .7, at 25000 feet altitude. The same engine cycle was used for all cases and the engine weight and fuel consumption parametrically changed from current base-line values. The STOL aircraft had 2 engines and short-field takeoff capability, whereas the VTOL plane had 4 engines with thrust vectoring (or rotation) capability, a low speed control system, and one-engine-out power transfer to the fan of the dead core. Engine technology changes were programmed simply as combined step changes in fuel consumption and an index of engine weight. The calculation involves finding the fuel-balanced aircraft and engine sizes that will fly the mission with the specified payload. Consequently, the necessary engine size reflects these combined effects and the resulting uninstalled thrust/weight ratio is shown (including the non-linear variation of engine weight with size). The results are indicated in terms of gross

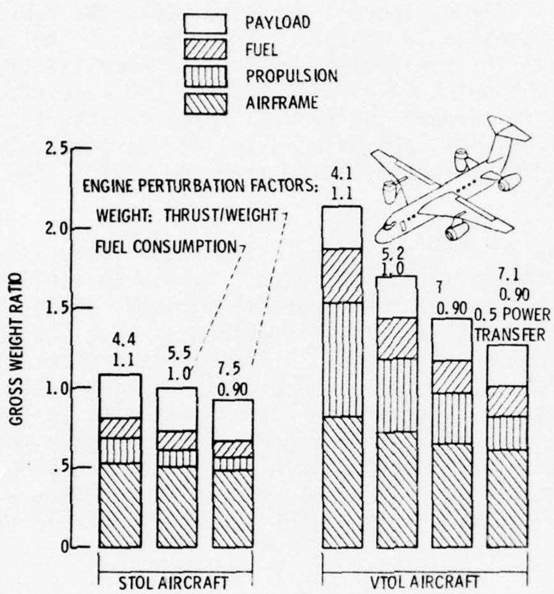


Figure 3. Effect of engine weight and fuel consumption on takeoff gross weight for STOL and VTOL short-haul aircraft (80 pass., 500 n.mi. range, M_{cruise} = 0.7 at 25000 ft.)

weight ratio relative to "today's STOL" for a step backward and forward in engine technology for both types of aircraft. The appropriate weight fractions of the major components are indicated.

The sensitivity to combined changes was relatively small for the STOL aircraft with gross weight ratio (GWR) varying from 1.07 to .94, whereas the VTOL GWR varied from 2.15 to 1.43 with a further reduction to 1.28 indicated for a power transfer system weight reduction of 1/2. For this example the VTOL engine thrust/weight ratio varied from 5.2 to 7.1. Thus, in the distant future the penalty for VTOL operational capability could become even smaller with further increases in engine thrust/weight ratio. But, because of the complex requirements of VSTOL operations the weight and fuel

used will probably always exceed those of STOL or CTOL planes. Therefore, in terms of direct operating costs (DOC), the VTOL aircraft would be more expensive to operate. The classic criterion of minimum DOC is inadequate for evaluating the overall benefits of the VTOL concept. From the point of view of society the more encompassing answer would be to compare the overall total transportation system costs, i.e., smaller, less congested satellite airports, lower community noise, lower surface transportation time and cost to the airport, etc. (Refs. 5 to 7).

Another way of viewing the relative sensitivity is shown in Figure 4 where the perturbation factors have been applied only one at a time. In the top of the figure the greater sensitivity of VTOL compared with STOL aircraft is evident. Here the abscissia is uninstalled thrust/weight ratio. It should be noted that the QCSEE engine to be discussed later has a projected uninstalled thrust/weight ratio between 6.2 and 7.4. The sensitivity to fuel consumption shown in the middle of the figure also is higher for VTOL aircraft because the fuel fraction is higher, since the propulsive weight fraction is much higher. The fuel consumption changes shown are rather modest, and advances in engine cycles could well exceed the SFC

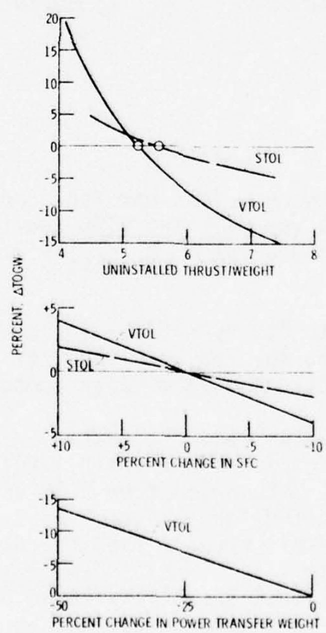


Figure 4. Sensitivities

reductions shown and in, addition, offer secondary benefits, such as reduced frontal area and lower drag. In the bottom of Figure 4 the sensitivity to changing the power transfer system weight for the VTOL aircraft is shown. Here, again, the sensitivity to propulsion related empty weight is rather high, a 50 percent reduction decreased gross weight 13 percent. This shows not only the importance of power transfer system components, such as shafts, gears, bearings, or gas ducts and valves, but also the need for configuration arrangement and control systems to allow the distance between thrust units to be reduced. It should also be remembered that sensitivity is a two-way street; hence, failure to meet design goals would be more detrimental.

NASA PROPULSION TECHNOLOGY PROGRAMS

An overview of some of NASA's propulsion technology programs is presented with the intent of highlighting activities that either directly or indirectly have a potential impact on VSTOL engine capability. The programs selected for discussion, shown on Figure 5, include the aircraft fuel conservation technology program, hot-part thermal-barrier coatings, variable cycle engines, and the quiet, clean short-haul experimental engine. Many other such activities, such as low speed inlet and nozzle performance, gears, shafts, bearings, and lubrication have been omitted for brevity.

Aircraft Fuel Conservation

The fuel conservative engine program is directed at a new generation of engines which would have fuel conservation as a primary design objective. As shown in Figure 6, the initial projected cost was 175 million dollars spanning fiscal years 1975 to 1983. The improvement goal is about 12 percent in specific fuel consumption, 5 percent in direct operating costs (DOC), noise not to exceed FAR 36 minus 10 EPNdB, with 1980 emission levels. The program

°FUEL CONSERVATIVE ENGINES

°THERMAL-BARRIER COATINGS

°SCAR VARIABLE CYCLE ENGINES

°QCSEE - QUIET, CLEAN SHORT-HAUL
EXPERIMENTAL ENGINE

Figure 5. NASA propulsion technology programs

GOAL: IMPROVE SFC 12%, DOC 5%
NOISE FAR 36-10
1980 EPA EMISSIONS

FY'75-83, $\$175 \times 10^6$

°ADVANCED TURBOFANS, TURBOPROPS, GEARED FANS
FAN AND COMPRESSOR PRESSURE RATIOS
TURBINE TEMP. - COOLING TECH.
SEALS, BEARINGS, LUBRICATION

°ADVANCED CYCLES-REGEN., REHEAT, INTERCOOLING

°AMSAC-ADV. MULTI-STAGE AXIAL FLOW COMP.

°MATE-MATERIALS FOR ADV. TURB. ENGINES

Figure 6. Fuel Conservative engine - new designs

proceeds from feasibility, to component rig/model tests, to full scale engine tests. Study areas include the components of advanced turbofans, such as fans compressors, turbines and cooling techniques, plus mechanical components such as seals and, perhaps, gears. The advanced turbofan engines (Ref. 8) show specific fuel consumption reductions of from 13 to 15 percent relative to today's turbofans but require very high overall pressure ratios and turbine temperatures. The selection of such advanced cycles for VSTOL engines would have to be evaluated on a benefit/cost basis. The AMSAC and MATE programs are examples of on-going research. AMSAC is aimed at advancing the technology of very high pressure ratio compressors. This technology might be applicable to VSTOL engines in terms of reducing the number of stages and, hence, compressor weight required to achieve any given pressure ratio. The goal of the MATE program is to accelerate development and demonstrate in engine tests advanced materials technologies for engines for the 1980-1985 timeframe.

Thermal-Barrier Coatings

If specific thrust can be increased by raising turbine temperature without offsetting penalties in weight, cooling airflow, or expensive blading, a better engine thrust/weight ratio can be used to increase payload or decrease size or cost. Alternately, if cooling air bleed flow requirements can be reduced, specific fuel consumption can be improved.

An area of research related to such engine performance improvement is that of thermal-barrier coatings applied to hot parts, such as turbine blades and vanes or combustor liners. Promising results have been obtained in a variety of applications. A two-layer, thermal-barrier coating consisting of a stabilized zirconia coating over a metal bond coating was plasma sprayed over super alloy substrate material. Figure 7 shows turbine vane test results obtained in a research engine. Vane temperature reductions of about 190 deg K (342 deg R) were obtained. Several different coatings and bond coatings in a range of thicknesses were evaluated in a series of aerodynamic, hot fatigue cycle, and full scale engine cycle durability tests. In a particular engine cycle durability test, 500 two-minute cycles from full power to flameout were accomplished. The best combination, to date, based on durability and costs was Yttria stabilized zircronia ceramic (.028 to .064 cm, .011 to .025 in) over a NiCrAlY bond coating (about .01 cm, .004 in). There is a limiting ceramic-bond interface temperature of 1367 deg K (2416 deg R). The temperature reduction is a function of coolant flow rate and coating thickness and material.

Figure 8 shows calculations of gas temperature versus coolant to gas flow rate for convection, convection plus thermal-barrier coating, and full-coverage film cooling. At the present level of turbine gas properties the cooling requirement can be decreased by a factor of three. The calculations indicate that convective cooled blades with the thermal-barrier coating could be as effective as full-coverage film cooling without the costly blade surface cooling passages. At the turbine gas properties anticipated for future advanced engines the indicated coolant flow rates would be less than today's uncoated blades. This would result in better cycle performance. For example, at subsonic cruise, changing coolant bleed flow from 7 to 2 percent would reduce fuel consumption about 5 percent.

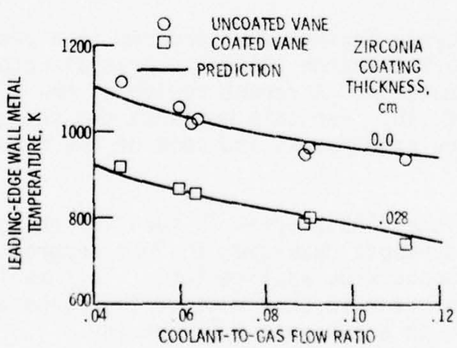


Figure 7. Turbine vane test results

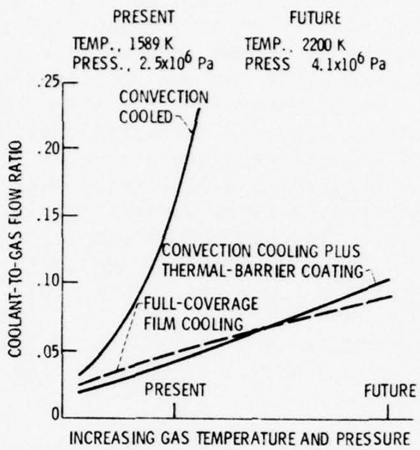


Figure 8. Cooling requirements for several cooling methods

The coating technique has been successfully applied to combustor liners, and many manufacturers have brought in components, such as diesel piston heads and valves for coating and evaluation following company tests.

SCAR Variable Cycle Engine Program

The Variable Cycle Engine (VCE) program is a part of the Supersonic Cruise Aircraft Research (SCAR) program directed at both civil and military applications. A recent review of the civil VCE program can be found in Ref. 10. For this workshop the scope, objectives, the two prime engine candidates, and some of the background material will be reviewed.

By definition the SCAR program is focussed on supersonic flight. However, some VCE concepts developed in that program are being studied for military supersonic VTOL applications. This review of the VCE program is given in order to show some of the component features that might be applicable to a subsonic VSTOL engine.

First, a variable cycle engine is functionally defined as an engine which has at least two distinct modes of operation: (1) a high airflow, low jet velocity mode for low takeoff noise and/or

A SERIES OF ENGINE STUDY CONTRACTS AIMED AT IMPROVING THE OVERALL PERFORMANCE OF SUPERSONIC CRUISE AIRCRAFT HAS PROGRESSED TO A COMPONENT TEST PROGRAM

°OBJECTIVES

DEMONSTRATE COANNULAR NOISE REDUCTION IN LARGE SCALE
EVALUATE PERFORMANCE OF UNIQUE VCE CONCEPTS
PROVIDE BASIS FOR FUTURE EXPERIMENTAL ENGINE

°TESTBED PROGRAM BASED ON EXISTING CORES

P&W - F100, DUCT BURNER, COANNULAR EJECTOR NOZZLE
GE - YU101, VARIABLE FLOW FAN, COANNULAR PLUG NOZZLE

Figure 9. SCAR variable cycle program

efficient subsonic cruise; and (2) a turbojet-like higher jet velocity, lower airflow mode for good supersonic cruise.

An overview of the VCE program is given in figure 9, which shows that the series of engine study contracts (cross-fertilized with airframe studies) has progressed to a test of engine components. Objectives are to: (1) demonstrate the coannular noise reduction in large scale (discussed later), (2) evaluate performance of unique VCE components (discussed later), and (3) provide a basis for a future experimental engine. The test bed program is based on existing cores with hardware added in order to statically test the previously identified objectives.

The concept of coannular noise benefit as shown in Figure 10 (see insert) requires that the outer stream velocity be significantly higher than the inner stream and that the radius ratio be high. Small scale test data has been used to calculate sideline noise relative to FAR 36 in EPNdB versus jet velocity (of outer stream) for a full scale multi-engine airplane. For single stream conventional nozzles the sideline noise exceeds the FAR 36 level for jet velocities greater than about 1700 feet/sec; however, the coannular nozzle does not exceed FAR 36 until about 2200 feet/sec. Thus, the noise signature with coannular nozzles promises to be 8 to 10 dB lower, and the VCE concept may be able to operate below FAR 36 since it can take off at lower jet velocities.

The first VCE engine to be discussed is shown in Figure 11 and is referred to as a Variable Stream Control Engine (VSCE); three operating modes are depicted. This engine has the flow path of a

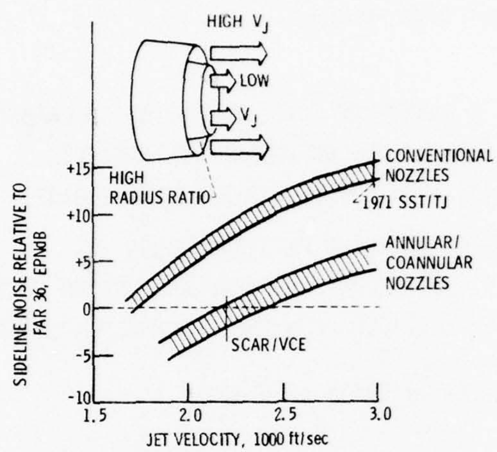


Figure 10. Annular/coannular noise benefit

conventional duct burning turbofan but, incorporates a unique main combustor power schedule and extensively controls rotor speed, as well as variable geometry in the fan, compressor, and both nozzles to regulate the operating bypass ratio. For subsonic cruise the duct burner is not lit, and the bypass ratio is about 1.5. During takeoff the duct burner is lit for added thrust and for higher outer stream jet velocity to give the coannular effect. At supersonic cruise the core is speeded up by increasing main combustor temperature and by manipulating variable geometry features, thus reducing bypass ratio and fuel consumption.

The second VCE concept, termed a double bypass engine, is shown in Figure 12. The fan is divided into blocks of stages with an auxiliary duct from the interface; flow control valves (not shown) regulate the path of interfan air and rear fan air through different flow paths in the engine for different operating modes. Some of the auxiliary duct flow can be routed to the exhaust plug for coannular noise suppression. Thus, at takeoff the front fan block is "high flowed"; the core is at maximum power. At subsonic cruise the auxiliary duct is open and passes the excess airflow provided by the front block; thus, at constant airflow a wide range of throttling is possible, thereby minimizing throttle dependent inlet and boattail drags. For higher power operation the auxiliary duct is closed and the tailpipe heater is lit as needed. In this mode the cycle is essentially that of a low bypass engine.

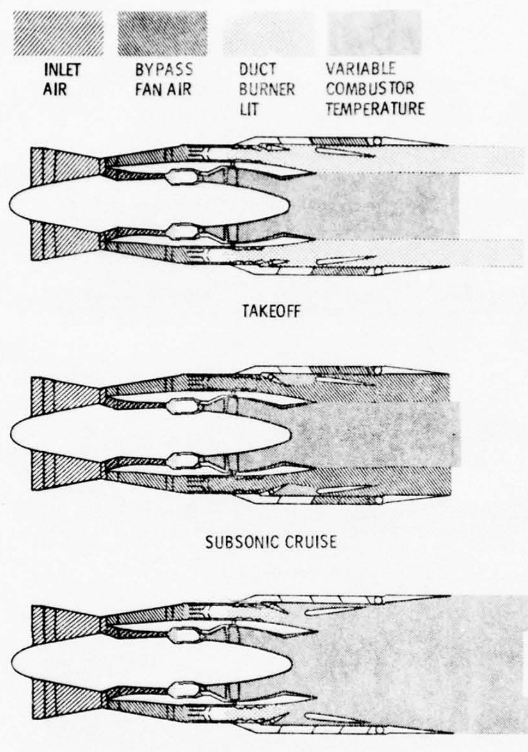


Figure 11. Variable stream control engine

The combined cycle and noise reduction benefits can be seen in Figure 13 where range has been increased 25 percent and noise reduced 8 EPNdB compared to turbojet powered aircraft. Although not shown the mission performance with large subsonic segments also benefits.

It must be admitted that this discussion of VCE concepts is in terms of a different application than the subject of this paper. However, a typical VTOL mission is similar in many respects: the need for two different operating modes (high-thrust takeoff and low-thrust cruise) aggravated by the engine-out concern; low takeoff noise; the need to accommodate highly throttled operation with good efficiency, etc. It seems quite plausible that future studies may help identify a potential for some form of a VCE to solve these problems for subsonic VTOL applications.

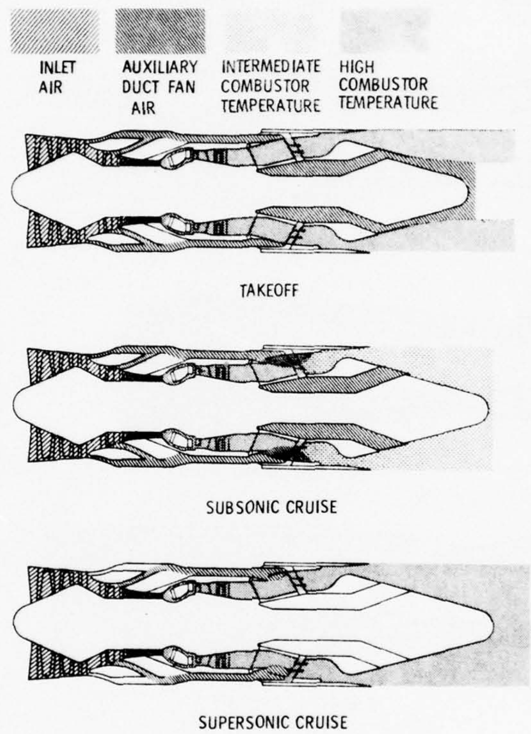


Figure 12. Double bypass engine

Because of the high installed thrust to lift-off weight of VSTOL aircraft, operation at cruise or loiter conditions requires generally low throttle settings with concomitant penalties in thrust specific fuel consumption as shown in Figure 14. A typical loiter throttle setting is about 30 percent and about 50-80 percent for cruise. The difference in installation effects indicated is related to the flight conditions and the particular nacelle configuration (inlet and nozzle effects). One option for reducing this effect is to shut down half of the cores and power the fans with the remainder (2 fans on 1 core or, for the previous short-haul transport, 4 fans on 2 cores); thus, the core throttle setting would be increased to a more favorable SFC, and lapse rate effects would need to be considered. This is, in a sense, a variety of variable cycle. Another possibility is to investigate whether or not a true VCE might be adapted to this mode of operation and thereby allow full airflow at reduced power by

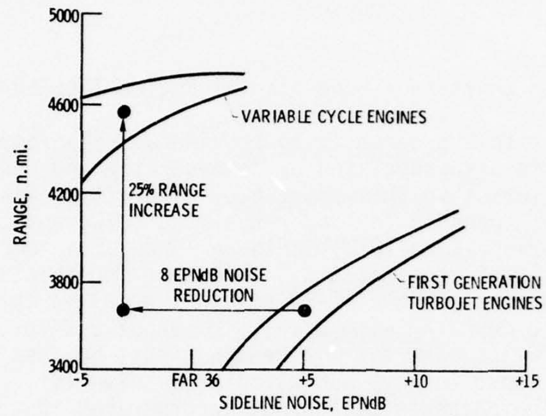


Figure 13. Potential impact of advanced technology on aircraft range and noise

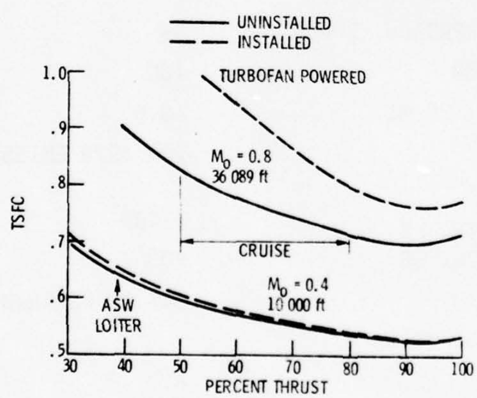


Figure 14. Effect of part throttle operation and installation penalties on thrust specific fuel consumption

making use of variable geometry components, flow paths, and/or power schedules.

Quiet Clean Short-haul Experimental Engine (QCSEE) Program

The goal of this program is to provide a technology base for powered lift aircraft propulsion by demonstrator engines and, thus, is of direct interest to this short-haul aircraft workshop. One configuration is intended for under-the-wing (UTW) application and another for over-the-wing (OTW). Figure 15 details some of the noise, pollution, and thrust requirements. The noise footprint is about 10 times lower than that of the DC-10, and the installed thrust/weight ratios are to be compared with a value of about 3.5 for the CF6 engine in the DC-10. The dynamic response must be fast because of operating from short runways and using propulsive lift. Figure 16 lists some of the advanced technology incorporated in the designs. The variable pitch fan is lighter and has faster response than a conventional thrust reverser. The inlet is designed to nearly choke the inlet flow in order to prevent the emergence of fan noise; the actual throat Mach number was set at 0.79 in order to avoid excessive flow distortion. Reduction gearing reduces the number of low pressure turbine stages. The bypass and fan pressure ratios are 11.8 and

NOISE, 500 FT SIDELINE		
TAKEOFF & APPROACH, EPNdB		95
REVERSE, PNdB		100
FOOTPRINT AREA, SQ MI		<0.5
POLLUTION		EPA 1979 EMISSION LEVELS
INSTALLED THRUST		
FORWARD	UTW, LB	17400
	OTW, LB	20300
REVERSE		35% OF FORWARD THRUST
INSTALLED THRUST/WEIGHT		
UTW		4.3
OTW		4.7
DYNAMIC RESPONSE		
APPROACH TO TAKEOFF THRUST, SEC		1.0
REVERSE THRUST, SEC		1.5

Figure 15. QCSEE technical requirements

HIGH BYPASS RATIO ENGINES

VARIABLE PITCH FAN

VARIABLE AREA FAN NOZZLE

HIGH MACH INLET

DIGITAL ELECTRONIC CONTROLS

REDUCTION GEARING

COMPOSITE COMPONENTS

FAN BLADES

FAN FRAME

NACELLE

Figure 16. Advanced QCSEE technology

1.27 for UTW to reduce jet/flap interaction noise and 10.8 and 1.34 for OTW, respectively. Both engines use the F 101 core, which gives a compressor pressure ratio of about 14. The specific fuel consumption is 0.71. The indicated composite material components, in particular, are considered advanced technology. The fan frame material is graphite fibers and epoxy resin.

A cut-away drawing of the UTW engine is shown in Figure 17. The primary distinguishing features are the reversible thrust, variable-pitch composite blade (18) fan and the variable flare fan nozzle and core plug nozzle. This engine has an uninstalled thrust/weight ratio of 6.2. During recent reverse thrust tests, this engine suffered significant fan damage when one of four exhaust nozzle flaps failed. These flaps are in the outermost position for reverse thrust and have been given the coined word "exlet." NASA Lewis tests are scheduled for February 1978.

The OTW engine, shown in Figure 18, has many common features, but has a fixed-pitch, titanium fan to lower costs, a target type thrust reverser for over-the-wing discharge, and a variable area mixed flow "D" nozzle for better flap performance. The uninstalled thrust/weight ratio is 7.4. NASA Lewis tests are scheduled for October 1977. A double annular dome low emissions combustor has been designed (NASA Clean Combustor Program), will be tested, and is intended for installation in the QCSEE engines. Some of the features used to suppress noise are shown in Figure 19. Regions of acoustic treatment are indicated, as well as blade spacing ratios. The rotor/

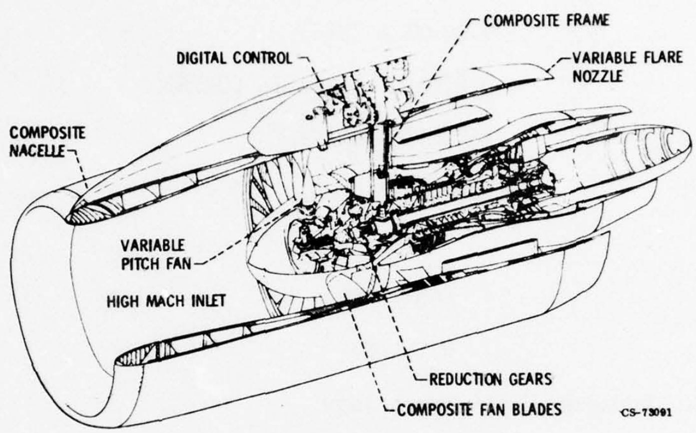


Figure 17. QCSEE UTW engine

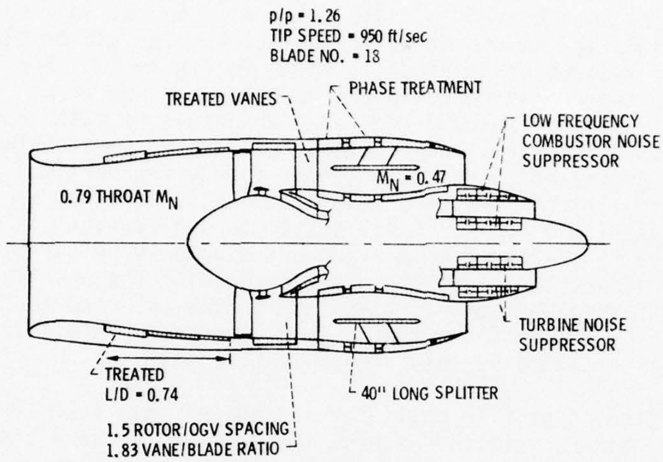


Figure 19. QCSEE UTW engine

guide vane spacing ratio was selected to reduce blade interaction noise and the vane/blade ratio for reduced blade passing frequency noise.

The technology, concepts, and performance being developed in the UTW QCSEE engine make it a good candidate for use in a VTOL airplane. The engine thrust/weight ratio is high (6.2), noise footprint is very good, the variable pitch fan can be used in matching the core for operating more than one fan (4 fans on 2 cores) during cruise as well as for generating attitude control vectors, and fan water injection could be used for better emergency thrust rating (discussed subsequently).

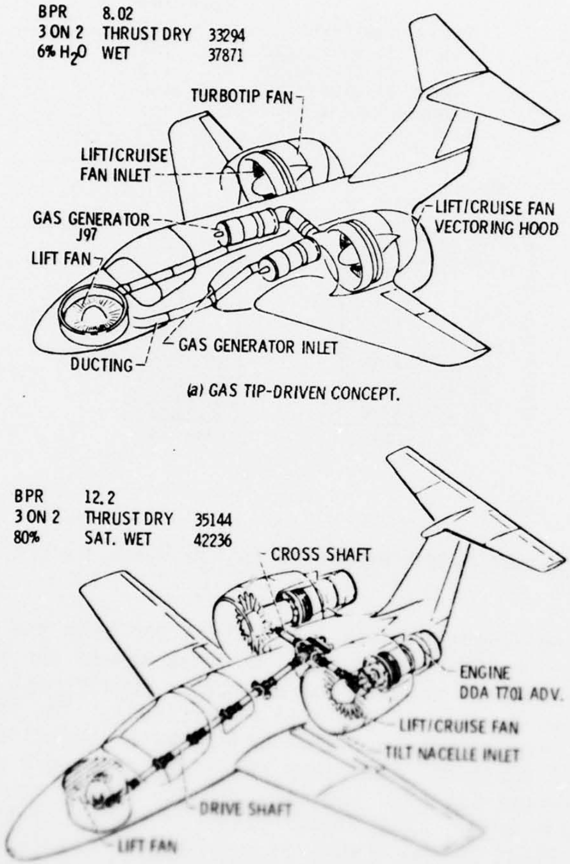
SOME VSTOL PROPULSION STUDY AREAS

Several VSTOL propulsion applications will be illustrated using an in-house study of the early proposals for the Navy lift-fan Anti-Submarine Warfare airplane.

Water Injection Performance

thrust when vertical lift-off is required and providing for emergency vertical landing in the event of core failure. One possible solution to these problems is the use of water injection. The two different lift fan power systems considered are shown in Figure 20. For the GAS system, the two cruise turbotip fan nacelles and one front turbotip fan are powered by two body mounted gas generators with appropriate gas ducting, valves, and controls to transfer power between nacelles (for engine-out) and to the front fan during vertical operation. Vertical thrust for the cruise nacelles is attained by vectoring the exhaust flow. For the SHAFT system the arrangement is similar, except that the fans are variable pitch units powered by turboshaft engines, which offers the advantage of supercharging the nacelle cores. Power is transferred between nacelles (for engine-out) and to the front lift fan by means of shafts and gears; vertical thrust from the nacelles is attained by rotating the entire unit.

The ASW mission profile calls for a range of only 150 miles with maximum loiter time on station at best loiter speed at an altitude of 10,000 feet. The proposed powerplants had fans of approximately the same diameter (150 cm, 59 in GAS and 158 cm, 62 in SHAFT). The GAS system turbotip fan was powered by a J 97 gas generator giving a bypass ratio of 8.02. For the SHAFT system the DDA T701 (advanced) turboshaft core was used giving a bypass ratio of 12.2. No attempt was made in this phase of the study to optimize bypass ratio. Thus, with the respective engine sizes and thrust ratings fixed and with somewhat different airplane geometric parameters, the empty weights and aerodynamics were calculated. Fuel was added until the ratio of takeoff thrust to gross weight was 1.08 for VTO and >.8 for STO. The mission flight profile was flown and the resulting loiter time on station determined. Figure 21 shows the WET/DRY thrust ratios for various operating modes, the ratio of emergency thrust to landing weight, and a comparison of loiter time on station for the two systems. Two cases are compared for each system and in both cases the water injected emergency thrust rating was used. However, in the first case, labeled DRY, the water system was installed for emergency use only. In the second case, labeled WET, water injection was used for routine normal takeoff operations, thus providing greater lift-off thrust. Normal rated turbine inlet temperature is not exceeded for WET or DRY. However, for emergency power after loss of an engine the remaining engine is operated at a short-life, overtemperature rating. The bar graphs show the loiter time for vertical and short take offs. The SHAFT system uses water injection aft of the fan, whereas the GAS system uses combustor injection. The dry takeoff performance of the two systems are nearly equal; although the SHAFT system has greater power transfer weight, the fuel consumption is better due to the supercharged core and a bypass ratio of 12 compared with 8. Water injection shows no benefit for short take offs; however, for vertical take offs the VTO time on station is at least doubled by using water injection and for the SHAFT system, almost equals the STO performance. The tabular data on the top of the figure shows ratio of emergency thrust [illegible] to



(a) GAS TIP-DRIVEN CONCEPT.

(b) SHAFT-DRIVEN CONCEPT.

Figure 20. - VTOL aircraft configurations.

Figure 20. VTOL aircraft configurations

		SHAFT	GAS
WET/DRY THRUST RATIO:			
VTO	3 FANS ON 2 CORES	1.20	1.14
STO	2 FANS ON 2 CORES	1.07	1.13
EMERG.	3 FANS ON 1 CORE	1.06	1.13
EMERG. THRUST/LAND WT:			
DRY		1.05	.908
WET		1.116	1.027

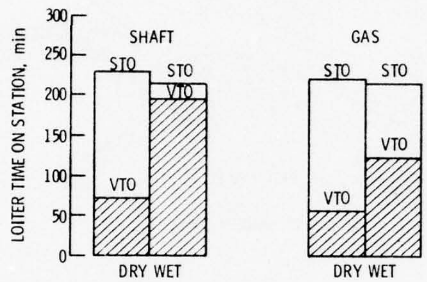
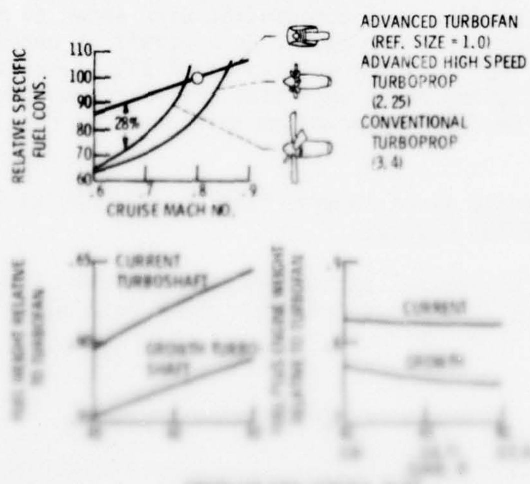


Figure 21. Effect of water injection on ASW VSTOL aircraft

emergency landing weight for the two systems for both the dry and wet modes. Based on the system modeling used herein the GAS unit in the dry mode did not satisfy the emergency requirement and, in the wet mode, gave a ratio of 1.027 (1.03 desired). The SHAFT system easily met the requirement in either mode. Water injection (of the respective type) increased emergency thrust by about 6.2 percent for SHAFT and 13 percent for GAS; however, the GAS system required about 3.7 times as much water. The higher ratio of emergency thrust to landing weight shown in the figure is primarily a result of the higher baseline thrust of the SHAFT engine. These benefits of water injection must be weighted against the perceived disadvantages, such as decreased dynamic stability and greater vibration, decreased life of hot parts, the logistics of providing sufficient pure water, etc. Another question usually raised about compressor face water injection is: what happens on a humid day? At the temperature and pressure aft of the fan, enough water could be injected to give about 70 percent of the dry day performance; if necessary, an additional combustor injector could be provided for full emergency thrust.

Fuel Conservation by Using Turboprops

Large fleets of ASW airplanes accumulating many thousands of hours of patrol mission time suggest that the use of turboprop engines should be studied in the national interest of fuel conservation. Accordingly, a study was made using the same turboshaft engine as the previous ASW study, but driving conventional propellers. The configuration assumed tilting turboprop nacelles and a small horizontal tail propeller for thrust balance. Results are shown in Figure 22. In the top of the figure is a comparison of relative specific fuel consumption versus cruise Mach number for advanced turbofan, conventional turboprop, and turboprop with advanced high-speed propeller engines. The conventional turboprop offers 28 percent fuel saving compared with the turbofan at Mach number 0.65. Although some ASW multi-purpose mission profiles call for M=0.8, the percent of mission time at that condition is small. If some compromise in the mission flight profile is acceptable, turboprops were considered a viable candidate; therefore, the study was concentrated on the 4 hour loiter performance. The curves of fuel weight and fuel plus engine weight for current and growth turboshaft engines relative to turbofans (same core) as a function of propeller disk loading are shown in the bottom of the figure. Fuel savings of between 35 and 45 percent, depending on disk loading and core, are attractive. Since



the trend of propeller plus core plus gear box weight is the reverse of that for the fuel weight, the combined effect appears rather flat as a function of disk loading as shown in the right hand part of Figure 22. The savings in fuel plus engine weight vary from 17 to 25 percent compared with turbofan propulsion. This saving could be invested in a smaller airframe to do the mission. The propeller diameters varied from 18 to 12.8 feet with disk loading. The smaller diameters are more attractive from a configuration point of view, but cost several percent in fuel consumption. Mini-shrouds might be considered for further diameter reductions. The advanced high-speed propeller turboprop is included in the top of the figure for comparison of the relative high speed performance; however, the high-speed propeller technology is not applicable to the application previously given where a high value of static thrust to horsepower is needed in the vertical mode. This is the result of designing to minimize high-speed losses and gives very thin, low cambered blades, which is the reverse of what is required for high static performance. If, however, the airplane has no vertical lift-off requirement, the advanced high speed propeller would be a possible choice.

SUMMARY

The possible impact of advances in engine thrust/weight ratio and reduction in specific fuel consumption were shown to be of greater significance for short-haul VTOL aircraft as opposed to STOL or CTOL aircraft. For military multi-purpose VTOL aircraft the potential benefit of using water injection for emergency power was shown, as well as the fuel conservation that could be realized by using turboprop engines.

QCSEE could serve as an interim VSTOL engine and will aid development of future VSTOL engines.

Various NASA propulsion technology programs were reviewed with the suggestion that these programs would contribute to VSTOL propulsion advancement.

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DISCUSSION

REESE: (Purdue University)

You talked mostly about the propulsion problems. NASA is doing work on the engine-airframe integration to reduce aft end drag and to understand those flows. Is a report on that work scheduled? Would you care to comment on the front end and back end engine-airframe integration questions?

ALLEN:

I should have mentioned that, because of time, I was not going to address many other related programs such as low speed aerodynamic testing of inlets and exhaust nozzles. If that is the kind of thing you had in mind, I am not familiar with what is being done, except that there are active programs involving engine-airframe integration.

GOETHERT: (The University of Tennessee Space Institute)

You showed that by going to the high-speed propeller, you get much better fuel consumptions. I wonder, how does this relate to noise?

ALLEN:

I do not know. I know the advanced high speed propeller [illegible]

has work to be done in the noise area, and they are also looking in that program at what you have to do to the fuselage to dampen out propeller noise and things of that sort.

I think your question refers to Figure 22, which indicates lower fuel consumption at large propeller diameters compared with turbofans. A noise comparison was not calculated in that study.

GOETHERT:

Could you at least give us some indication on the noise? Is it not so that for some of your test points, with high or low fuel consumption, you may have high speeds? I do not know whether you go into supersonic RPM-range, and then the noise should be very bad.

ALLEN:

I do not know what the noise in comparison would be.

GOETHERT:

Could I have another question? I noticed that most of your curves were shown for thrust-to-weight ratios (of the engine) up to 7 or 8. This ratio would go up considerably for lift engines which are designed for short operation time. Can't we expect, then, a gain in thrust-to-weight ratio, say, up to 10 or 15?

ALLEN:

That is where we would like to be, and if you use some of the design procedures used in the lift jet engines, you may be able to get there if you compromise perhaps on engine life.

HILL:

What happens if one considers the ducted or shrouded propeller type of propulsion system? Is there anything special that develops? Or does it fit in between the fan jet and the propeller in efficiency and cruise Mach number of best performance and so forth?

ALLEN:

[illegible]

turbofan and a turboprop in terms of installed specific fuel consumption and weight.

HILL:

That is essentially the same kind of problems that you have with a high bypass turbofan also, though. I was wondering if there was not the possibility of essentially compromising between the two.

ALLEN:

I think there could be, and Reference 8 does point to a contractor study that covered ducted shrouded propellers in comparison with turbofans in terms of weight and performance. Basically, adding the shroud introduces parasitic losses, such as shroud drag, internal ducting, and nozzle pressure losses, which can become very dominant at the low pressure ratios associated with propollers.

DENNING: (Rolls-Royce (1971) Limited)

You have put a lot of emphasis in your talk on high thrust-to-weight ratios and coating for two inch blades - to get the leading edge temperatures down - and water injection and composite materials; as I have understood, the Navy is very interested in reliability. And, I am wondering whether you think the two can go together?

ALLEN:

You are thinking about the coating surviving water injection?

DENNING:

That sort of thing, yes. But generally the philosophy of design of the jet engine: whether it is compatible to use all these advanced technology aspects of engine design and still retain the sort of reliability the Navy is asking for, where you have 20 men sitting in an airplane supported on 3 fans with cross-shafting?

ALLEN:

Well, first I think the designer seems to favor putting the reliability into the fan rather than the core. They think they can make the fan much more reliable, so they provide for core failure by means of cross-shafting. The thermal-barrier coatings are being tested now with further treatment to toughen them up and to make them more impervious to chemical action. The reliability of the total system is something you will have to evaluate, judge the complexity, and take your choice, I guess. I tried to indicate the benefit of advances in technology at one extreme of the cost - reliability - benefit evaluation.

PINSON: (University of Dayton)

I want to ask a question in regard to performance of the advanced turboprop propeller. For years, we have been seeing curves showing propellers cruising at Mach 0.85. Is there sufficient research in being (not the propfan, but related to the turbo-prop) to indicate that this is feasible?

ALLEN:

Well, I did not indicate any conventional turbo-prop cruising at 0.85. The performance used in those calculations was from the Hamilton Standard Propeller Performance Estimation Manuals. Reference 8 presents an overview of 1950 research propellers and operational propellers. At Mach 0.85, the highest efficiency for research propellers was about 0.7. The advanced high-speed propeller (prop fan) model SR-2 achieved 0.79 efficiency at Mach 0.8.

ENGINE-AIRFRAME INTEGRATION, CURRENT PRACTICES AND FUTURE REQUIREMENTS FOR ARMY AIRCRAFT

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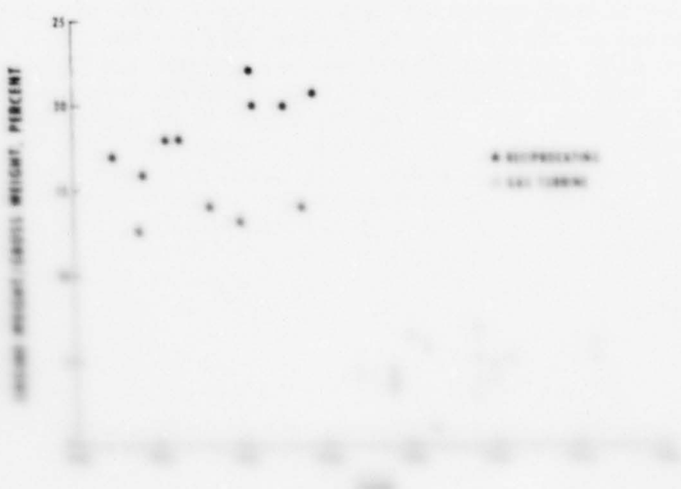
ABSTRACT

When turbine engines were introduced for helicopter propulsion, they were not readily accepted, even though dramatic improvements in load-carrying capabilities were promised. There were many debates on affordability and reliability, which led to questions on whether gas turbines could prove themselves suited to such vehicles. The questions were valid, to some extent, in that they dealt with whether this relatively new power plant could survive the helicopter environment. However, many of the negative aspects were rapidly erased, and the benefits were exploited just as quickly; this was largely due to the interest of the US Army during the 1960's. Now, in a relatively short time, practically all new helicopters are turbine powered. In addition, many old vehicles are being converted from reciprocating engines to turbines. But, despite the emphasis on dedicated research and development programs, many new and unforeseen difficulties have surfaced and need to be overcome. This paper discusses some of these difficulties, such as: the complex dynamic conditions that can promote fatigue; the punishing operational environments; the unpredictability of flow patterns around the vehicles and engine inlets, particularly in low-speed and hover; the demands on complex propulsion control systems aimed at reducing pilot workload; and the deteriorating effects of continuous power cycling of the engines, with flights usually less than one hour in length. As a result of the experience and background gained during recent development programs, it has been necessary to discard earlier practices and develop all-new standards of acceptability.

1.0 INTRODUCTION

The advent of turbine engines has added significantly to the utility of helicopters and has opened new areas of application. Although most of the developments have been aimed at military needs, we now find that many of these vehicles meet unique requirements that could not be met by any other machine. However, it has been an easy transformation, and helicopters still demand heavy engineering attention to complex details. Foremost among the areas of continuing concern is engine-airframe integration. The demands for reducing pilot work load have, in themselves, introduced new design requirements along with more complex and more costly subsystems.

Perhaps a few illustrations to describe progress over recent years might be appropriate. Figure 1 shows one of the major reasons for adopting the turbine engine. As can be seen, the large reduction in power plant weight led to dramatic improvements in load-carrying capability. Not too long ago, when reciprocating engines were the main power source, the engine alone accounted for roughly 15 percent of the vehicle gross weight. Now, the figure is about 5 percent. While we have reduced engine weight, we surely would be in a stronger position if we could make further improvements in fuel consumption and reliability. Figure 2 shows typical curves of fuel consumption versus horsepower for two representative levels of technology. Although the gains have been impressive, the penalties at part power in the range of normal flight must occupy a large share



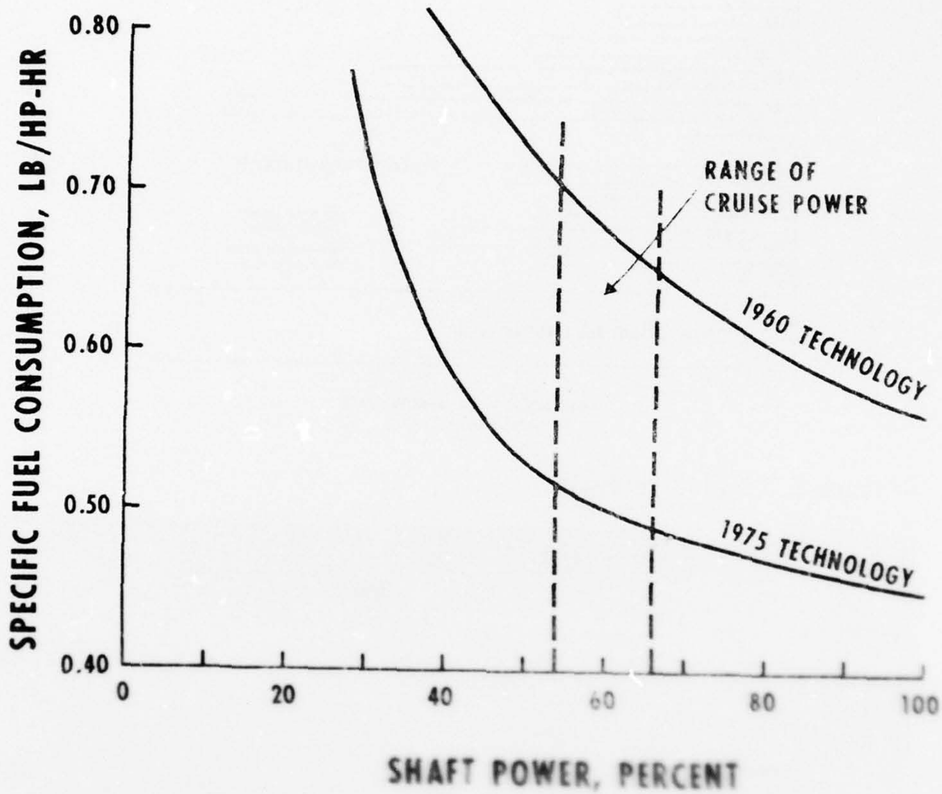


Figure 2. Turboshaft Performance Trends

of our attention, especially in view of the widely acknowledged concern over continuing fuel shortages. As related to reliability, too, the gains have been notable, but this is the single most important element to be considered in viewing the future of helicopters. Figure 3 shows the removal rate of helicopter engines as compared to those used in fixed-wing aircraft.

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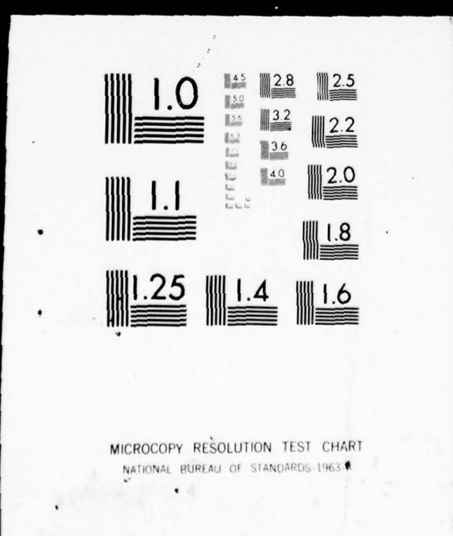

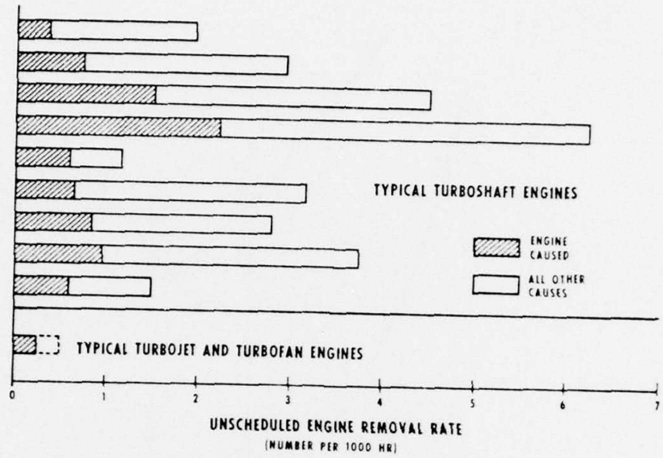

Figure 3. Engine Removals

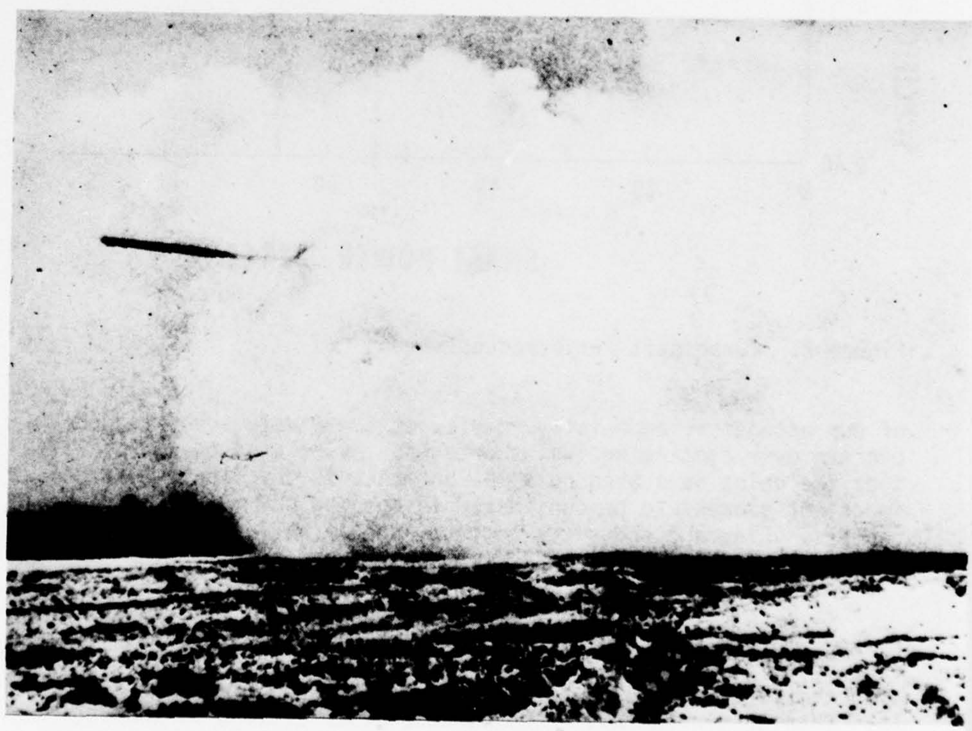

Figure 4. Sand and Dust Environment

conditions that are extremely difficult to handle. An indication of the severity imposed is shown in Figure 4. Hidden well inside that dust cloud is a helicopter, and you can appreciate the effects on man and machine, alike. The engines, of course, will ingest the contaminated air unless some form of protection is provided; therefore, new engines being introduced into the Army inventory will have integral particle separators as part of the engine. Thus, we have given attention to at least one aspect of the solution, but we are still far from an understanding of the interface problems related to engine placement and the variety of inlet conditions that the engine will face. Despite this concern, the newest engine to enter the inventory, the T700, thus far has shown a commendable degree of reliability and attention to interface that plagued earlier development projects. An illustration of that engine is given in Figure 5. As can be seen, it includes the integral particle separator mentioned earlier. This engine has been chosen to power the Army's Utility Tactical Transport Aircraft System (Figure 6) and the Advanced Attack Helicopter (Figure 7). Very likely the engine, or versions of it, will find a home in commercial and other military vehicles.

The trend toward the lightweight and small propulsion package usually is accompanied by the term "high performance." To the Army, this means pushing toward higher pressure ratio cycles with higher turbine-inlet temperatures. For the large commercial and military aircraft powered by turbine engines, it is safe to say that they are where we want to be. Figure 8 shows a typical plot of specific fuel consumption versus specific power for the small turboshaft engine of interest to us. Also shown is the influence of compressor pressure ratio and turbine inlet temperature on performance. Just as with the large turbojets and turbofans, we are trying to move downward and to the right. Today, we are in the shaded area. But we are talking about engines that are dramatically different in size, as illustrated in Figure 9. This kind of difference, alone, introduces problems unique enough to force emphasis on performance-related technologies beyond those considered in large engines. Not only does it prevent us from moving rapidly down the technology trail, but it plays a role in making our interface with the airframe a more complex one to manage or to predict. Some of our concerns are easily recognized, such as: shaft speeds as high as 50,000 rpm and more; cooled turbine blades less than one-inch in height; compressor blades less than a half-inch tall; and small but high-speed bearings. For this reason, the nature of our limitations may be different in many respects from those normally faced in other vehicles. Therefore, we have had to learn different lessons, and applying them has been both expensive and time consuming.

Of course, there are many items critical to proper integration of the engine and airframe. To us, it means achieving overall compatibility, including consideration of a variety of operating

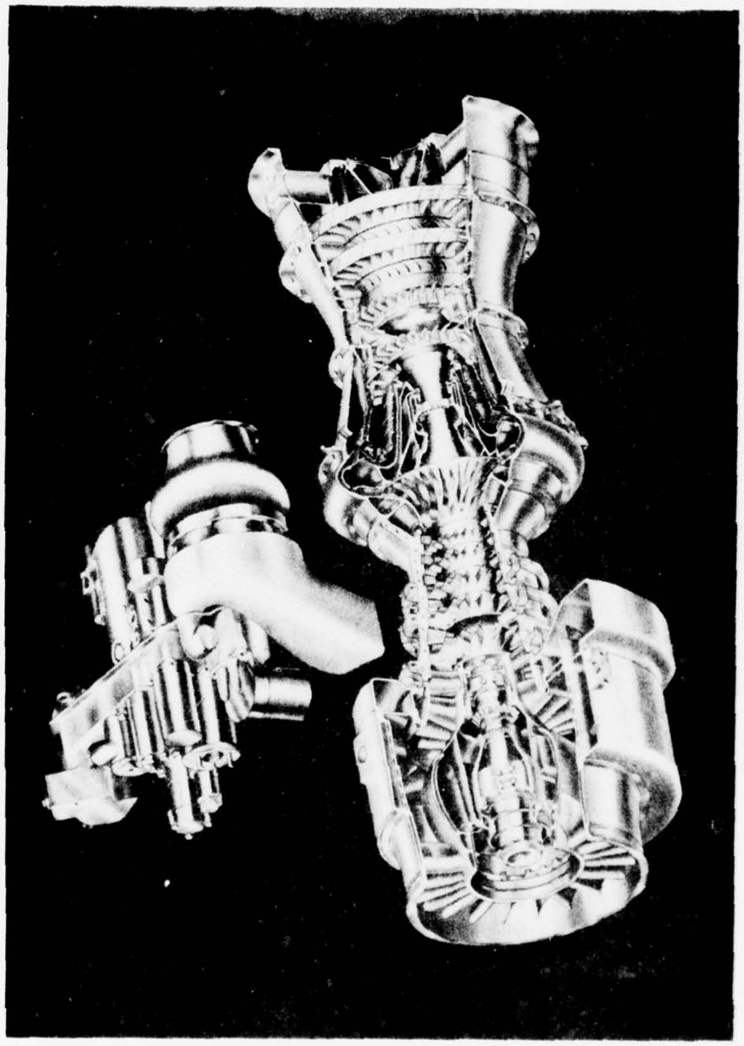

Figure 5. T700 Engine

Figure 6. Utilizing Tactical Transport Aircraft System

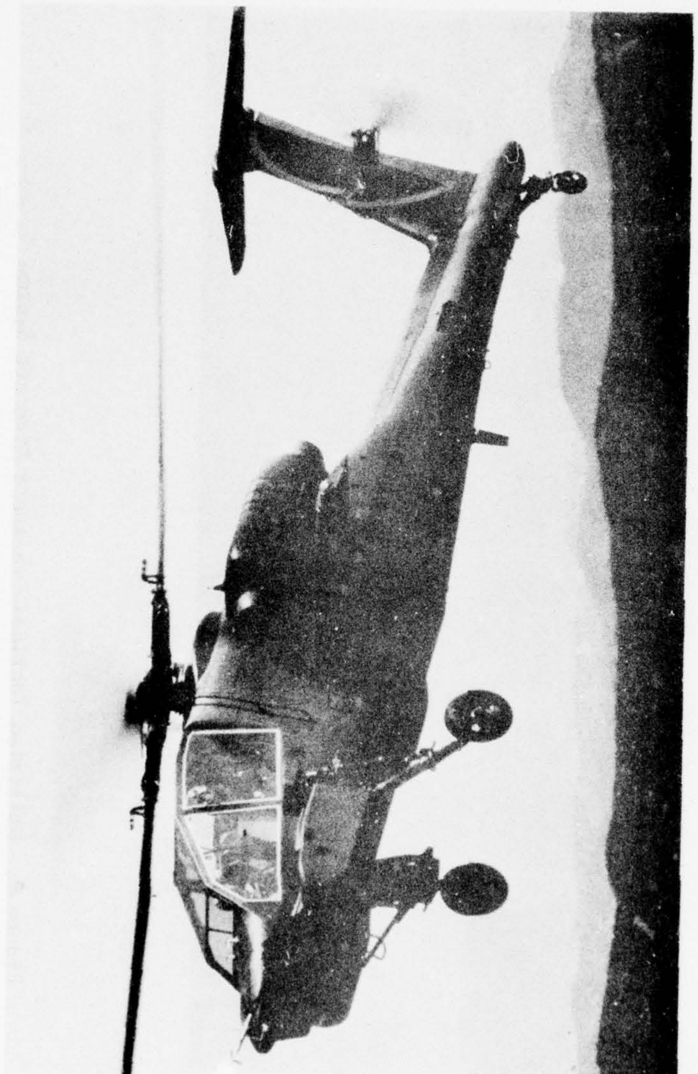

Figure 7. Advanced Attack Helicopter

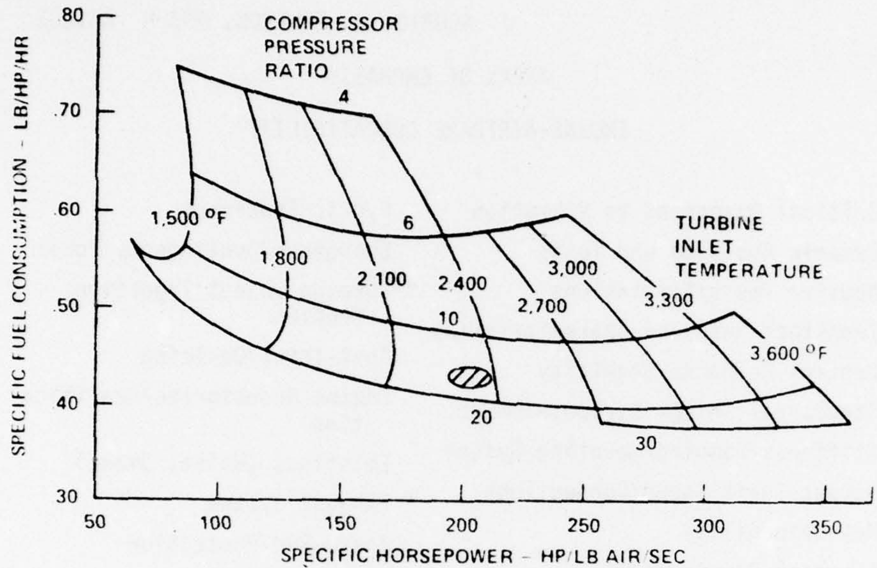

Figure 8. Influence of Turbine - Inlet Temperature and Pressure Ratio on Engine Performance

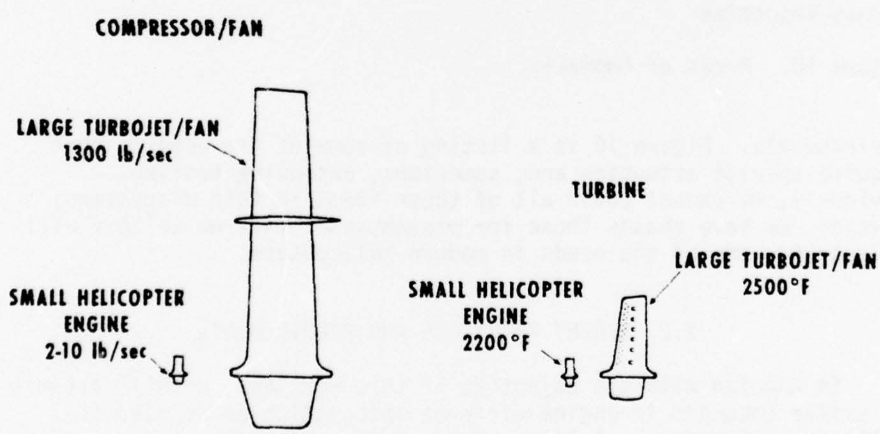

Figure 9. Component Size Comparison

AREAS OF EMPHASIS

ENGINE-AIRFRAME COMPATIBILITY

* * Critical Responses to Vibration
* Dynamic Analyses and Tests
* Abusive Tests/Simulations
* * Transient Response Characteristics
* * Control Response/Stability
* Structural Integrity/Containment
* Stiffness/Damping/Mounting System
* * Output-Shaft Speed/Connections
* Maintainability
* Starting Requirements
* Electronic Interference Signals
* Cyclic Endurance
* Emergency/Contingency Power
* * Foreign Object Ingestion/Erosion
* Anti-Icing/De-Icing
* Engine Accessories/Qualification
* Emissions (Noise, Smoke)
* * Exhaust System
* Infra-Red Protection
* * Inlet Distortion
* Subsystem Interfaces (Electrical, Hydraulic, Pneumatic, Fuel, Oil)
* Hand-in-Hand Communication Among Parties (Government and Contractors)
* Fire Detection
* * Heat Rejection

* DISCUSSED IN PAPER

Figure 10. Areas of Emphasis

environments. Figure 10 is a listing of some of the aspects that require special attention and, sometimes, extensive testing. Obviously, we cannot cover all of these items in this discussion; however, we have chosen those for presentation that we believe will illustrate some of the needs in modern helicopters.

2.0 RECENT PRACTICES AND REQUIREMENTS

In keeping with the objective of this workshop, we will attempt to evolve concepts in engine-airframe integration as related to military helicopters. A large share of the experience in that area has been with the US Army, and the primary emphasis has been on gas turbines. For that reason, some of the Army's recent practices and problems will be reviewed prior to addressing future requirements and advanced concepts. Due to difficulties encountered during recent engineering development/qualification testing and after field operations, our present-day helicopters have added to the lessons

learned. In particular, over the last ten years we have been faced with some new system development programs, such as the Utility Tactical Transport Aircraft System (UTTAS) and the Advanced Attack Helicopter (AAH), both of which have given us an opportunity to apply the lessons. In addition, we have pursued major modifications or continued development of the AH-1 (Cobra), the CH-47 (Chinook), and the OH-6/OH-58 Light Observation Helicopters (LOH).

Throughout these programs, there have been two major concerns--one is the development requirements described by the specification, and the other is the end use of the vehicle in the Army environment. Bringing these two aspects together realistically, without introducing unnecessary development cost and, at the same time, accounting for past problems so as to avoid repeating mistakes, has been the dilemma. It forces a continual reassessment; although the process has not eliminated the problems, it has paid great dividends in the UTTAS and AAH.

2.1 Turbine Engines

Prior to 1971, the Army had generated requirements based on MIL-E-8593 through MIL-E-8597 series engine specifications, which were generated in the mid 1950's. Although tri-service approved, these specifications were primarily Air Force documents. The two points to be made here are that the specifications were relatively old and were primarily concerned with turboprop engines. Many important design considerations for helicopter applications were not addressed. Examples of such omissions are polar moment of inertia and torsional spring constants (referred to engine output-shaft speed) that define stability and compatibility requirements for installed operation of the engine/fuel control/aircraft rotor drive system. Others are turbine overspeed and overtemperature protection and demonstration requirements; environmental concerns, such as sand, dust, and ice ingestion; and structural aspects involving low-cycle fatigue and engine stiffness.

The initial efforts and experience in the early 1960's showed that if turbine engines were to be successful in the Army environment, major new design and test requirements would be needed. Briefly, the Army environment meant that turbine engines would be exposed to the deteriorating effects of high concentrations of sand and dirt, high vibration levels, temperature extremes, and structural fatigue. It was clear that considerable enlargement and redefinition of design requirements were necessary; therefore, in early 1970, the Army began the task of generating its own engine specification. Primarily, this specification was aimed at development of the new 1500 SHP turbine engine, later to become the T700-GE-700 engine now used in the UTTAS and AAH. The intent of the new specification

was to take into account the expected technology level, to address the Army environment, and to close the gap between the design requirements and the demonstration tests.

In part, this activity later contributed to a new tri-service MIL-E-8593A specification, which was approved late last year after a lengthy and detailed coordinated effort.

2.2 Airframes

Until recently, engine-airframe interfaces were defined in a general sense, with no specific design or demonstration tests required by specification. Initial programs, such as the Light Observation Helicopter (OH-6/OH-58), were based on requirements established by the Federal Aviation Administration (FAA). By this approach, it was necessary that some specific requirements be negotiated with the FAA. With that recognition, some programs attempted to adopt Air Force and/or Navy requirements, but, in general, they were slanted toward fixed-wing aircraft and were not totally satisfactory. Therefore, in the late 1960's, through a series of in-house investigations and industry contracts, the Army undertook preparation of the Army Designers Handbook for Detail Design and Test Assurance (AMCP 706-202 and 706-203). The Test Assurance volume (706-203) was incorporated in all subsequent helicopter projects, including the UTTAS and AAH. While primarily an air-frame-oriented document, it established requirements for propulsion systems "Surveys" and "Demonstrations" in all propulsion-related areas, and it placed special requirements on the airframe prime contractor as the system integrator.

To establish general compatibility, propulsion system surveys are required early in the airframe development program. The preferred approach is to use "YT" Preliminary Flight Rated (PFR) engines, so that if engine or airframe design changes are required, they can be identified early enough to be implemented prior to full engine qualifications. Along with general compatibility, the surveys include engine installed vibration, propulsion system cooling, air induction, and exhaust. The airframe contractor is required to investigate his propulsion systems thoroughly prior to finalizing his design and entering into the official demonstration. These demonstrations are expanded surveys, but are directed toward specific requirements (test points) in both ground and flight tests. They are the basis for the approval of:

a. engine air induction system

b. propulsion system temperature

c. engine vibration

d. lubrication subsystem

e. fuel subsystem

f. engine-airframe compatibility

g. transmission and drive train

h. clutches, brakes, shafting, couplings, and bearings

i. accessories

3.0 HELICOPTER FIELD DATA

To establish design criteria for future helicopters, it has been necessary to update and incorporate information on field use, based on various combat, supply, and utility missions. Typical of these recent studies are one performed for the Southeast Asia arena and one conducted in the Alaska environment. In both cases, several types of helicopters were instrumented extensively to establish operating conditions, power excursions, and general use of engines so as to provide background and feedback for design purposes. Initially, the data were used to assess helicopter structural loads and to analyze helicopter drive system operation. Data were obtained from 336 flight hours of AH-1G, 216 hours of OH-6A, and 203 hours of UH-1H operation during combat missions in Southeast Asia and about 160 hours of OH-1H operation in utility missions in Alaska. A summary of the pertinent engine flight data gathered from the Southeast Asia operation is shown in Table 1.

TABLE 1. ENGINE FLIGHT DATA

	AH-1G	UH-1H	OH-6A
Total Flights	259	249	218
Total Flight Time (hr)	336	203	216
Total Engine Starts	342	242	242
Average Altitude (ft)	2000	1780	2400
Average OAT (°F)	85	84	92

For this paper, interest is focused on the load spectrum, the flight duration, the number of starts, and the number of rapid power excursions. Our concern has been to define properly those specification requirements which must be met to realistically represent field use. For example, it gives us some insight into the stress-rupture design requirements for the hot end of the gas turbine, in addition

to providing background for bearing design/life criteria, low-cycle fatigue (LCF), and overall deterioration. Today, our new specifications call for 5,000 hours design life with the expectation that 15 percent of this time (750 hours) will be spent at Intermediate Rated Power (IRP). The specification also requires a capability to perform 15,000 LCF cycles (3 LCF cycles per hour).

Figures 11 through 14 present only a few of the histograms of the percentage of time at various power levels and flight lengths. The report from which the data have been derived shows some significant operational differences. For example, the UH-1H helicopters in Alaska spent about three-fourths of the time in the 60 to 65 percent power range, while those in Southeast Asia averaged about one-fourth of the time in this range. For comparison, during short and medium length flights with the AG-1G, an average of 40 percent of the time was spent above 60 percent power. With these variations, it is clear that we cannot establish specification requirements that will apply universally. The same concern is illustrated in Table 2, where rapid power excursions for the three helicopter types operating in Southeast Asia are shown.

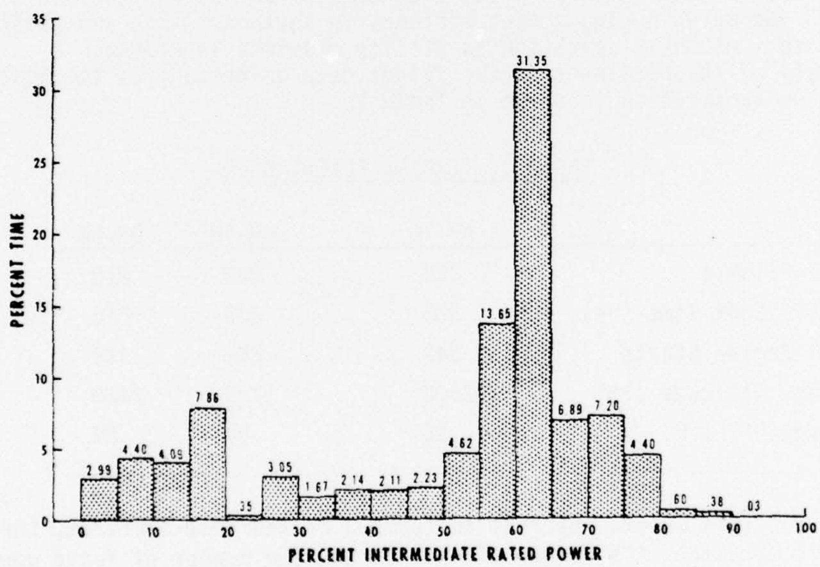

Figure 11. Time and Power, Single - Engine Helicopter (Combat Support)

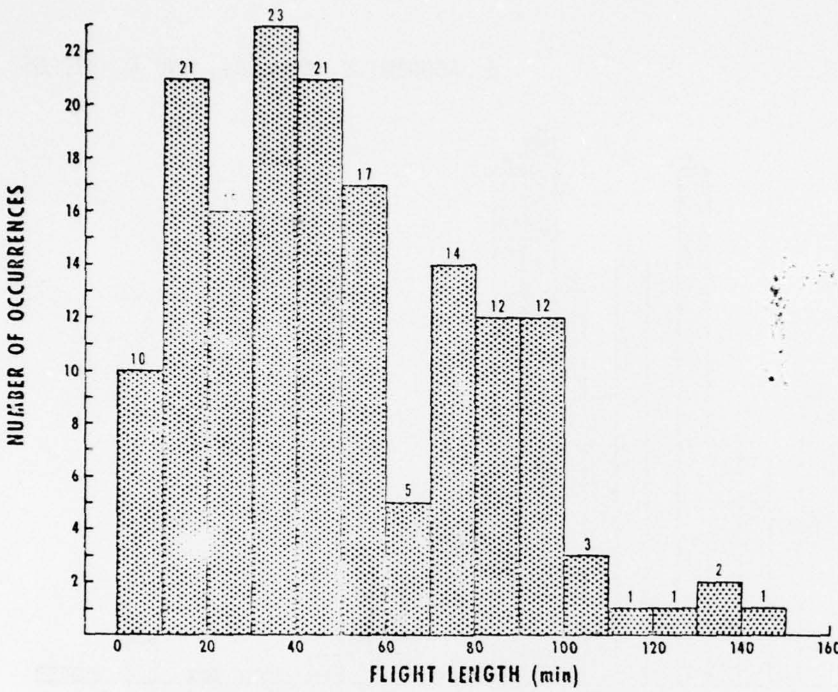

Figure 12. Flight Length, Single - Engine Helicopter (Combat Support)

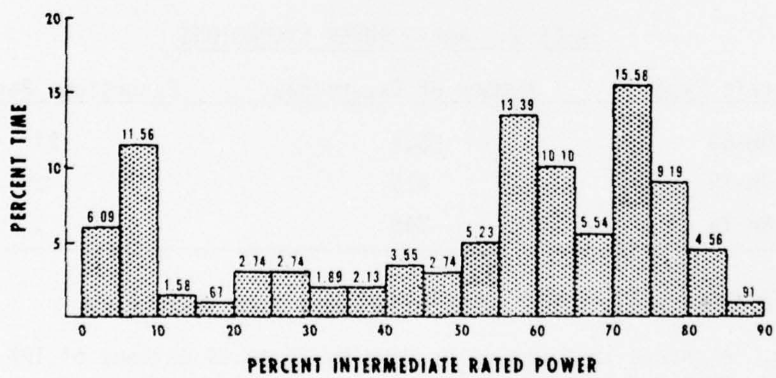

Figure 13. Time and Power, Single-Engine Helicopter (Combat (Assault)

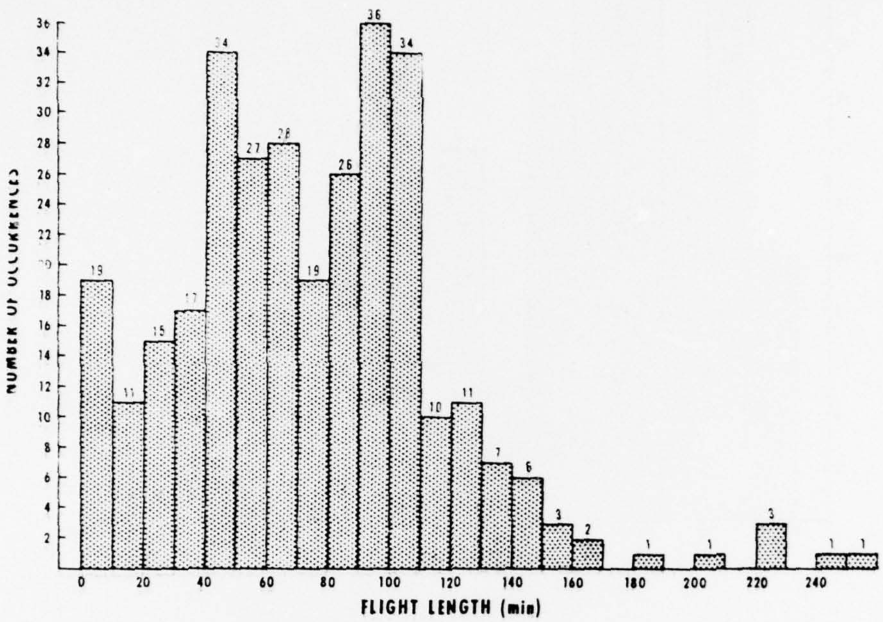

Figure 14. Flight Length Single - Engine Helicopter (Combat Assault)

TABLE 2. RAPID POWER EXCURSIONS

Aircraft Type	Number of Excursions	Excursions Per Hour
OH-6A	5884	27
UH-1H	415	2
AH-1G	348	1

In this study, rapid power excursions were defined as:

a. a change in power up or down by 25 to 34 percent of IRP within 3 seconds or less

b. a change in power up or down by 35 to 40 percent of IRP within 6 seconds or less

c. a change in power up or down by 50 to 75 percent of IRP within 9 seconds or less

As can be seen, the OH-6A experienced over ten times as many rapid power excursions as the UH-1H. Again, the difference is due, in large measure, to their basic missions. Another factor to be considered is that the data for the AH-1G and the UH-1H were obtained during the early years of the conflict when the helicopters flew at relatively high altitudes with little hover time so as to avoid small-arms fire. On the other hand, the OH-6A appears to have been used more extensively to fly low-level, high-speed missions, similar to nap-of-the-earth (NOE).

Based on these studies, one can conclude that our requirements to design for 15 percent of the engine's life at IRP is conservative and will enlarge our field capability. As regards the requirement for 15,000 LCF cycles, again it would appear that we are conservative, based on the UH-1H and AH-1G data. However, we now have a further complication with present emphasis toward NOE type of flight, and we must determine whether the 5,000-hour design life with 15 percent of this time at IRP is adequate. In this case, if the OH-6A data are representative, our 15,000 cycles (LCF) design requirement might not meet the need. This remains as a continuing area of concern, and we are still struggling to devise representative test cycles to account for field usage. Thus, we need to find ways to correlate and accelerate testing performed at the factory, so as to shorten the development and qualification time.

4.0 SELECTED AREAS FOR R&D EMPHASIS

Although it is clear that the new development requirements described in the documents noted above have avoided a repeat of some of the many earlier obstacles, the complexity of engine-airframe integration leaves many areas open for continued research and development. They include engine heat rejection, structural integrity, high engine-output-shaft speeds, air-induction systems, power management (controls) engine vibration limits, and engine-airframe dynamics.

4.1 Heat Rejection

There has always been a motivation to improve gas turbine efficiency in terms of Specific Fuel Consumption (SFC) and power-to-weight ratio (HP/WT). Along with these desires, and of equal or greater importance to the user, is acquisition cost. One way to reduce engine cost is to reduce the number of parts, particularly the rotating elements. During the past decade, there have been remarkable gains in gas turbine component technology. Some of them partially satisfy this desire for lower cost; as an example, Figure 15 illustrates an increased compressor loading (work per

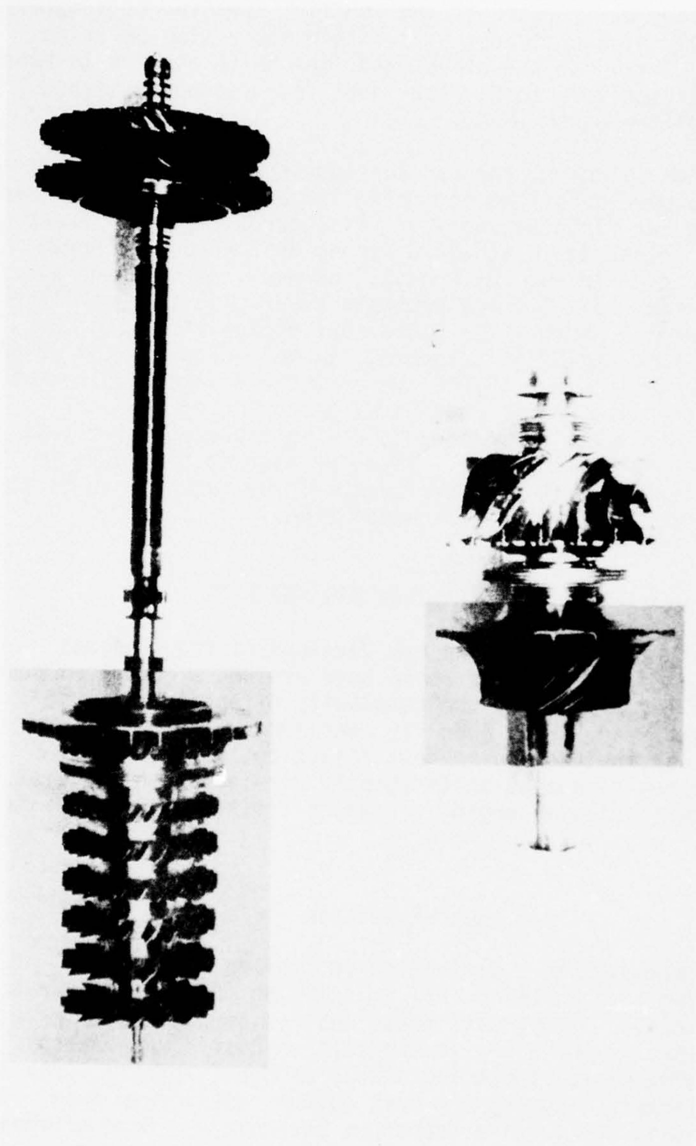

Figure 15. Increased Stage Loading

stage), by which fewer compressor stages are needed to reach the desired pressure ratio. At the same time, efficiency improvements have demonstrated that gains in SFC, as well as cost, are being realized. While the push toward higher pressure ratio and increased gas temperature has been essential, we also find that we must give more attention to heat rejection from the engine bay. With the new technology, compressors are operating with discharge air temperatures in excess of 800°F; consequently, what has been referred to as a cold section of a gas turbine is no longer cold, and the higher surface temperatures present a greater challenge to the airframer. Therefore, we must recognize the requirement to provide extra cooling at several key locations to prevent critical controls and accessories from exceeding their temperature limits.

Beyond these temperature-related concerns, the desire for ease of maintenance at the depot level usually means more split lines for the compressor section. For small engines, it introduces more chances for high-pressure leakage and a likelihood that hot air and gases will impinge on critical areas in the engine compartment. Although the burden is on the engine designer to prevent leakage, the airframer now must recognize the potential difficulties, and must locate the aircraft controls (actuators) and route the fuel lines and electrical harnesses in such a way that they will not be affected. Obviously, the engine and airframe manufacturers must work even closer together during the entire development program.

Before initiating the 1500 SHP Advanced Technology Demonstrator Engine program (1966), the Army solicited comments and suggestions from the helicopter industry in an attempt to incorporate their desires wherever appropriate. This approach proved to be valuable, and many suggestions were adopted. One criticism was that the engine specifications did not provide the needed information on heat rejection; the typical required information provided to the airframer is shown in Figure 16. It was a very simple display of surface temperatures at several locations, and there was no indication of the amount of cooling required. Meeting the requirements for cooling accessories and controls involved expensive flight tests, and it was common to find trial-and-error solutions to overheating problems. Therefore, the joint effort, stimulated by the 1966 exercise, resulted in a new approach to defining requirements as shown in Figure 17. The recently revised and approved MIL-E-8593A engine specification now incorporates heat rejection information similar to this illustration. Note that the curves show the amount of heat removed (BTU/HR/IN), along with the effect on surface temperature. In the area of the combustor case (stations 218 to 222), for example, if no heat were removed by the airframer (stagnant air), the surface temperature would be in the range of from 932°F (500°C) to 1202°F (650°C). If 4,000 BTU/HR (1000 BTU/HR/IN x 4 inches) were removed, the surface temperature would drop to a range of from 572°F (300°C)

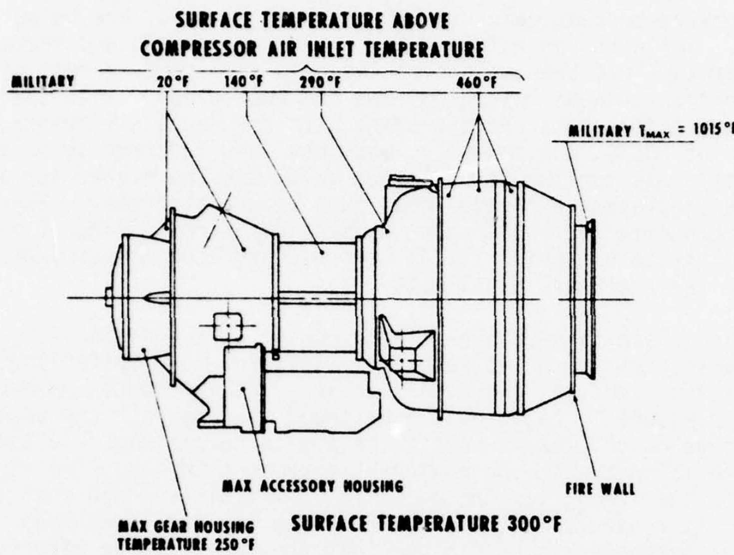

Figure 16. Engine Surface Temperatures (T53)

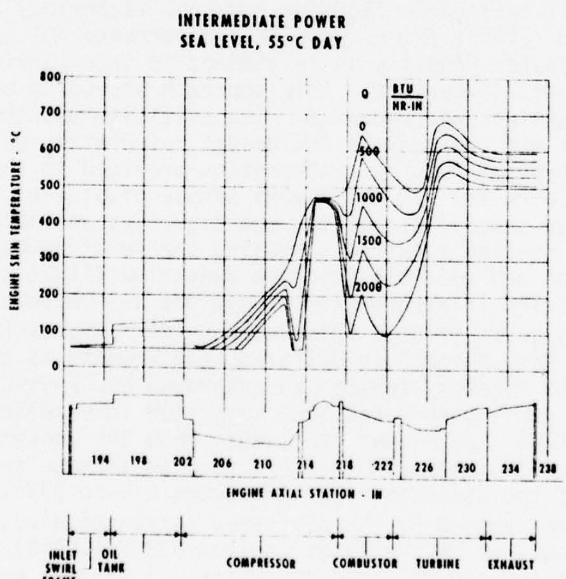

Figure 17. Engine Surface Temperatures (T700)

to 842°F (450°C). A different example can be developed at station 216. Here, near to the centrifugal compressor, the internal air velocities are high, and cooling the external surface will be quite difficult. In addition, space where accessories and controls can be cooled effectively is limited, and the challenge to the airframer is increasing with each generation of engines. At the same time, the engine manufacturer must consider using engine-mounted components that are capable of withstanding the higher temperature environment without malfunctions. This point is emphasized when recalling the potential leakage of hot air and gases through joints and flanges. The effect on performance can be expensive, in addition to compounding the cooling problems.

To place the performance penalty in perspective, every one percent of compressed air lost results in about a 3 percent loss in power. In small engines, the amount of air lost through the flanges can be as high as 2 percent of the main flow, and it should be recognized that leakage losses need much more attention than those in large turbojets and turbofans using the same design approaches. The problem can be reduced to the simple relation shown in Figure 18, where it can be seen that the potential leakage area compared with the core mass flow increases with smaller engines. At the same time, the trend toward a small size for a given amount of power, brought about by component technology improvements, diminishes the available space for mounting temperature-critical components. Figure 19 shows this trend. However, because of the need to use inlet particle

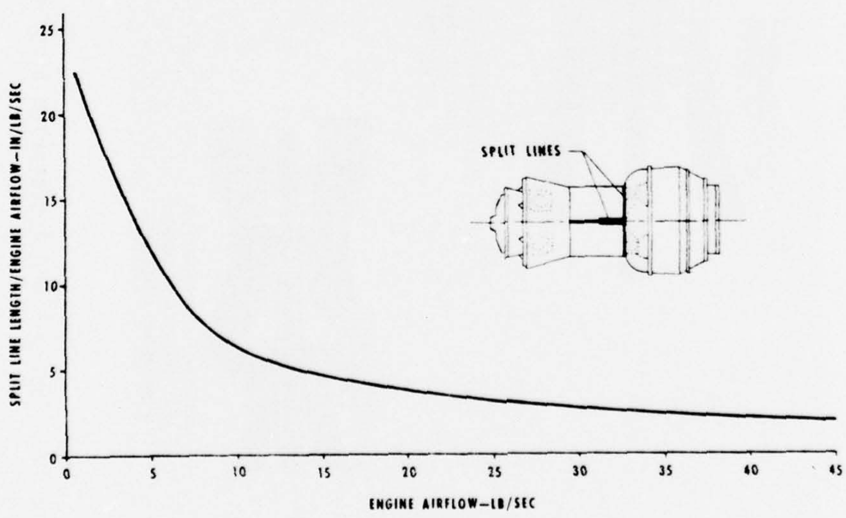

Figure 18. Typical Flange Leakage Areas

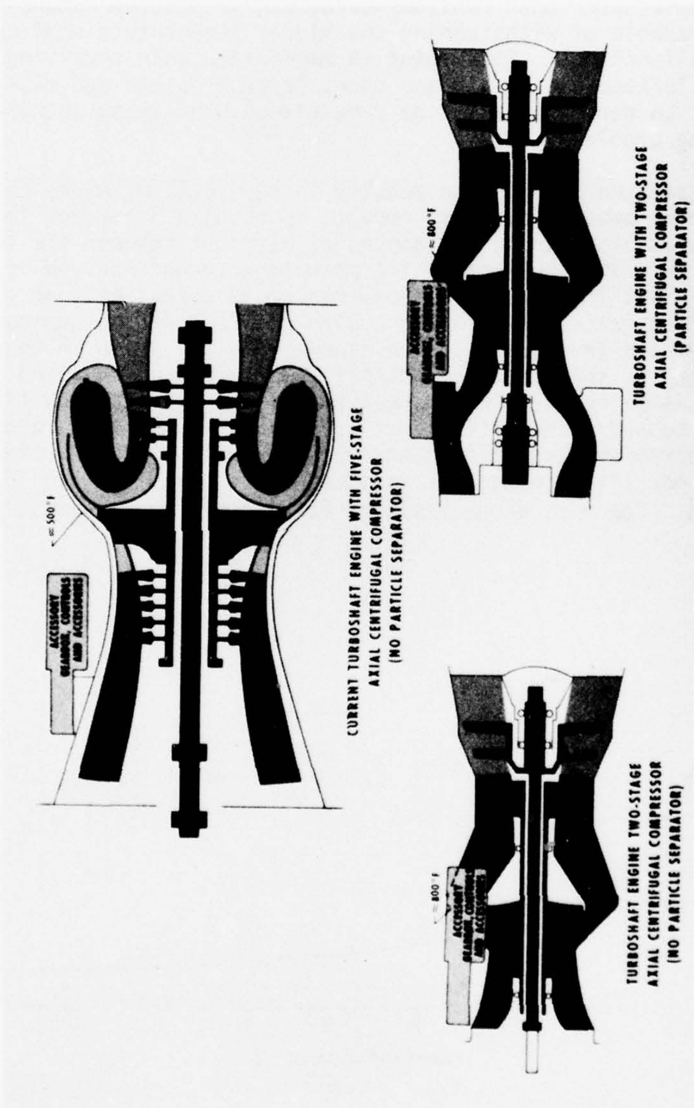

Figure 19. Controls and Accessories Size/Location (≈1500 SHP)

separators, such as the one used on the T700 engine, there is some opportunity to mount these components forward in a new "cold end."

4.2 Shaft Speed

With the trend toward higher turbine-inlet temperature and increased specific power (HP/LBS/SEC), a 2,000 SHP engine of today requires about 12 pounds of airflow per second, compared with 18 pounds per second in the mid-1960's. Therefore, the aerodynamic components are smaller, but the shaft speeds are forced upward to maximize efficiency and to minimize weight. A simplified relationship is shown in Figure 20. In particular, the higher engine output speeds (input at the transmission) pose some problems to the mechanical designer, and he may be faced with such situations as supercritical shaft speeds and bearing limitations. Therefore, it has become common to consider flexible bearing supports to minimize vibrations. One such method uses squeeze-film dampers (a typical installation is shown in Figure 21). The outer face of the rolling element bearing is fitted in the bearing support structure with an oil-filled radial clearance that can range from 0.01 to 0.0005 inch, depending upon the design. The dynamic forces acting on the bearing, and the resulting motions of the bearing outer race, produce corresponding hydrodynamic forces in the squeeze-film which oppose the dynamic motions. Thus, the squeeze-film action takes the form of a spring

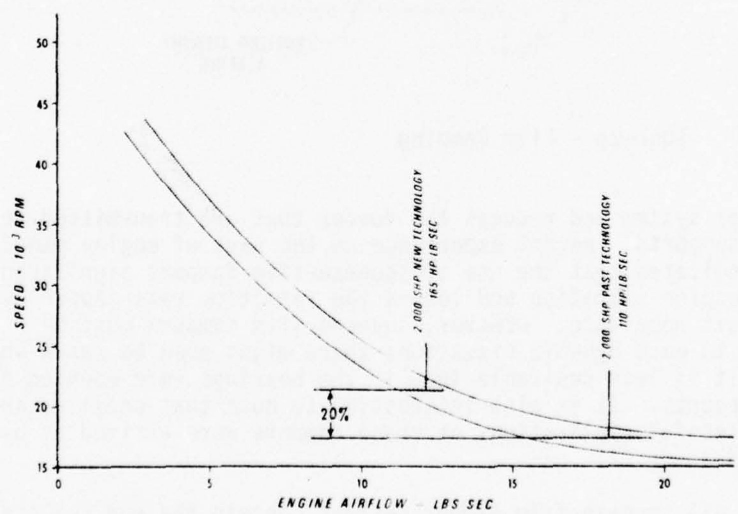

Figure 20. - Size - Speed Relationship

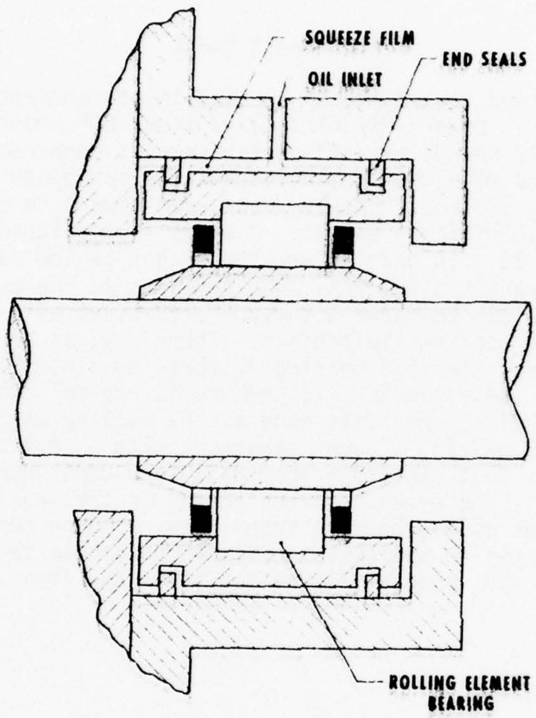

Figure 21. Squeeze - Film Damping

and damper system and reduces the forces that are transmitted to the bearing supports. Recent experience on the part of engine manufacturers indicates that the use of squeeze-film dampers significantly reduces engine vibration and lowers the rejection rate caused by rotor-shaft unbalance. However, squeeze-film dampers must be tailored to each dynamic situation; there might even be cases where the result is less desirable than if the bearings were mounted in rigid supports. It is also interesting to note that until recently, many successful applications of these dampers were arrived at by trial-and-error.

Not all squeeze-film damper designs contain the end seals shown in Figure 21, and there have been cases where special seals were added. They can take the form of "O" rings or piston seals, and, in most cases, an improvement in damping has been noted. There is no single solution to integrating the high-speed output shaft in the

airframe, but it is apparent that the engine and airframe manufacturers must share the burden and the development of new techniques.

4.3 Air-Induction Systems

Helicopter inlet designs must consider a multitude of factors; among them are temperature rise, pressure loss, anti-icing, and protection against FOD and sand and dust. In addition to the basic pressure and temperature considerations, the system must minimize the effects of flow distortion under disturbed airflow conditions caused by rotor downwash and wakes during hover, low-speed maneuvers, ground effects interactions (vortices), wind shear, and exhaust gas recirculation. It is recognized that because a helicopter travels at a relatively slow speed when compared to most fixed-wing military aircraft, engine-inlet pressure distortion and its effect on compressor performance/durability is less. Therefore, during recent development programs, inlets were designed primarily to protect against ingestion of foreign objects and sand and dust. In the T700-GE-700 engine, for example, the integral inlet-particle separator tends to minimize distortions inasmuch as it provides additional length of travel for mixing the low-velocity, distorted air mass. In addition, it incorporates swirl and deswirl vanes, which also tend to mix out the distortion ahead of the compressor. In this case, it is also recognized that the separator introduces some performance penalty over that of a conventional inlet; however, we have learned that the protection must not be added as an afterthought. Therefore, from an engine-performance standpoint, the engine is rated with the built-in protection, and the separator is standard as an integral part of the T700. The inlet also includes total anti-icing protection. Here, too, there is some penalty due to the extremely large area that must be protected, and the use of engine bleed air for heating has an adverse effect on engine performance. Even though there are some installation restraints, the system provides the needed separation efficiency and the first totally anti-iced inlet.

Despite this positive development, helicopter engines have not been totally free of hot-gas ingestion and an occasional compressor stall. Inlet heating and temperature distortions occur, as a general rule, during low-speed flight near the ground (landing, for example), or during low-speed flight with strong quarter/tail winds. This situation is depicted in Figure 22. In such cases, the rotor downwash tends to turn the exhaust toward the center of the rotor when engine exhaust velocity is low. Here, the exhaust gases become entrained in the downwash, and the resulting flow patterns are unpredictable. The causes can be understood in a general sense by recognizing that the engine manufacturer strives to minimize back pressure at the turbine-exit plane for performance reasons. Therefore, a diffusing tailpipe is used, and the result is a low exhaust

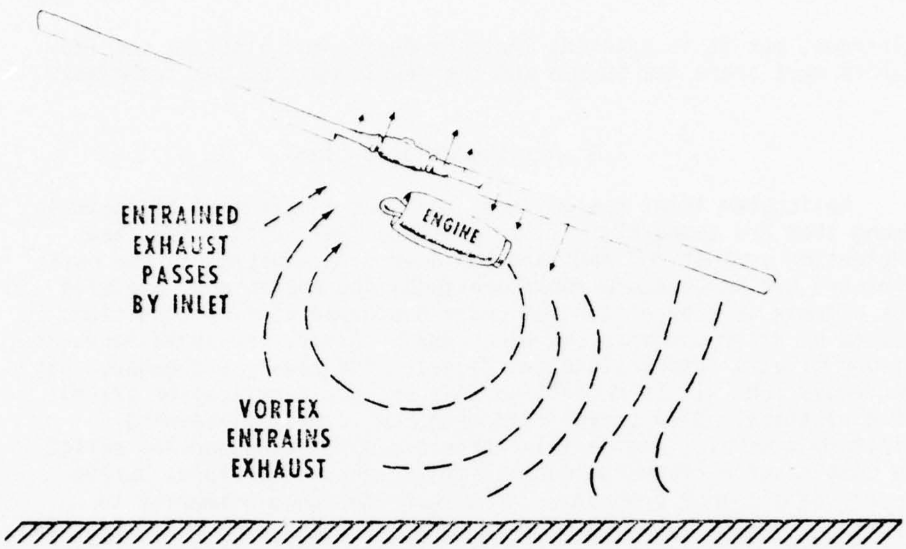

Figure 22. Exhaust Circulation

velocity at the end of the pipe. Having completed the development, qualification, and acceptance tests on that basis, the engine manufacturer has met his requirements, and the airframer then must make an installation decision. He can either use the engine specification exhaust pipe or use a smaller one to increase the exhaust velocity and accept some loss in power. If he chooses the first, he may encounter the low exhaust velocity and the reingestion problem. If he elects to take the performance penalty and use the smaller pipe, he will stand a better chance of having the higher velocity exhaust gases escape the downwash.

Due to the random and unsteady nature of the inlet-flow field, it is difficult to identify all the contributing factors to engine stalls. After extensive testing of both the engine and the aircraft system, it has been determined that in certain combinations of flight attitude and speed, a rapid temperature rise can occur over an extremely short time frame. This phenomenon was observed only after rapid response thermocouples and pressure transducers were integrated into an airframe instrumentation package to survey the inlet profile. Depending on the rate of pressure change, it is possible to produce a phase mismatch between the inlet and exit. It also forces the compressor to operate at a pressure ratio too high for its perceived corrected speed, and the flow becomes unstable. Two effects should be noted: with pure pressure distortion, the mismatch is produced by raising the pressure ratio at constant corrected speed; with pure temperature distortion, the mismatch results at approximately constant pressure ratio, but at a reduced corrected speed. Both types

of dynamic distortion have been observed, and it is clear that stalls could be initiated by a combination of both, even though the magnitude of steady-state distortion is within engine limits for surge-fee operation. A situation showing the difference in temperature effects between steady-state and dynamic conditions is given in Figure 23.

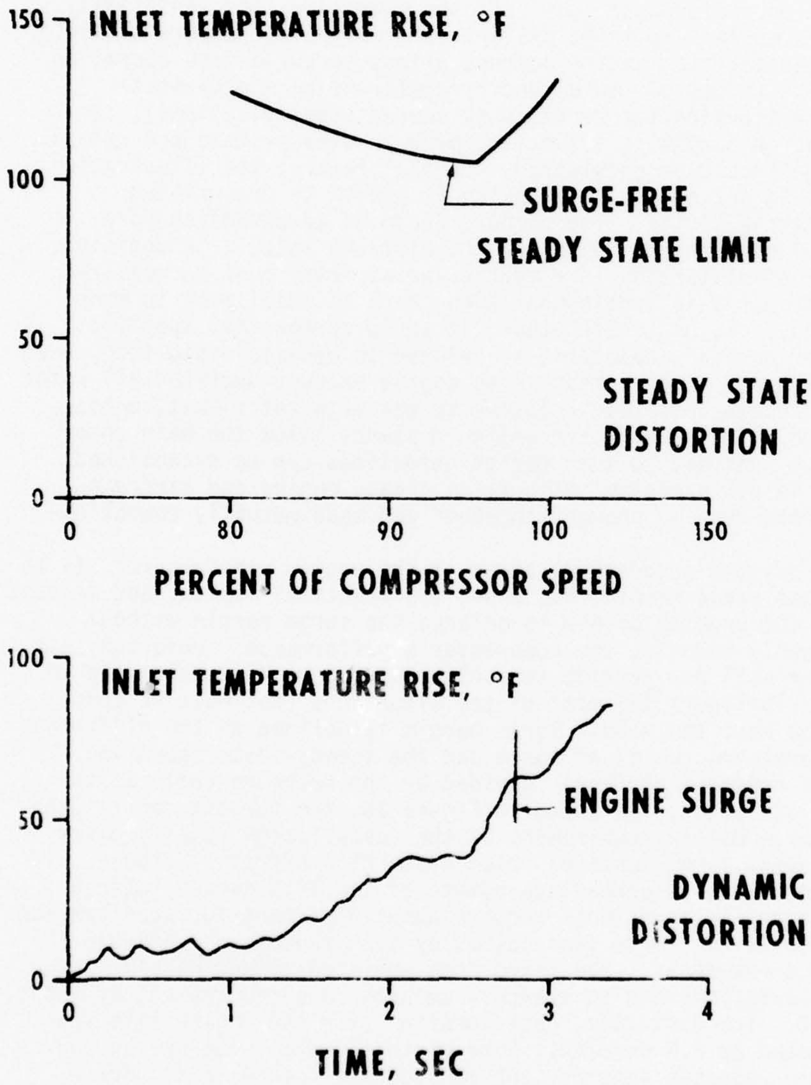

Figure 23. Inlet Temperature Distortion

Several actions are required to preclude the occurrence of such inlet problems in the future. Perhaps we should consider incorporating a minimum exhaust gas velocity in our engine specifications. Although it will not eliminate engine stalls, it will contribute to correcting two situations that now exist--exhaust ingestion and installation performance loss. At the same time, the sensitivity of turbine engines to inlet distortion needs better understanding. Rather than limiting engine demonstrations to surge-free operation at discrete values of radial and circumferential steady-state pressure distortion (as required by current specifications), the loss in surge margin as a function of both steady-state and dynamic distortion should be considered. In that regard, the illustration in Figure 24 serves as a suggestion to assist in prescribing representative limits. The purpose would be to establish more meaningful distortion limits for the airframe inlet. In addition, the level of distortion in a particular airframe must be measured accurately early in development when there is still time to make corrections. Beyond these steps, it would appear that specific helicopter design parameters, as related to dynamic distortion, need to be reviewed. Such variables as engine exhaust duct-to-tail rotor distance, engine position relative to the main rotor mast, engine height above the ground, and engine distance below the main rotor need to be analyzed so that design guidelines can be established. Here, as in all airframe integration areas, engine and airframe requirements must be brought together and made mutually compatible.

One obvious suggestion, aimed at the engine manufacturer, is to improve the surge margin; but there are practical limits, and we must consider the problem of how to enlarge the surge margin without significantly reducing the compressor's performance. Unfortunately, this paper will not provide the solution, but we have attempted to identify (in Figure 25) most of the situations that must be considered to meet the need. Surge margin is defined as the difference between pressure ratios at surge and the steady-state operating point (at constant airflow) divided by the pressure ratio at the steady-state point. As shown in Figure 25, the largest contributor to the surge margin requirement is the installation itself, based on the latest assessments of inlet distortion effects. Alone, this item accounts for 8 percentage points of the 18.5 percent needed. Next, the engine transients require about 3 percent for acceleration and shifts in the surge line caused by such factors as variable guide vane movements. The third item, related to engine and fuel control variations and tolerances, amounts to a requirement of about 4 percent. The last item, deterioration over the engine life span, is estimated as 3.5 percent. Some of these values can be reduced with rapid-response sensors, but many devices cannot communicate rapidly with hydromechanical fuel controls. On the other hand, such modern sensors as thermistors are well suited to electronic fuel controls, and that appears to be the direction for the future.

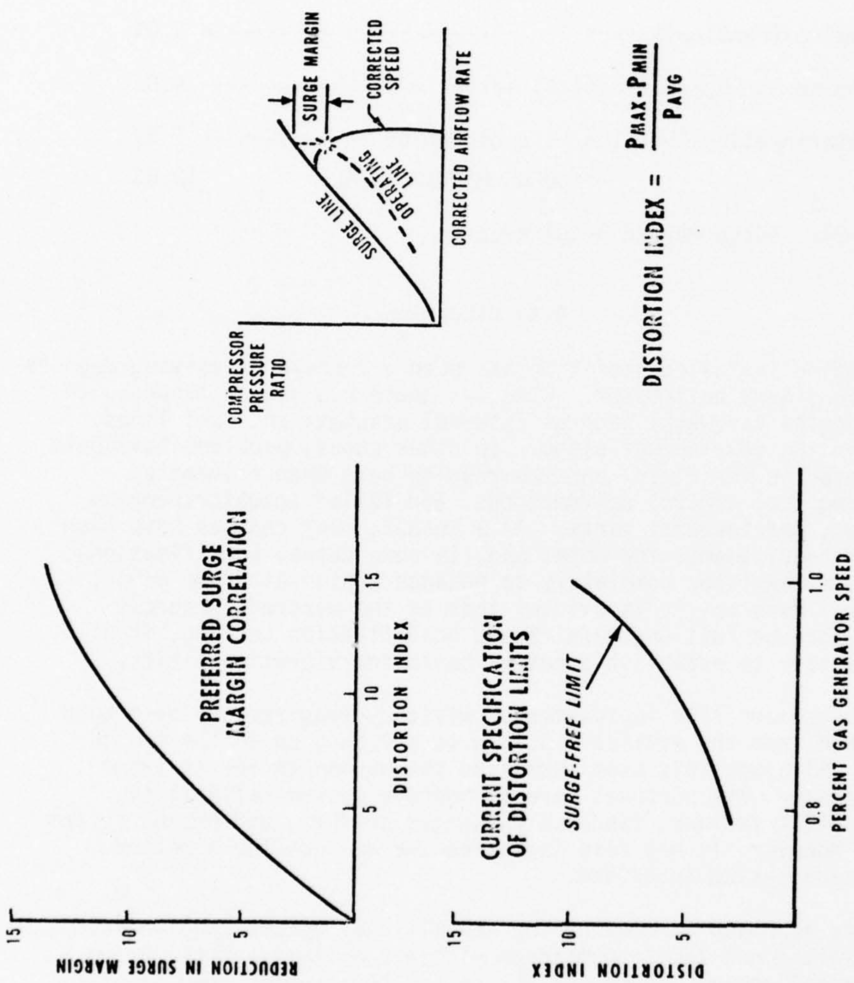

Figure 24. Surge Margin

SURGE MARGIN REQUIREMENTS

Installation Effects:	
Inlet Pressure Distortion Gas Ingestion 10° F/sec	8.0%
Engine Transients	3.0%
Engine to Engine & Control Variations	4.0%
Deterioration Over the Life of Engine	3.5%
SURGE MARGIN TOTAL	18.5%

Figure 25. Surge Margin Requirements

4.4 Vibration

Engine installed vibration has been a concern in varying degrees with every Army helicopter. Clearly, there has been a tendency to break engine hardware, such as external brackets and fuel lines, early in the development stage. In other cases, problems have been discovered in the field, and the results have been traumatic, involving fuel control malfunctions, and failed actuators, engine housings, and internal parts. As a result, many changes have been made in requirements for data, and, in some cases, specifications have been rewritten completely to enhance engine-airframe compatibility. Even so, it is evident that as the aircraft programs mature through full engine/airframe qualification testing, it will be necessary to establish a better basis for vibration limits.

To achieve life improvements, military requirements have been tightened from the earlier ± 0.15 g to ± 0.05 g up to the cruise speed. Although this step increased the burden on the airframe manufacturer, its purposes were to improve system reliability, reduce pilot fatigue, increase passenger comfort, and extend system life. However, it has also increased the demands for a better propulsion system interface.

One approach to meeting the new military system requirements is to soft-mount the transmission within the limits of the drive shaft misalignment. However, the desire for greater freedom of motion, along with high engine-shaft speeds, places greater demands on shaft couplings than before. Most of the couplings are limited to approximtely one-half degree of angular misalignment at the continuous rating. Typically, they have been stacked in series to accommodate the misalignment, but such an arrangement is undesirable,

especially when some helicopter manufacturers envision a future requirement for as much as six degrees. To meet the demand at an affordable cost and weight will be most difficult and will involve considerable research and development. As a first step toward that goal, the Army has initiated one such program, with the near-term objective of developing a high-speed, flexible coupling that will satisfy the following requirements:

° Horsepower	1500
° Speed, rpm	20,000
° Weight, lb	<1
° Angular Misalignment:	
Continuous, deg	3
Transient, deg	5

One coupling derived from this program is shown in Figure 26. Its size is approximately 5 inches in diameter. Although the work is not yet completed, tests conducted to date cite that this coupling has the potential of meeting the program objectives in terms of weight, speed, and power, but at roughly half of the misalignment requirement. The development effort is continuing; however, reaching all of the objectives will require further research.

In reviewing the experience documented during the recent UTTAS and AAH competitions, it was clear that the airframe manufacturers had different approaches to engine vibration assessment and analysis. Evaluations were difficult, partly because there was no military or industry-wide standard for measuring installed vibration of turboshaft engines. Typically, each engine manufacturer establishes the initial vibration requirement (tolerable limits) for his own engines, and in general, they are based on his past experience. Further, the method of measurement and analysis in the vehicle environment has been based on the particular airframe manufacturer's experience. The techniques generally are his own, exclusively. As such, combining the engine manufacturer's parochial view with the airframe manufacturer's approach results in such differences that correlations and comparisons between engine installations are very difficult.

Again, relating to the experience gained in the UTTAS and AAH competition, normal vibration limits were defined that would apply for 95 percent of the engine operating time. Outside of that range, extended limits were adopted for the remaining 5 percent to cover higher loads encountered during adverse maneuvers. The combination allowed engine-life issues to be primary. Recognizing that this approach is not precise, its success will be proven only after several years of experience. Here again, considerable basic research and analysis is still required to define acceptable vibration criteria.

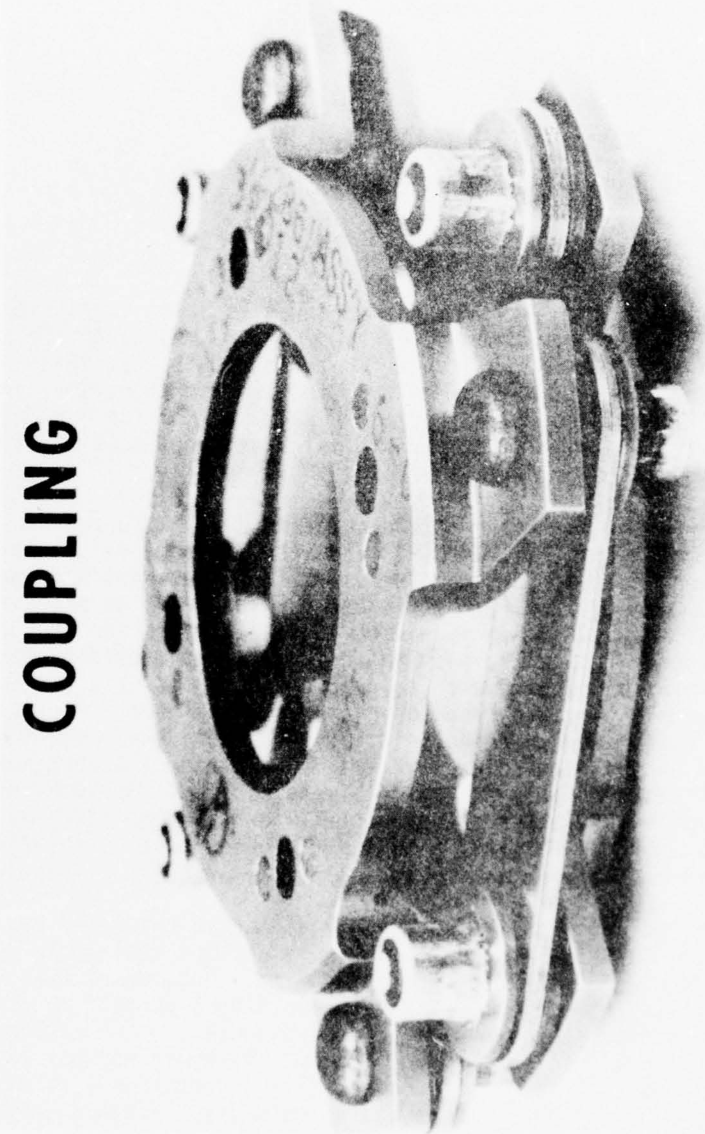

Figure 26. Coupling

4.5 Engine-Airframe Dynamics

In every helicopter engine control system incorporating automatic rotor-speed governing, the possibility of closed-loop feedback instability exists. This instability normally is referred to as torsional instability, or dynamic instability. Again, as in the vibration area, torsional instability has been a concern to some degree in all turbine-powered helicopters. Primarily, the engine fuel control system is designed to maintain a constant rotor speed, while the pilot demands more or less power by changing collective pitch. With the free-power turbine, manual control is extremely complex and dictates the requirement for an automatic rotor-speed governor. The most commonly used systems are hydromechanical and pneumatic-mechanical types; some use electronic control trimmers. Very briefly, power-turbine speed is sensed by flyweights which open against a valve-spring arrangement, changing fuel flow in proportion to a speed error. With an increasing load on the rotor (collective input), a proportional control will have some droop in steady-state speed (for stability). As the gain of the governor increases, the amount of rotor-speed droop decreases. The usual stability requirements that dictate the upper gain limits are complicated by control function and time constants, by a multitude of rotor-shaft dynamic parameters, by rotor aerodynamic damping, and by dynamics of the free-power turbine engine. Clearly, the design must be aimed at defining the fuel control characteristics which, when combined with the airframe transmission/drive system, will result in acceptable control response without introducing oscillations or instabilities.

To improve helicopter handling quality and to reduce pilot work load, new fuel controls, either of the hydromechanical or electronic type, feature an isochronous speed-governing capability. This feature is not new or unique, but it allows the pilot to select an engine/rotor speed against which the fuel control will govern, regardless of the load. In the case of a hydromechanical unit, the engine/control must meet a demanding acceleration/deceleration rapidly with wide variations in load. To reduce the response time, a separate feature, sometimes referred to as a load anticipator, is incorporated. The load anticipator receives its signals from the helicopter collective stick as it is moved by the pilot to change the rotor-blade pitch. For large excursions, the load anticipator comes into play by giving the engine a head start, rather than waiting for the power-turbine speed to increase or decrease before the isochronous speed-governing circuit reacts. These two fuel control features complement each other during normal flight maneuvers; however, in extreme cases, such as autorotation or a high-speed banked turn, they tend to conflict. An example of this conflict is depicted in Figure 27 for a helicopter in an autorotative mode.

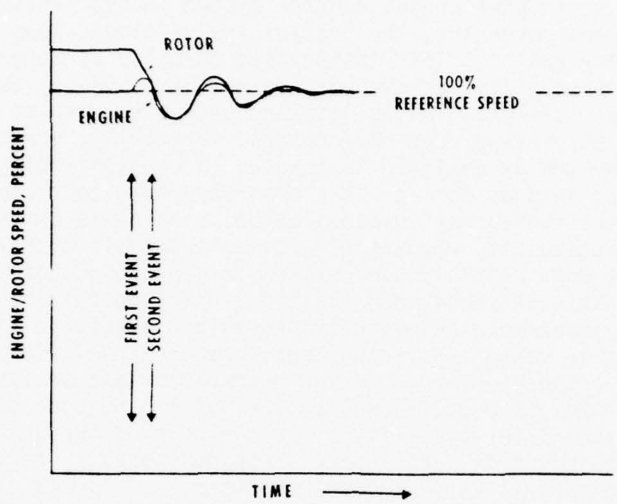

Figure 27. System Response

In this case, the rotor is unloaded and is uncoupled automatically from the power-turbine shaft through an overrunning clutch. As such, the reduced power demand of the rotor allows the power-turbine speed to fall off dramatically during the autorotative descent. During the pull-out maneuver, however, when collective pitch is applied by the pilot (first event), rotor speed starts to decay, and the load anticipator instructs the control to increase the fuel flow and the power to match the new demand. At this time in the event (Figure 27), there is no load on the engine; therefore, the power turbine accelerates and can exceed the set speed. The isochronous governing system then exercises its authority by instructing the fuel control to reduce the power with engagement of the clutch (second event) until the two speeds match and become stable. Although the situation described is undesirable, marked improvements have been made over previous control systems. However, each design demands that speed-control characteristics be based on incorporating high enough gain in the governor for acceptable response characteristics while maintaining stability. Whether the trade-off leads to the best balance of response and stability is determined by how well the engine and airframe manufacturers have communicated their integration requirements. It is expected that with the advent of an all-electronic fuel control and its inherent flexibility for reprogramming with numerous bits of intelligence, the rotor-speed droop and dynamic instabilities experienced in helicopters will be minimized.

Regardless of the type of control used, it will still be necessary to analyze and establish suitable control/engine/drive system compatibility. With the many variables and complexities, the use of computers for analysis has become mandatory to investigate changes in system parameters over discrete time periods. The results of the analysis can be plotted on the well-known Bode or frequency-response plot. The present approach is to compute frequency response of a linear system described by a set of up to 40 simultaneous equations and 40 variables. Input for the program is obtained by taking Laplase transformations of the equations of motion for the rotor system, engine Electrical Control Unit (ECU), and Hydromechanical Unit (HMU). The output of the frequency response program is interfaced with an interactive graphics package that displays phase and gain as a function of frequency. The values of the fundamental parameters used to evaluate torsional stability, i.e., phase margin and gain margin, are calculated by the program and displayed on a frequency-response plot, as shown in Figure 28. The phase margin is defined as the difference between the phase angle at zero gain and at 180 degrees. A phase margin of 35 degrees is normally considered to be adequate. The gain margin is represented by the difference between the resonant peak of the gain curve and the zero gain axis. Generally, a gain margin of at least 7.5 decibels below the gain axis is considered to be adequate.

In past programs, torsional stability was not investigated or analyzed early, and the situation was uncovered only during Army/

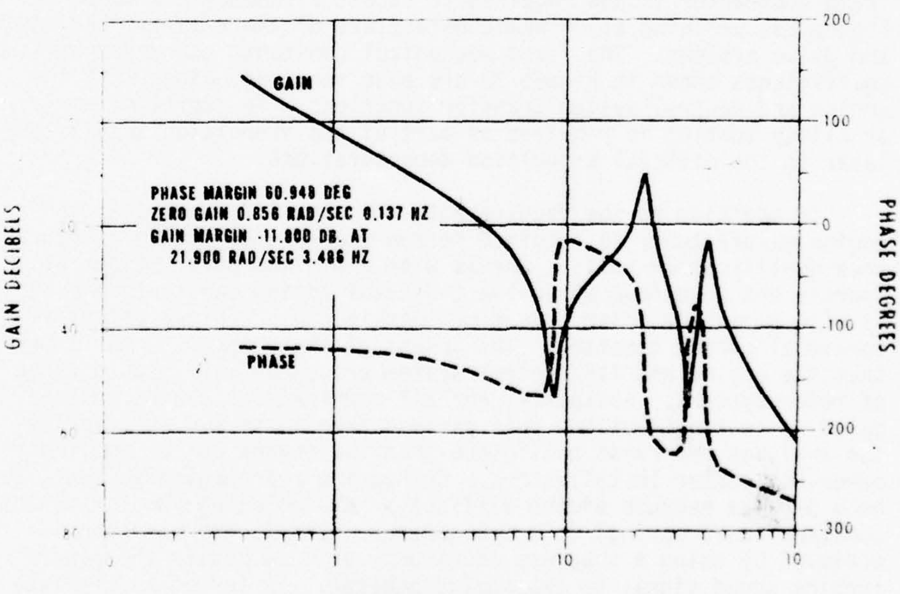

Figure 28. Dynamic Simulation

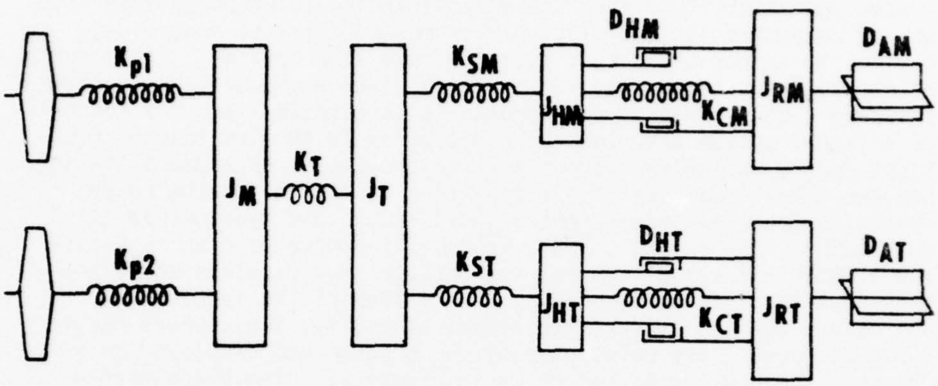

Figure 29. Helicopter Rotor System Schematic

contractor flight testing. Later analyses also led to questionable results and surprises. Therefore, the approach used in preparation of the Army's Designer's Handbook was to require torsional stability analyses with a specific procedure to model fuel control/engine and drive-system parameters, as described above. In addition, the airframe contractor is now required to submit a schematic similar to Figure 29, defining all dynamic parameters of their particular rotor and drive systems. The fixed mechanical constants and aerodynamic coefficients shown in Figure 30 are also required, along with the engine and control system transfer functions. In addition, early stability testing is required as part of the propulsion surveys and later in the official propulsion demonstrations.

In addition to the requirements for stability analyses, the engine manufacturer is required to run two special 150-hour endurance qualification tests. One is with a minimum polar moment of inertia and a maximum effective torsional spring constant; the other is with a maximum polar moment of inertia and a minimum effective torsional spring constant. The intent of these tests is to insure that the engine and its control system are compatible with a range of rotor systems. Obviously, not all combinations can be investigated by testing, but the data derived from these two runs verify the analyses and raise confidence that the engine can be applied to other helicopter installations. Conducting representative tests can be a problem because of the difficulty in simulating the low spring constant (soft spring) on a dynamometer. Similarity has been achieved by using a separate electronic unit to modify the power-turbine speed signal to the engine control. It served to simulate the transient response so that the engine control reacted as though the output shaft were connected to a helicopter rotor system. Using

MECHANICAL CONSTANTS

# 1 Engine Coupling Shaft Spring Constant	Kp1	ft#/rad @ rpm
# 2 Engine Coupling Shaft Spring Constant	Kp2	ft#/rad @ rpm
Main Gearbox Input Shaft Spring Constant	K_t	ft#/rad @ rpm
Main (Forward) Rotor Shaft Spring Constant	K_{SM}	ft#/rad @ rpm
Tail (Aft) Rotor Shaft Spring Constant	K_{ST}	ft#/rad @ rpm
Main (Forward) Rotor Lag Hinge Damping Coefficient	D_{HM}	ft#/rad @ rpm
Tail (Aft) Rotor Lag Hinge Damping Coefficient	D_{HT}	ft#/rad @ rpm
Combining Gearbox Inertia	J_M	slug ft^2 @ rpm
Main Gearbox Inertia	J_T	slug ft^2 @ rpm
Main (Forward) Hub Inertia	J_{HM}	slug ft^2 @ rpm
Tail (Aft) Hub Inertia	J_{HT}	slug ft^2 @ rpm
Main (Forward) Rotor Inertia	J_{RM}	slug ft^2 @ rpm
Tail (Aft) Rotor Inertia	J_{TR}	slug ft^2 @ rpm

AERODYNAMIC COEFFICIENTS

HP	K_{CM} ft#/rad	D_{AM} ft#/sec	@N rpm	K_{CT} ft#/rad	D_{AT} ft#/sec	@N rpm

HP	Total Engine Horsepower
K_{CM}	Main Rotor Centrifugal Spring Constant
D_{AM}	Main Rotor Aerodynamic Damping
K_{CT}	Tail Rotor Centrifugal Spring Constant
D_{AT}	Tail Rotor Aerodynamic Damping

Figure 30. Mechanical Constants and Aerodynamic Coefficients

this approach, T700 tests were run with torsional spring constants of 20 lb-ft/rad and 1689 lb-ft/rad, with a polar moment of inertia of 1.942 slug-ft^2 and 0.3963 slug-ft^2, respectively.

A considerable amount of effort has been expended in this area, and although some improvements have been made, the problems have not been erased. Increased emphasis is required, and the need for improved methods or extensions of existing analytical techniques is evident. More appropriately, the most effective computer techniques that consider the significant nonlinearities in a drive train and governor should be considered.

4.6 Electronic Controls

It is expected that with the introduction of full-authority electronic controls, some of the problems that we have experienced can be minimized. Figure 31 shows some of the advantages and features that could be available. It is hoped that the potential will be realized through two recently awarded contracts to develop an 800 SHP Advanced Technology Demonstrator Engine (ATDE). Each contractor (Detroit Diesel Allison and AVCO Lycoming) will incorporate full-authority electronic controls. We believe that this effort will stimulate the industry and will promote development of electronic units for other helicopter engines. If successful, it is envisioned that existing engines will be retrofitted to overcome some of the field problems that have caused an inordinate amount of maintenance. The ability to detect a malfunction, in itself, will be of great benefit to the user. In that regard, there have been many cases where a large share of the fuel controls removed in the field were found to have no fault when checked on a control rig, and reasons for the removals could not be confirmed. Similar experience exists in the commercial airline industry, leading to the belief that the difficulty lies with trouble-shooting equipment and procedures.

The opportunities afforded by recent advances in electronic technology indicate that significant improvements in other areas are possible. In that regard, there is some promise that the control components can be applied universally, that is, incorporated into units that will be used in other types of engines and applications. In addition, there is the potential for reduced cost in performing such functions as automatic starting, overspeed protection, torque limiting, load matching or sharing, temperature limiting, isochronous governing, hot-start protection, and engine history recording. Still to be addressed and quantified is whether these units will live up to expectations in a helicopter environment.

- ° Lower acquisition cost
- ° Automatic engine start
- ° Overspeed protection
- ° Torque limiting
- ° Temperature limiting
- ° Isochronous speed governing
- ° Failure detection of control
- ° Torque matching (multiple engine application)
- ° Hot start protection
- ° Engine condition history recorder
- ° Potential for universal engine application
- ° Greater flexibility for matching of engine to rotor characteristics (improved rotor stability)
- ° Less vulnerable area
- ° Lower weight

Figure 31. Electronic Fuel Control

5.0 CONCLUDING REMARKS

Throughout this discussion, we have attempted to summarize only a few of the engine-airframe integration topics that have particular importance to the user. For many years the emphasis has been on fixed-wing aircraft, and the helicopter, by comparison, has been relegated to a lesser role. Yet, this special-purpose machine requires attention to highly complex technology disciplines that are critical to its continued growth and success. Many tasks to eliminate voids in the technology remain, and each of them would warrant a separate paper to cast it in its proper light. However, because this discussion has been presented in the form of a survey paper, we felt it necessary to comment further on the needed research and development, along with the topics discussed in this paper. The following points are offered in the hope that they might provide an added stimulus to the technical community:

- ° Accessories mounted on the engine must be qualified at realistic temperature and vibration limits. Our concern is based on situations where components have been qualified to operate at temperatures that were not representative of the engine compartment. Worse yet, the limits could have been set higher without a dramatic increase in cost. Instead, the airframe manufacturer was forced to make modifications for increased

cooling, sometimes by trial-and-error. The process has been expensive, often requiring repeated flight tests. On occasion, he has found it necessary to request a waiver, realizing that he could not meet the requirement in the engine manufacturer's specification.

° Fuel controls must be qualified at conditions that represent the airframe environment. It must be demonstrated that they will provide all normal functions while subjected to vibration amplitudes and frequencies at temperatures within the operating range of the engine installation. In addition, it must be shown that they will not malfunction at any of the response frequencies in that range.

° Containment and structural integrity of the engine must be proved at its highest power and speed. In particular, containment of failed blades must be demonstrated, and there must be no condition which would allow separation of the discs.

° High-speed clutches for use in helicopter drive trains must be refined to permit rapid and repeated engagements/disengagements without malfunction. In this case, torsional oscillations must be minimized to preclude the situation where a momentary unloading of the clutch might occur.

Again, it is not possible to identify all items that need attention. However, one item is worthy of further emphasis---the future of full-authority, electronic fuel controls. It is known that several manufacturers are deeply involved, but the challenges of meeting demands of the helicopter environment suggest that more attention is needed. Toward that end, it is hoped that significant contributions can be made by developing units that will survive the vibration and temperature. When that point is reached, and when the electronic fuel control has met the requirements shown earlier in this paper, we are confident that it will be adopted with enthusiasm.

The authors have not attempted to provide a detailed technical paper on any one subject. Instead, it was considered important to draw attention to the needs of engine-airframe integration. At the same time, we wish to acknowledge that Army progress in this area over recent years has been most encouraging.

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DISCUSSION

GOETHERT: (The University of Tennessee Space Institute)

On the curves for the surge of the compressor, you showed the reductions due to steady state temperature changes and due to the dynamic changes. There must be some requirement which you put on these curves; for instance, you might want to tolerate a degradation by a certain amount. Otherwise you can put quite a number of curves there. What do you use as a requirement to get those particular curves?

ACURIO:

If you are asking whether we were willing to accept moving to the right over the knee of the speed line to stay away from surge, the answer to that is: not as far as we have to go because we fall off in efficiency very rapidly. The margins we show are for a two-shaft engine, and the power-turbine speed has nothing to do with surge margin in this case. Primarily it is the gas producer that we are concerned about.

HILL: (Grumman Aerospace Corporation)

With respect to the problems associated with heating of the subcomponents, are you aware of the type of work that has been done in a space vehicle?

ACURIO:

Yes. In fact, a good deal of work has been sponsored by one of our directorates at Fort Eustis. We are aware of that work. You may contact the authors for reference or to obtain a published listing of contract reports.

JONGENEEL: (Douglas Aircraft Company)

You mentioned the desire to get the electronic controls off the engines. There are successful precedents for this, such as the DC10 auxiliary power unit.

ACURIO:

Yes, we are aware of that. In fact, we have debated that question several times. We are questioning where we want to put the electronic fuel control, as I said earlier. We need to determine if it should be deliverable as part of the engine package or as a separately identifiable unit. That has been the debate. It has been our desire to place it in the engine compartment. We might find that the answer will be just as you suggested.

SECTION II

SYSTEM STUDIES

ENGINE CYCLE CONSIDERATIONS AND ASSOCIATED INSTALLATION PROBLEMS FOR SHORT HAUL AIRCRAFT

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Bristol, England Bristol, England

ABSTRACT

A changing world economy makes it appear likely that in the coming decade there will be diverging requirements for propulsion systems for civil and military short-haul aircraft. For civil aircraft, the emphasis today is on cost and fuel efficiency, the desire for V/STOL being tempered by economic considerations. For new military aircraft, STOL and V/STOL characteristics have an operational significance which overrides economic criteria, although cost to achieve a given objective must remain of prime importance. Nevertheless, both types of aircraft may well require low specific thrust engines with more complex propulsion systems necessary on military aircraft.

New research requirements are likely to stem particularly from the use of large diameter low pressure ratio fans and involve the achievement of low pod drag and weight, low airframe interference, proper operation of fans on their characteristic and efficient thrust reversal. Military V/STOL may prompt additional work on problems of high intake incidence, thrust modulation, and control of power-sharing multiple engine systems.

1. INTRODUCTION

This paper is concerned with the broad subject of engine and installation aspects of future civil and military short-haul transport aircraft. The requirements and likely configurations of such aircraft may well be diverse and are therefore treated separately. In both fields, some preferred aircraft configurations

are now being more clearly identified and, with them, the specific problem areas requiring research.

While the need in the civil market is moving towards energy-conserving low-cost aircraft, in the military field the main requirement is for a specialist V/STOL aircraft to operate from a new generation of small aircraft carriers and ships with platform facilities.

Success in the short-haul civil market will be governed more by financial considerations and, in particular, the standards set by the fuel consumption and first cost of propeller driven aircraft. In the military V/STOL field, the primary obstacle at present is technical feasibility, although it should be remembered that one basic reason for the move to small ships is that of cost. It is therefore to be expected that the research requirements in these two parts of the field will be different, at least with respect to their installations.

2. SHORT-HAUL CIVIL TRANSPORTS

A good definition of a short haul aircraft would be one optimised for 200 nautical miles stage length with the capability of 3 or 4 unrefuelled stops. This implies that the aircraft might have a maximum unrefuelled range of 700 to 1000 nautical miles.

It is well known that as operating stage length is reduced, the D.O.C. optimum speed for civil subsonic aircraft is reduced. (Fig. 1). At, say, 200nt.miles the slope is steep, and speeds as low as 0.6 Mach number can be justified on a D.O.C. basis. Such aircraft spend most of their time climbing and descending, and cruise is a relatively unimportant condition except when the aircraft is operating away from its normal pattern. It is shown in Reference 1 that the propeller driven aircraft, with its propulsive efficiency near constant over a wide speed range, has an uninstalled performance advantage at the lower climb and descent speeds (Fig. 2).

Studies by aircraft manufacturers indicate that, even at $M = 0.6$, the advantage of the propeller is reduced and may disappear when full account is taken of all the installation advantages of the turbo-fan.

Much of the short-haul civil aircraft market is disadvantaged because of the smaller aircraft required. Engine performance is then reduced because cycle pressure ratios cannot be held at the high values possible on larger engines due to the physical size of blades. Reynolds number effects are increased, and manufacturing tolerances worsen turbo-machinery

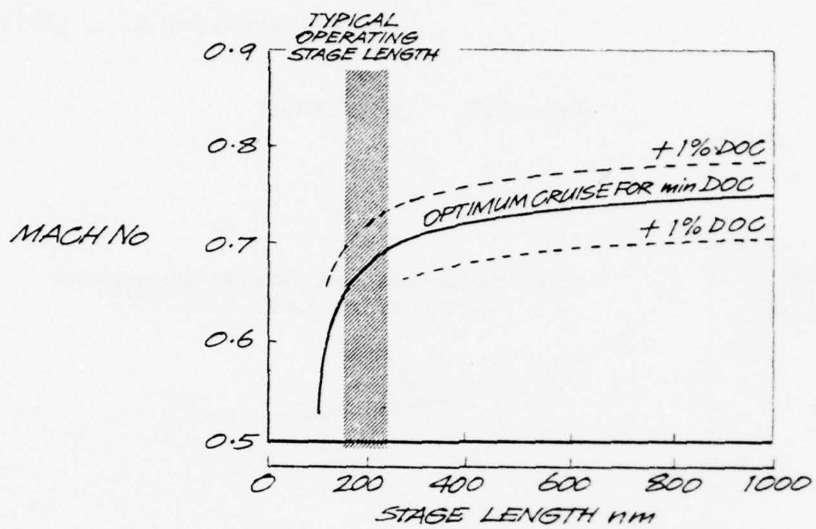

Figure 1. Cruise Speed for Minimum DOC

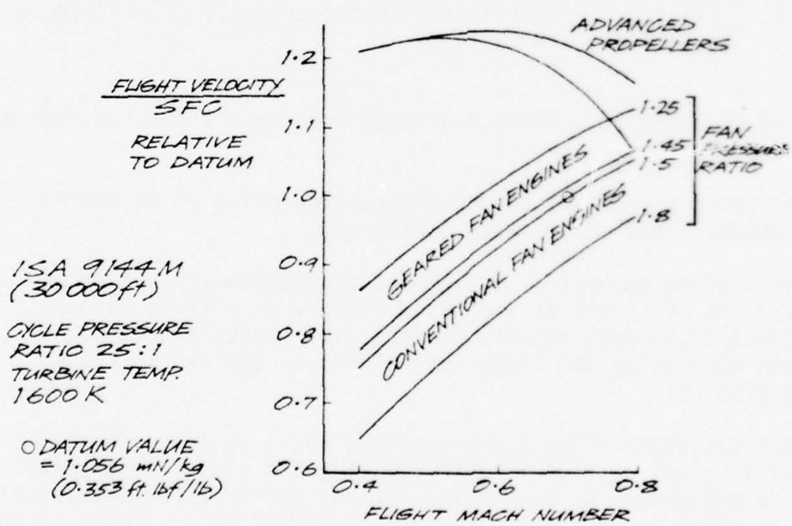

Figure 2. Subsonic Engines-Uninstalled Performance Variation of $\frac{\text{Flight Velocity}}{\text{SFC}}$

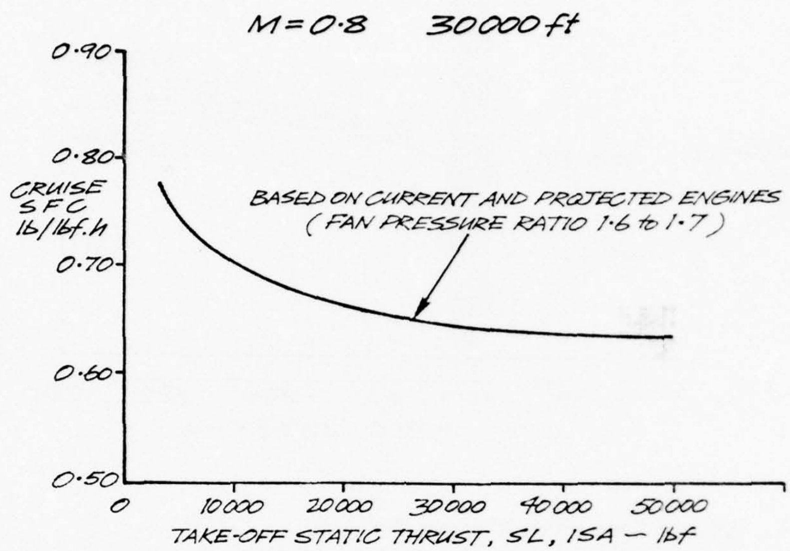

Figure 3. Effect of Size on Engine Characteristics (Civil Single Stage Front Fan Engines)

geometric standards. All these effects lead to a situation such as shown in Fig. 3.

The factory cost and subsequent selling price of an engine is not proportional to its size or thrust.

For a given mechanical design and thermodynamic standard, cost per unit of thrust is increased rapidly as size is reduced. This effect is, to some extent, offset by the market choice of increased complexity and higher efficiency as engines increase in size (Fig. 4).

When all these effects are combined and a twin-engined aircraft is simply scaled in absolute size, the effect on aircraft D.O.C. is shown in Fig. 5. These aircraft suffer a combination of penalties which increases D.O.C. by a factor of approximately 2 in reducing seating from 120 down to 40 passengers. The effect on a new design of using a very low specific thrust engine with, say, a 20% improvement in installed SFC is to reduce D.O.C. by 8½% for a constant passenger load. (11% at constant engine size and increasing passenger load). This will be a large proportion of a short haul operation profit margin, particularly using smaller aircraft.

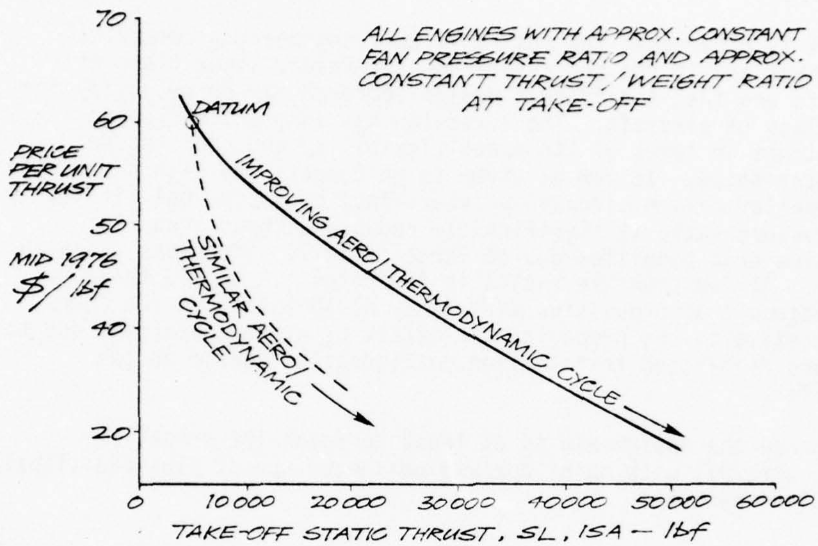

Figure 4. Effect of Engine Size on Costs (Civil Single Stage Front Fan Engine)

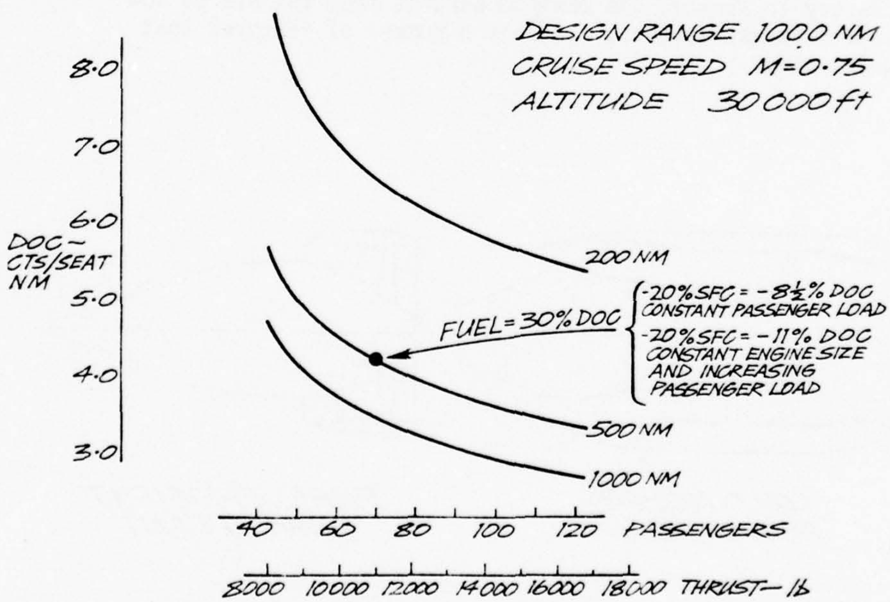

Figure 5. Direct Operating Cost V Engine Size Twin-Engined Aircraft

It is clear that the most difficult engineering/commerical objectives are in the short-haul smaller Feeder-liner class of aircraft and that a different design approach may be necessary for this class of aircraft. The turbo-fan has inherent marketing attractions in terms of its speed flexibility and cabin noise characteristics. It can be shown to be competitive with new designs of propeller driven aircraft on short-haul operation only if its fan pressure ratio is significantly reduced without incurring excessive drag penalties due to larger nacelle dimensions (Figs. 6 and 7). At fan pressure ratios in the range 1.2 to 1.3, there are indications that propulsive efficiency disadvantage of the turbo-fan relative to the propeller are offset by higher penalties due to the more restricted installation configuration forced on the propeller.

Hence the design aim is at least to match the propeller driven aircraft with much improved passenger appeal plus additional operations benefits.

3. RESEARCH OBJECTIVES - CIVIL SHORT-HAUL

The type of powerplant shown on Fig. 6, which will be necessary to achieve the maximum benefit from the use of low specific thrust engines, presents a number of features that require study.

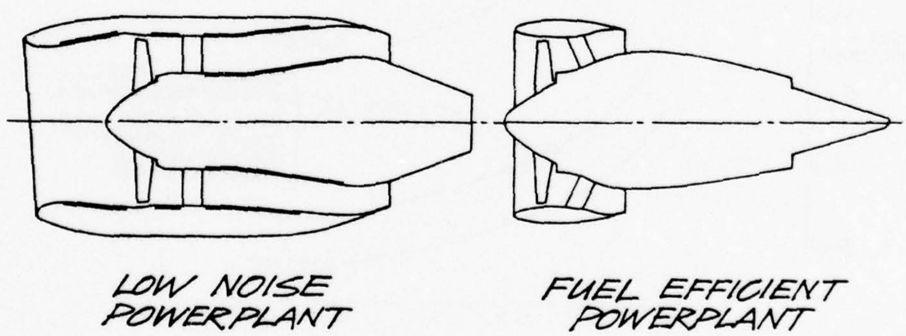

Figure 6. Lower Noise or Lower Fuel?

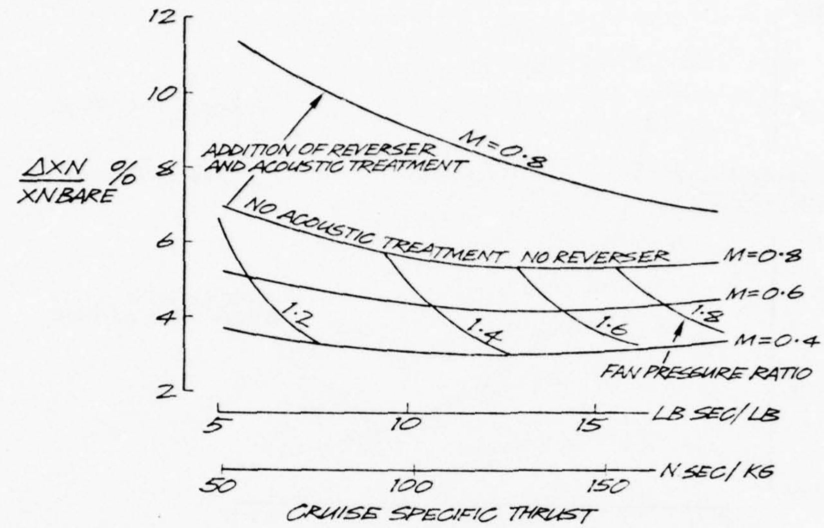

Figure 7. Total Net Thrust Loss Due to Powerplant

First, however, there are some aspects of the installation of these engines which are independent of the pod geometry. The need or otherwise for a variable area fan nozzle and the effect of fan entry flow Mach number are two. Engine/airframe interference effects may also be a major consideration, particularly for rear fuselage mounted engines.

4. FAN AND FAN NOZZLE

The effect of forward speed on a low pressure ratio fan is to increase its non-dimensional nozzle flow and duct Mach number if its propelling nozzle is fixed in area. To the fan designer, this is manifested as a change in the fan operating line which moves up towards surge at take-off and down towards choke at high forward speed. The designer may not, as a result, have freedom to choose his cruise design point for maximum efficiency.

A variable fan nozzle would remove this constraint and allow optimum SFC in cruise and maximize thrust at take-off. On the other hand it would be advantageous to avoid the geometric, mechanical, and weight penalties of such a nozzle if this is at all possible. If it were possible to design a fan to operate efficiently over a wider range of entry flow conditions, then it

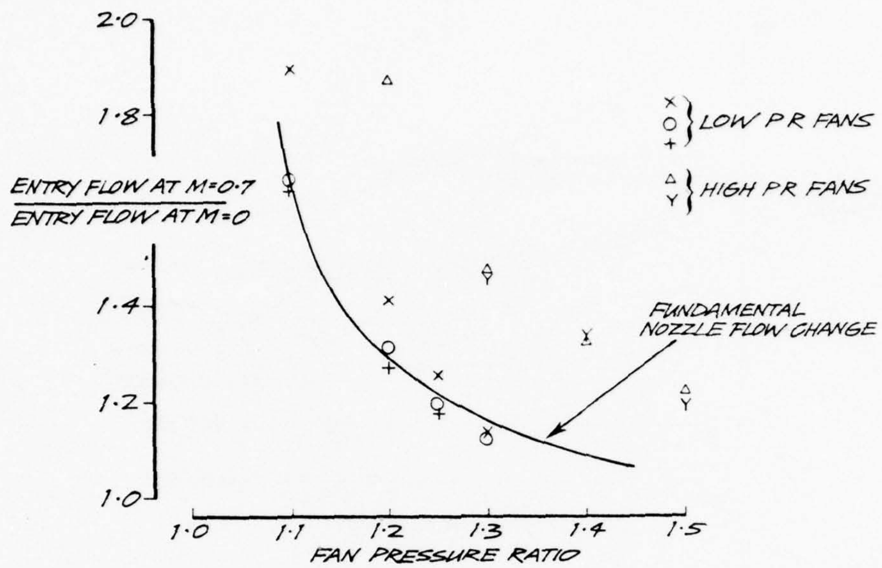

Figure 8. Low Pressure Ratio Fans Variation of Flow Capacity with a Fixed Nozzle

might be possible to dispense with nozzle variability. Fig. 8 shows the ratio of fan non-dimensional entry flow change which takes place between a flight Mach number of 0.7 and static conditions. The points shown represent the ratio of surge to choke flows for a number of different fans. From this it is evident that the fundamental adaptability of the fan to cater for the variation in its flow caused by forward speed roughly matches at all fan pressure ratios. However, on fans specifically designed for low pressure ratio, there is not an adequate flow range between surge and choke. High pressure ratio fans run back to lower pressure ratios show much better characteristics and imply that lower axial Mach numbers may be required, leading to larger fan diameters. Some simple form of nozzle variability which, in effect, blows-off at low speeds might be sufficient to cater for this problem.

The variable pitch fan (Fig. 9) is another possible method of achieving matching, but it must be emphasised that it is then necessary to design for a high axial Mach number through the fan at cruise and a low Mach number at static conditions and adjust the blade angles accordingly (Fig. 10).

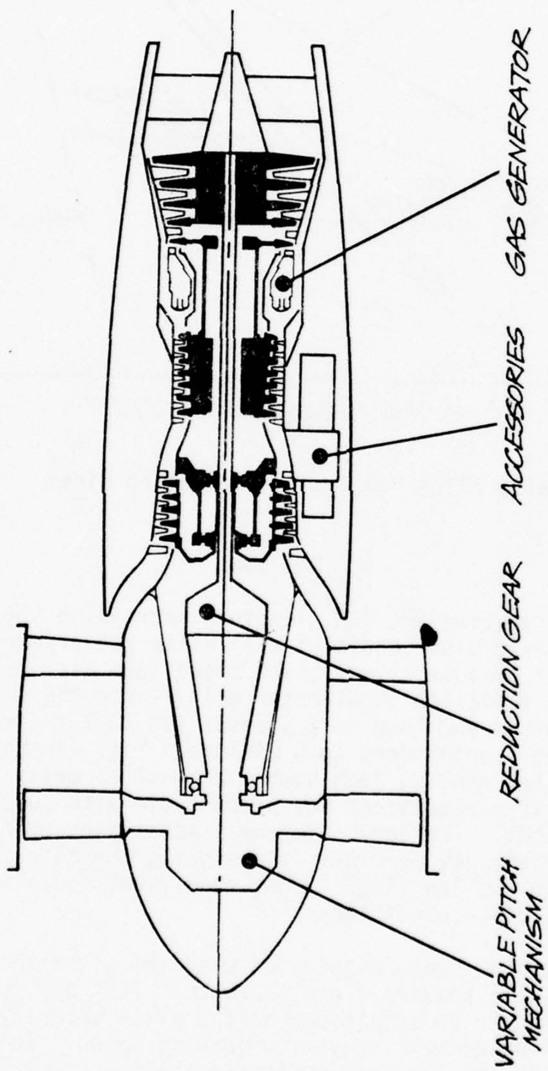

Figure 9. The High Bypass Variable Pitch Geared Fan Engine

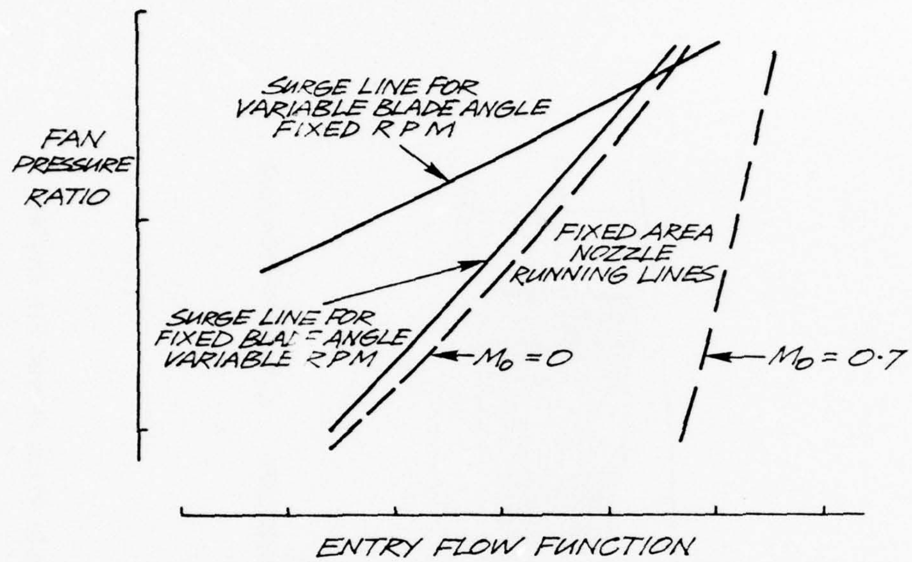

Figure 10. Variable Pitch Fan Typical Operating Lines

5. FAN COWL

As well as reducing the fan diameter, increasing the fan entry Mach number at the cruise condition will allow reductions in the fan cowl diameter because at any given speed less air needs to be spilled. The reduction in diameter will reduce the amount of boattailing required and lead to a shorter fan cowl thereby saving weight and keeping drag to a minimum. Fig. 11 shows the effect of fan entry annulus Mach number on cowl diameter and indicates that large reductions may be possible with consequent reduction in weight. The very slim cowls associated with this concept will present new problems in designing the nose shape for both the high and low flight speed conditions and also in catering for incidence conditions.

The zero-length intake associated with the ultra-short cowl shown in Fig. 6 will require a new approach. Test bed performance checks will no longer be sufficient, as the axial velocity gradient across the fan annulus will vary with forward speed. This could well cause a change in the fan performance between test bed and in-flight; therefore it will be necessary to revise design methods to take account of entry velocity gradients.

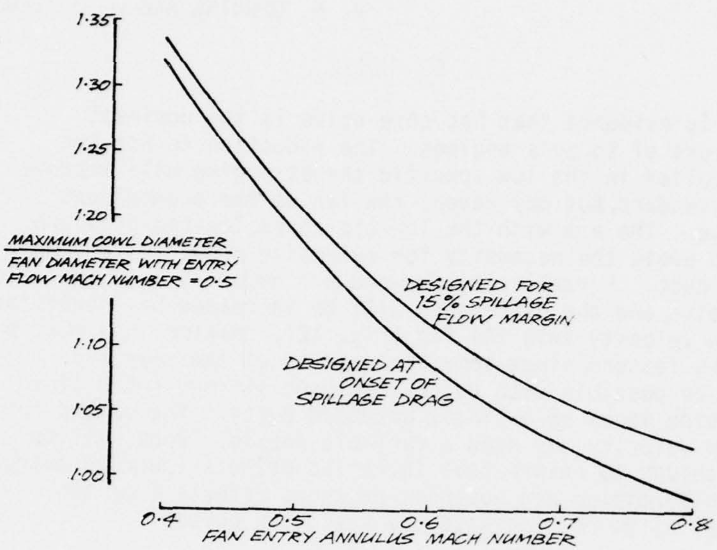

Figure 11. Fan Cowl Diameter as a Function of Fan Entry Flow Mach Number

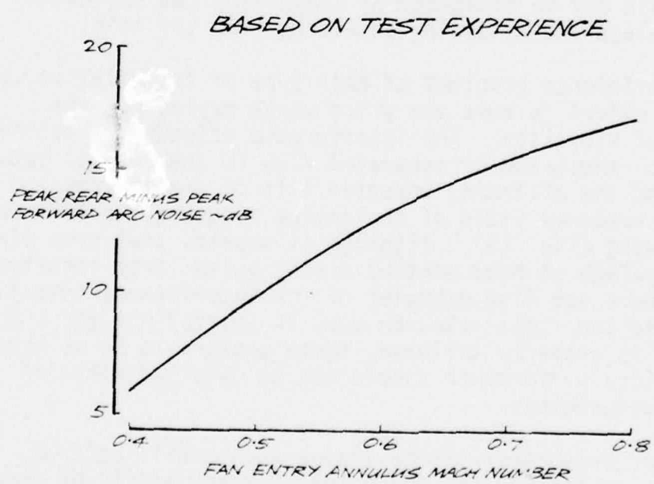

Figure 12. Fan Entry Mach Number Influence on Rear-Forward Arc Fan Broad Band Noise

6. NOISE

There is evidence that hot core noise is the dominant source on most of today's engines. The reduction in hot jet velocity implied in the low specific thrust engine will improve the noise standard, but may reveal the fan as the predominant noise source. The aim with the low-tip-speed, low-fan-pressure ratio is to avoid the necessity for extensive acoustic treatment in the fan duct. Normally, the forward arc noise is less than the rear arc noise, and the difference will be increased by increasing the airflow velocity into the fan (Fig. 12). Maximum use must be made of this feature, since some attenuation of the rear arc noise will be possible with noise absorbent linings (with lined fan exit guide vanes or a lined, extended duct). The demand for a high flow velocity may need a variable nozzle. Work will be necessary though to ensure that installed effects (such as entry flow non-uniformities and upstream pressure effects from the mounting pylon) do not increase the basic fan noise.

7. INSTALLATION

Cross flows in flight, due to the wind upwash for example, may lead to vibration caused by the changing incidence on the fan blade as it rotates - a problem well known to the propeller designer. This may be mitigated by installing the engine on the rear fuselage and obtaining shielding from the wing.

The interference problems of this type of installation would need special effort to make the price worth paying for the elimination of vibration. The interference effects are manifested in three ways: shocks and/or separated flow in the channel between the engine and the airframe, unwanted lift on the engines, and the upstream pressure field of the engine reducing the lift on the inboard wing (Fig. 13). Although it appears that some aircraft with rear fuselage mounted engines suffer quite large interference penalties, there are also examples of low interference installations. Hence provided the right research work is carried out and the installation is properly tailored, there appears to be no reason why satisfactory performance should not be obtained with low specific thrust engines.

The major proportion of the engine thrust will be from the fan, thus continuing the trend that already exists of reversing only the fan stream. It is suggested that a clear indication of the reverser thrust requirements is needed to help in the choice between variable pitch fan blades and a more conventional reversing system. It may well be that short-haul aircraft neither need nor can afford the luxury of thrust reversal. Some of the advantages of the two methods are listed in Fig. 14.

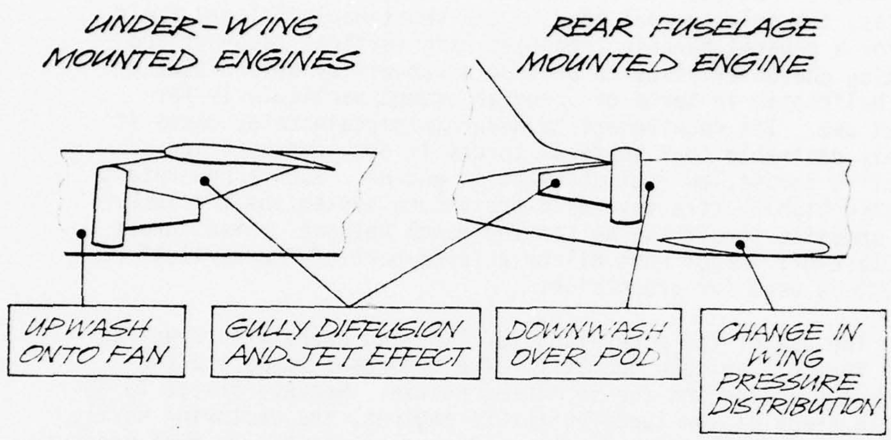

Figure 13. Sources of Interference

IS IT REQUIRED?

HOW MUCH?

<u>BY VP FAN</u>

Fast response

No self or cross ingestion

VP may be required for other reasons

<u>BY "CONVENTIONAL" THRUST REVERSER</u>

Development can be independent of the engine

Vast Experience of systems and aerodynamics

Parasitic losses may be more critical at low pressure ratio

Possible to control reversed efflux pattern

Figure 14. Thrust Reversal

8. SHORT-HAUL MILITARY V/STOL

Apart from the A.M.S.T., which is required for greater design ranges, the only current need in the short-haul military field is for a general purpose transport with vertical take-off and landing characteristics to provide a capability beyond that of the helicopter in terms of speed and range, particularly for Naval use. Its requirement to hover in certain roles makes it highly desirable that vertical thrust is provided by a low specific thrust, low fuel consumption engine. Such a powerplant is also highly attractive as a propulsion system, and the lower the specific thrust the better the match between cruise thrust and take-off thrust when all or a large part of the vertical thrust is used for propulsion.

The use of the powerplant for lift and propulsion requires some form of variable geometry. Two important forms are the vectoring engine and the vectoring nozzle. Because of the large nozzle areas of low specific thrust engines, the vectoring nozzle becomes somewhat cumbersome and will be susceptible to duct pressure loss. On the other hand, the vectoring nozzle system avoids the varying intake incidence of the vectoring engine together with the problems of engine rotation and oil system operation.

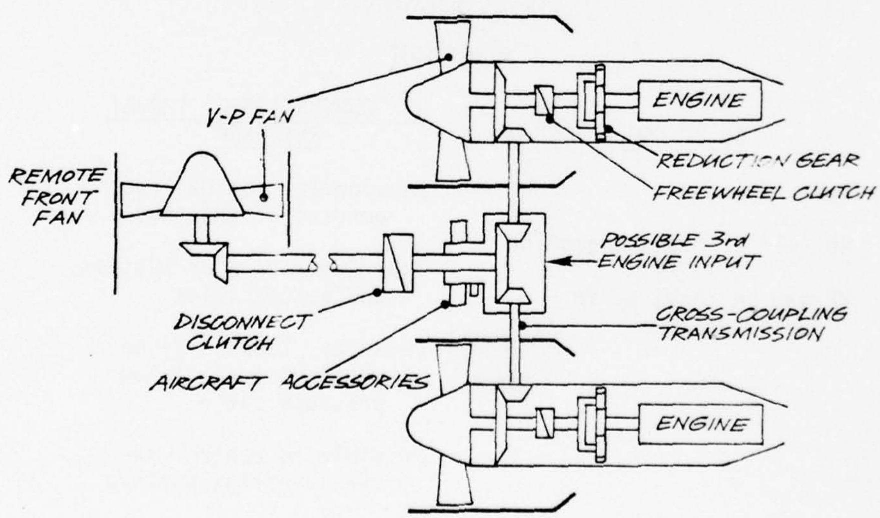

Figure 15. V/STOL Propulsion System Coupling and Power Transmission Layout

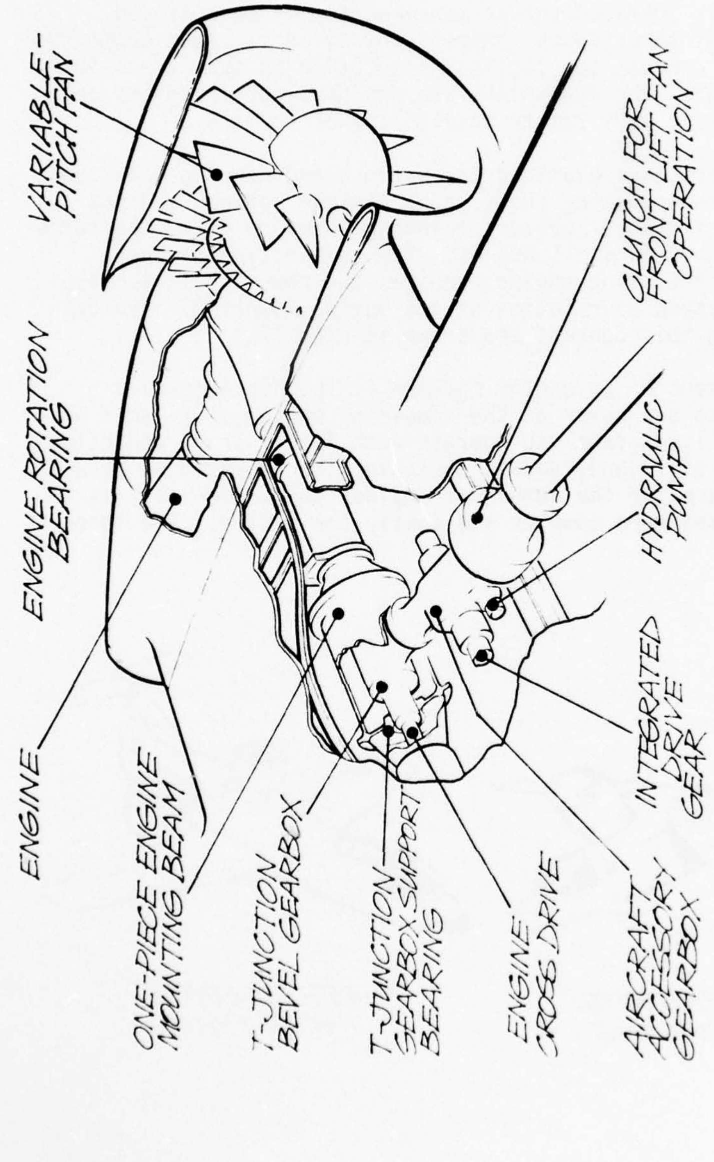

Figure 16. Engine - Installation

The main design problem of this type of aircraft, as with all multi-engined V/STOL aircraft, is that of total thrust management, particularly in the engine-failed case. Unless the principle of shutting down an opposite balancing engine in a failure case is embraced, thrust management must be achieved by some form of thrust/power sharing involving variable geometry. In the latter concept, the implicit assumption is that the main thrust producers, the fan units, are designed for integrity and hence have only a very remote possibility of failure.

Many papers have examined the various engineering solutions, the main contenders being shaft power linkage between engines and gas power linkage. Current thinking seems to favor the former as illustrated in Figs. 15 and 16. The combination of shaft linking and a vectoring engine requires the power shaft to pass through the centre of rotation of the engine mounting. Typical aircraft using this concept are shown in Fig. 17.

In the event of an engine failure, it is then necessary to redistribute the power of the remaining engine(s) in such a manner that all the fans can operate with little or no reduction in conversion efficiency and, if possible, with improved efficiency. Clearly, the greater the number of engines the easier this is to achieve, but the more complex and costly the system. The three-

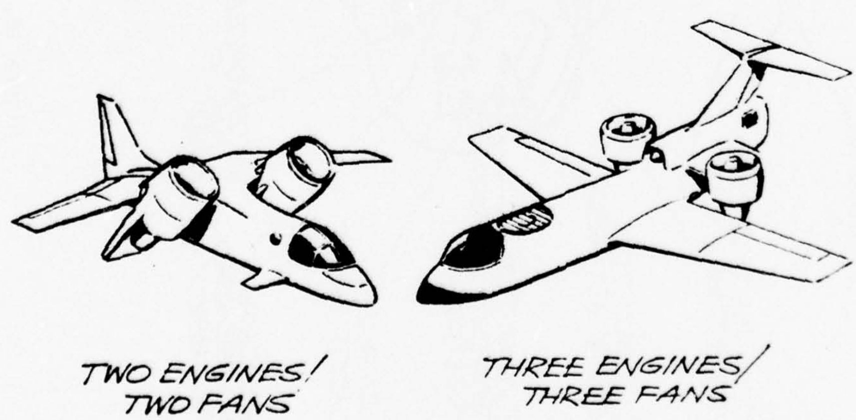

Figure 17. Typical V/STOL Aircraft Concepts with Shaft Power Coupling

power-generator system may be the best compromise.

To enable the fans to adjust rapidly to a reduced total power situation, some form of variable geometry is almost certainly required. The simplest in concept is the variable pitch fan, which can maintain speed while reducing pitch and fan pressure ratio. An alternative is the variable inlet guide vane system coupled with fixed fan pitch, which may have difficulty in coping with very large power differences. The simple fixed pitch fan is not possible because of speed matching on its power turbines in the engine failure case. Fig. 18 shows a typical V.P. fan with side-drive system.

The main cycle considerations can be summarized as follows:

1. The lowest specific thrust which is practical from geometrical and weight considerations
2. A moderate to high cycle pressure ratio obtained without excessive complexity and overall length
3. A choice of turbine temperatures biassed towards reliability and offering the possibility of significant emergency ratings

It would be fortuitous if the characteristics of such an engine were also suitable for the civil short-haul market and, this certainly cannot be ruled out. However the military engine requires the additional complication of side drive and free wheel clutch between fan and power generator. The choice of gas generator layout may be influenced by system failure considerations. A sudden disconnect between fan and power generator running at power would lead to power turbine overspeed, especially if a free power turbine were used. Such a failure is a greater possibility in this type of engine due to the increased mechanical complexity of the drive system. A single-shaft system would have inherent overspeed protection on its top speed governor, as would a fan/IP compressor system. Arguments for high efficiency point the choice towards the fan/IP system.

The system illustrated is of this type and is based on experience from a U.K. Government sponsored V.P. Fan Demonstration programme involving the M45SD-02 engine. The fan unit of joint Rolls-Royce/Dowty Rotol design is manufactured by Dowty Rotol, using their propeller background, and employs ductile aluminum alloys as used on conventional propellers.

9. RESEARCH OBJECTIVES - MILITARY SHORT-HAUL AIR INTAKES

Many of the features discussed for civil short-cowl transport aircraft are common to the military V/STOL aircraft as well.

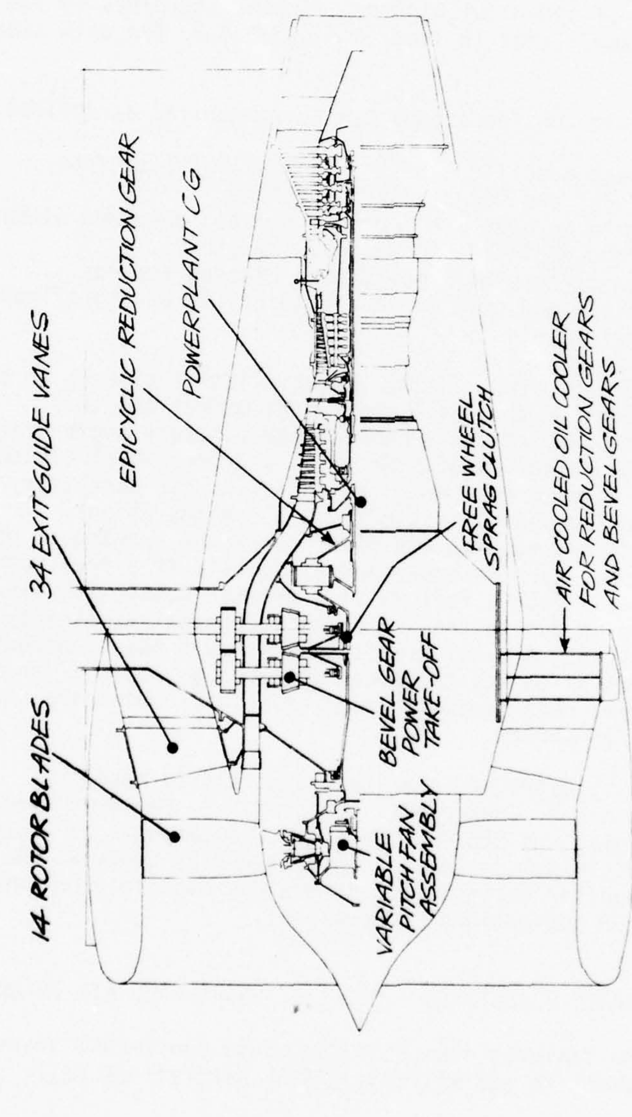

Figure 18. General Arrangement

The new problems will occur due to the need for powered lift, which may be obtained by vectoring engines or vectoring thrust. The former will lead to a large expansion in the incidence envelope over which the fan intake must work (Fig. 19). So far there has been little published work on the performance of fans and their inlet systems at high angles of incidence -- that of Tyler and Williamson (Ref. 2) being the most noteworthy. A performance map for a typical fan and intake system is shown in Fig. 20, and lines giving the variation of thrust for assumed take-off and landing manoeuvres are shown. The design of a fan and inlet system, to minimize the thrust losses shown and to ensure satisfactory fan blade stressing, needs careful consideration and should be the subject of further research. It should be remembered however that a number of aircraft with lift engines and vectoring fans have already been flown successfully in similar incidence situations.

10. VECTORING NOZZLES

The alternative to vectoring the engine is thrust vectoring. Although this is successful on aircraft such as the AV8A/Harrier and the VAK 191B, the use of a low specific thrust engine alters the problem considerably. The low nozzle pressure ratios mean that a given percentage pressure loss has a much larger effect on thrust

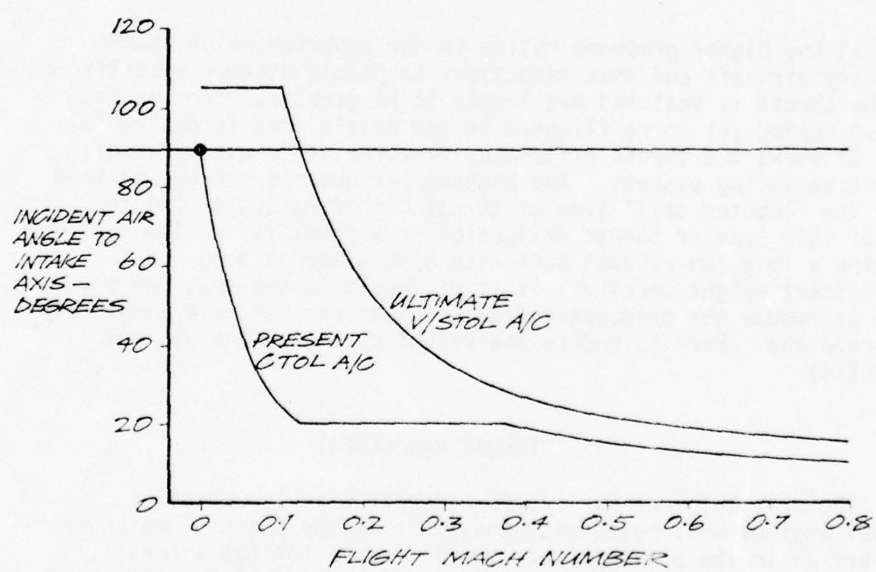

Figure 19. Intake Incidence Envelope where High Performance Desirable

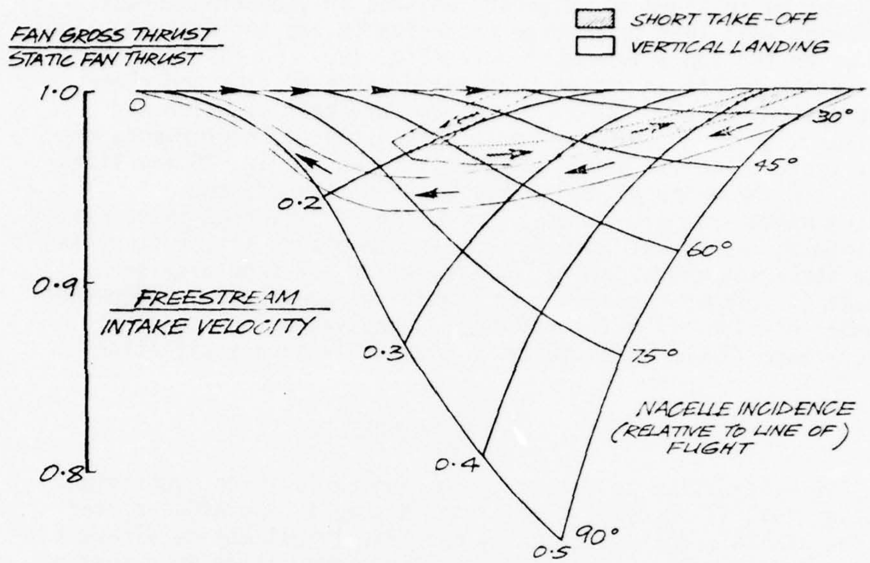

Figure 20. Typical Intake Performance at Low Speeds

than at the higher pressure ratios so far experienced on those military aircraft and that reductions in nozzle discharge coefficient as the thrust is vectored are likely to be greater. (For maximum thrust during jet borne flight, a larger nozzle area is desirable). Fig. 21 shows the thrust efficiency of three different types of thrust vectoring systems. The highest performance can be obtained from the "lobster tail" type of thrust vectoring system, but the use of this type of thrust deflection on a front fan engine will require a long fan exhaust duct with high external drag and a significant weight penalty. It is difficult to see what can be done to reduce the drag, apart from using an aft-fan or a completely separate one. Work to reduce the weight of the system will be essential.

11. THRUST MANAGEMENT

The critical area for V/STOL aircraft with low specific thrust engines will be thrust management in the event of an engine failure or in the provision of control forces for the aircraft. Regardless of whether lift is obtained by vectoring the engines or the thrust, some method of transferring power around the aircraft is needed, either by shafts or gas. Some form of fan variability would appear to be essential in order that engine speed (and

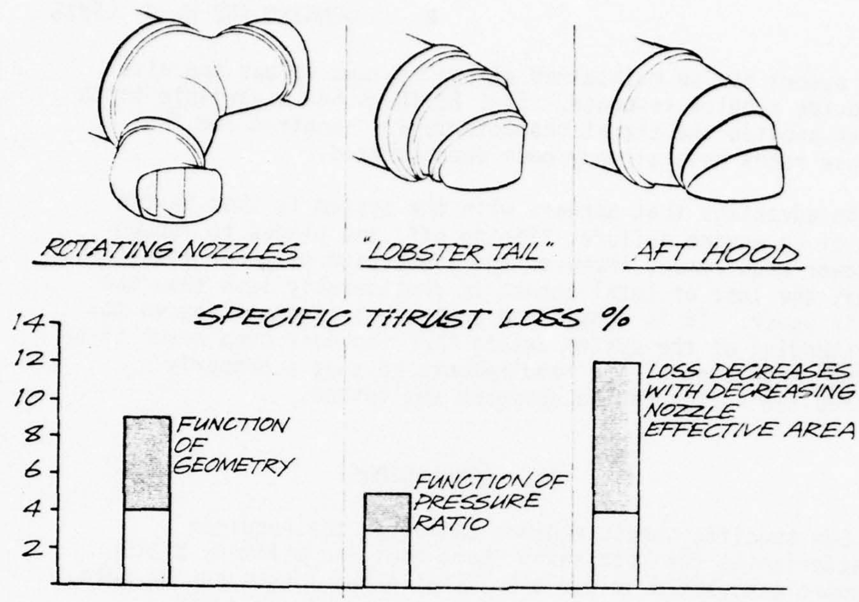

Figure 21. Performance of Thrust Vectoring Systems

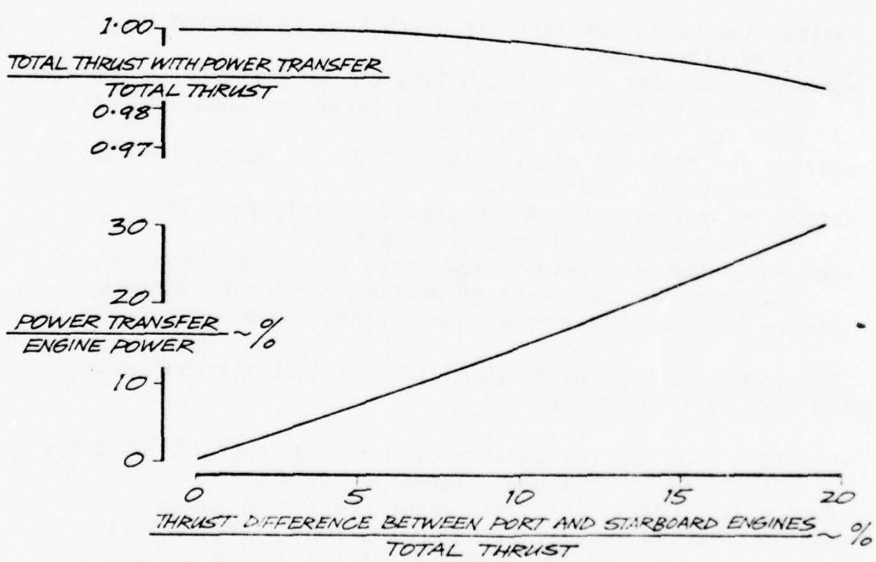

Figure 22. V/STOL Thrust Management with VP Fan Effect of Varying Blade Angle

hence power) can be maintained at the maximum values and also to provide a quick response. Fig. 22 shows how a variable pitch fan can provide the thrust characteristics required, and fast response rates have already been demonstrated.

An advantage that appears with the system is that in the event of an engine failure, "fining off" the blades to reduce the power requirement improves the conversion of power into thrust, so that the loss of total thrust is considerably less than the loss in power. It is considered that a satisfactory conceputal understanding of the system exists that the next step needs to be a firmer knowledge of the requirements, so that a properly tailored fan system can be designed and tested.

12. CONCLUSIONS

Low specific thrust engines can offer the required characteristics for both civil short-haul and military V/STOL transport aircraft, provided the installation losses can be held to an acceptably low value by using very short fan cowls.

The fan and its cowling is an area which is likely to be the critical component, and future work in this area will be required to--

- assess the merits of variable pitch fans as opposed to fixed pitch fans;
- determine whether fixed pitch fans can be designed to operate efficiently over a wide working range and thus avoid the need for a variable fan nozzle;
- design for high fan entry Mach numbers to reduce fan and fan cowl diameter and weight;
- study the performance of low pressure ratio fans in conjunction with ultra-short intakes;
- determine the requirements for noise and thrust reversal and study the best ways of meeting these requirements;
- ensure low engine/airframe interference installations.

Additional work peculiar to military V/STOL aircraft with powered lift will be needed to--

- maintain high fan performance over a wide range of incidence conditions by suitable intake/fan design;
- design low weight, efficient thrust vectoring systems;
- ensure good progressive thrust management characteristics to provide satisfactory aircraft control during jet borne flight and also to cater for the engine-out case.

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2. R. A. Tyler and R. G. Williamson - Thrust performance of podded lift-fans in crossflow. AGARD P.E.P. 17th-21st September 1973.

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DISCUSSIONS

WELLIVER: (Boeing Military Airplane Division)

With the advent of wide body jets and high bypass engines, one of the things that we got into was that the maintainability on the high bypass engine, across the board, was a little more than we had anticipated. And from the experience that we had on some of the narrow body jets up to that time, and now with these new engines as you are describing with the variable pitch and so on, how do you see, what do you see in the future in terms of the maintainability and reliability of these engines? Will they be equivalent?

DENNING:

Well, I don't have a fellow manufacturer here to help me with this one. I think that the history of the propeller operation is less satisfactory than jet and turbofan operation and admittedly there were problems with propellers and there were problems with gear boxes, but there were problems with engines generally in the past which we would not expect in the future. There is bound to be some increase in the complication of the engine due to the gear box and, undoubtedly, to the variable pitch mechanism. I can only hope that it would not constitute serious problems. The short-haul operators are operating those sorts of systems at the moment. The problems with the high bypass engines in the wide body airplanes are of a rather different nature. I feel they are connected in some cases with the size, the actual physical size, of the large engine and the flexibility of the engine, connected with the introduction of very high temperature air-cooled blades. I think perhaps it was not fully realized by any of the big

manufacturers of the time that there were very significant problems with thermal shock, thermal fatigue in the blades themselves, and that has probably constituted one of the biggest problems in that type of airplane. I don't see why we should be hit by that type of problem on this sort of engine because I hope we have learned our lessons. And also, I don't think that the technology levels that you will use with the smaller blades and the smaller engines will be as high as they are with the bigger engines.

CRAGIN: (General Dynamics)

In regard to the military side and again in relating to the question on reliability, have you done any looking at the trade-off between variable inlet guide-vane systems and variable pitch fans and whether or not you will need variable geometry LP turbines with those systems? How does all of that trade off against reliability? In your mind, is a variable pitch fan more reliable than VIG's or does it work better?

DENNING:

It is a difficult question to answer, about reliability, because we have no background on that, and we are studying the alternatives. We have a system with a variable pitch fan which we know will work. Obviously we have experience over the years on variable inlet guide-vanes, and we know that those will work. Whether they will work down to a situation where you lose 50 percent of your power and you still want to maintain the speed of the fan and get a sufficient thrust generation out of the fan, I am not absolutely sure. I don't have the facts and figures. But we are looking at the subject. We have also looked at the weight and geometric complication of variable inlet guidevanes on large diameter fans, and it does look as though you are not going to save very much in weight. You might not save anything. And then you have an anti-icing problem as well, and I must confess I do not know the final answer on comparison of the two systems. Clearly it will have to be evaluated properly before any final decisions arise on this sort of airplane. I just cannot give you a better answer than that, I am afraid.

MIKOLAJCZAK: (Pratt & Whitney Aircraft Group)

In the past, Rolls-Royce advocated the concept of using separate-lift engines. Have you now abandoned that approach, and if so, what was the reasoning behind it?

DENNING:

If you mean, have we abandoned it in the context of the Navy V/STOL airplanes, I would say the answer is no. There are still various

opinions in the Rolls-Royce Company about lift engines versus vector thrust engines and vectored engines. I think we have taken on board the idea that by cross-shafting you certainly keep down the total frontal area of the propulsion systems required for the airplane because it is quite obvious, if you lose an engine or you have separate non-coupled engines, you have to shut down another engine to balance that situation unless they are all clustered around the CG of the airplane and you can balance it out with some sort of control force. So I think that the answer is no, we have not ruled it out, but our attention at the moment is concentrated on the shaft coupled proposal.

WEINRAUB: (Naval Air Systems Command)

You showed a chart in the beginning of your presentation which, if I interpreted correctly, indicated that low pressure ratio fans are equal if not a little better in cruise specifics than these advanced technology propellers?

DENNING:

My second curve showed that advanced technology propellers just about broke even with low pressure ratio fans at about 0.8 Mach number, and as you go up in fan pressure ratio, I think the break even point will then move closer and closer to 0.9 Mach number.

WEINRAUB:

Does that graph only have implications for short haul aircraft or, could one also infer that you are suggesting that this is all the same for longer distance type of aircraft?

DENNING:

I think that if you say a long distance airplane -- its economic cruise Mach number is say about 0.8 to 0.84 -- then it has some implications. If you could mount short nacelle high bypass engines, then you might be able to get that advantage, which is on order of about 15 percent in internal performance. However, you have to remember that intake design is sensitive to Mach number. They require greater length of intake duct in order to get the correct match of intake velocity to flight velocity, so you will be stuck with a bit more weight of power plant and a bit more drag of nacelle than I have indicated for 0.7 Mach number where you can have 0-length intake. I am not drawing a firm conclusion that you ought to have very high bypass engines for very long range high speed subsonic airplanes.

KEMPER: (Vought Corporation)

What has your experience been with the angle of attack capability with 0-length inlets?

DENNING:

I think I have to admit that we have no experience of an engine with an 0-length intake running at an angle of attack.

KEMPER:

Are you proposing those for the Type A V/STOL activity,or are you considering their use on, maybe, the fuel efficient engines?

DENNING:

I was first considering them for fuel efficient engines. Clearly, I think you want to get the intake as short as possible on the NAVY V/STOL airplane,consistent with designing it for the right flight speeds. I think if you want to fly at 0.8 Mach number,you would have to think twice whether you made it at 0-length intake or not. You obviously want to keep the weight down. There is another aspect of the problem; namely, if you have a reasonable length of intake duct ahead of that fan and you do have a lip separation, then clearly it has time to mix out and create for you a large area of non-uniform flow. If you have an 0-length intake, even if it separates at the lip, it will go straight into the fan before it has had time to spread. That is perhaps another area where a bit of research is called for--looking at what happens to 0-length intakes when they have a separation at the lip.

KEMPER:

Yes, I agree with you.

BRADLEY: (General Dynamics)

Let me return to your shaft-driven concept for military application. What shaft speed do you envision for this kind of coupling?

DENNING:

We are envisioning taking the side-drive off after the gear box, so that the shaft drive in the particular example will be round about 4000 rpm; I think that is about doubled in the side-drive gear box, so that is about 8000 rpm.

CRAGIN: (General Dynamics)

In relation to high rpm shafting, what levels of horsepower are you considering to be feasible for about 30,000 or 40,000 pound airplanes? If you are having to run these shafts up to the nose and through gear boxes, things of this sort obviously are going to tend to get quite large and quite heavy when you go to high horsepower.

What are you considering to be a feasible, state-of-the-art level for horsepower in these transmission systems?

DENNING:

Well, if we intend roughly 50 percent of the power from one engine to the other engine, we are talking of shaft power up to around 6000 horsepower. The thing that worries one most of all is the weight of these gear boxes.

CRAGIN:

Well, the size of the shaft too?

DENNING:

Yes. A little bit of innovation is required there, I think, to get the weights down because you could kill the whole project if you are not careful and you do not design those gear boxes properly. We think it can be done, by the way.

MIKOLAJCZAK: (Pratt and Whitney)

You indicated that a short cowl can be used to achieve low noise level if the throat Mach number is increased to about 0.85. At take-off, the forward fan noise and the aft-fan noise contribute about equally to the total fan noise. Do you have any experience which indicates that when the Mach number at the inlet throat is increased the aft-fan noise is reduced? At approach, the inlet throat Mach number will be significantly lower than at take-off. Does the short cowl allow you to achieve low noise levels at approach?

DENNING:

The figures, based on a set of test results, show that you do reduce the forward noise relative to the rear noise. The rear noise, all things being equal, you would expect to go up. I cannot answer your question as to how much it goes up. You do have a certain amount of cowl surface area at the back of the fan, and this can be quite effective in dealing with rear noise. But if you have a low pressure ratio fan, our experience is that the fan is not the dominant noise source. Now clearly, we expect to be reducing aft-jet noise to a point where, maybe, the fan will not be a problem in meeting the anticipated noise regulations.

MIKOLAJCZAK:

I understand that you are talking about low speed fans which contribute significantly to the overall noise level.

DENNING:

Absolutely. I am talking fan tip speeds of the order of 800 to 1000 feet per second. And I am just saying that it is better to have noise coming out of the back than the front because you have, with an 0-length intake, some chance of dealing with it out of the back, if it should be a problem

THE SHAFT COUPLED LIFT/CRUISE FAN PROPULSION SYSTEM FOR V/STOL AIRCRAFT

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ABSTRACT

The Navy is planning for the development of a family of subsonic, *multimission*, V/STOL aircraft. In an aircraft of this type, the interrelationship of the airframe and the propulsion system is much closer than in conventional aircraft. The integration of the propulsion system into the airframe, therefore, naturally assumes much greater importance. The shaft-coupled propulsion system appears to be the best choice for the V/STOL application under discussion. Its components are either in existence or in an advanced state of development. Methods of flight control and provisions for loss of power are integral parts of the system and are based on the technology used in some of today's large helicopters. The subsonic V/STOL aircraft is a near-term reality, but additional research into some of the aspects of the propulsion system will offer a high payoff when the aircraft becomes operational.

INTRODUCTION

Within a few short years, V/STOL aircraft have passed through the novelty stage to become a major factor in naval force planning. Not too long ago, V/STOL aircraft were somewhat of a curiosity - interesting, different, but hardly ready to assume a serious role in aviation, either military or civil. Today, squadrons of U.S. as well as USSR V/STOL aircraft are operating off their respective carriers in the Mediterranean.[1]* Although their operational capabilities are somewhat limited, these aircraft have demonstrated the tremendous versatility afforded them by their unique V/STOL characteristics. Moreover, the V/STOL aircraft that are under development today will not only be capable of missions flown by the most effective aircraft in the current inventory, but will have the additional ability of operating from ships of almost any type or size. This ability will permit the Navy to place more reliance on small, dispersible aircraft cariers as well as an aircraft operating directly from ships of the frigate or destroyer class and will, in turn, have a decisive effect on naval force requirements and strategy.

Two types of V/STOL aircraft are expected to bring about this "revolution" - the subsonic (Type A) and the supersonic (Type B).[2] Of these two, the Type B appears to be far over the horizon; in fact, its requirements have not yet been fully defined and, consequently, it will not be discussed here because much of the discussion would be strictly conjecture. The Type A subsonic V/STOL, on the other hand, is a near-term aircraft for which the Navy has already begun a major development program that will lead to the introduction of advanced-design, multimission V/STOL's into the fleet by the late 1980's or in the early 1990's. Approximately 40 million has been budgeted toward this effort in fiscal 1978, and the development costs over the next five years could exceed $1 billion.[3] Figure 1 is a projected milestone chart that takes the program from development through initial deployment into service.[4]

Subsonic V/STOL aircraft will have true multimission capability in that one or two basic designs will be able to handle most antisubmarine warfare (ASW), advance early warning (AEW), carrier onboard or vertical on-board delivery (COD or VOD), aerial tanker, search and rescue (SAR), and USMC assult missions. They will be capable of operating from existing carriers or from the proposed new 40-50,000-ton CVV V/STOL carriers. More important, nearly every existing Navy ship - with only slight modification - will be able to accommodate these new aircraft, which will take over most of the work now performed by the Lockheed S-3A (ASW), the Grumman E-2C (AEW),

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* Numbers in brackets correspond to the references listed at the end of this paper.

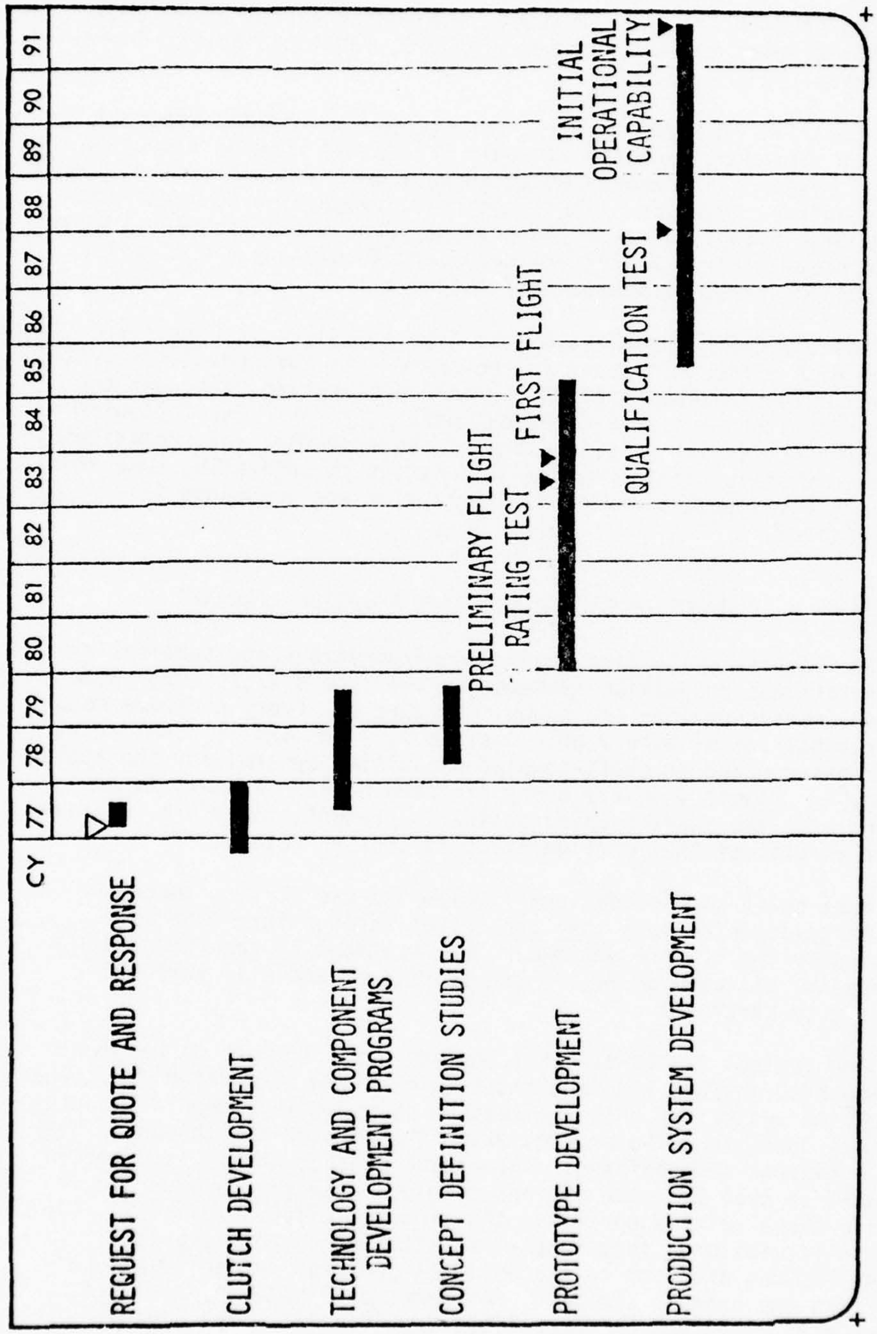

Figure 1. Type A V/STOL Development Schedule

the aging Grumman C-1 and C-2 (COD), the Boeing Vertol CH-46 (VOD and USMC Assault), the Grumman A-6 (Tanker), and the Sikorsky CH-53 (USMC Assault)[1, 2, 3, 4].

Naturally, an aircraft with such phenomenal ability does not spring up overnight, and the subsonic V/STOL has been no exception. Years of research and experimentation have been expended in the fields of aeronautics, avionics, flight controls, propulsion systems, and related disciplines to reach the present point where the Navy is now able to program the final effort necessary to develop a cost-effective, mission-capable, operational aircraft.

Probably the key element of the Type A V/STOL aircraft - and the one with which this paper is concerned - is the propulsion system. Its many unique characteristics for this application have been the subjects of intensive investigation over the years. The following discussion delves into the findings of these studies and summarizes the present status of the system in addition to suggesting some areas in which additional research and development would enhance its capabilities.

PROPULSION SYSTEM FOR TYPE A V/STOL AIRCRAFT

The Navy is evaluating several configurations and combinations of aircraft and propulsion systems for the Type A V/STOL.[5] Some of these configurations are shown in Figure 2. Every airframe manufacturer has one or more viable designs[6, 7, 8] and is considering either gas-coupled or shaft-coupled propulsion systems for the application.[9] However, nearly every airframe builder appears to be working with the shaft-coupled design at present; therefore, we have chosen to discuss that configuration in greater detail.

Most shaft-coupled designs involve the use of two, three, or four large-diameter fans that are driven through interconnected shafting by two or more engines. The locations of the fans and the engines are dictated by design considerations and will vary from aircraft to aircraft.

One typical V/STOL aircraft configuration (Figure 3) has two side-mounted engines and fans and a third, forward-mounted fan. The propulsion system for this application is shown in Figure 4. During vertical operation, the engines drive the forward fan through a gearbox, a clutch, and shafting. The engine exhaust thrust is diverted downward so that the side engines, together with the forward fan, provide three-point support for the aircraft (Figures 3 and 5). (In some configurations, this support is attained by rotating the lift/cruise engines and fans to the vertical position rather than merely diverting the exhaust stream.) For forward flight, the clutch is disengaged. This action immobilizes the lift fan. The exhaust

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Figure 2. Representative type A V/STOL Configurations

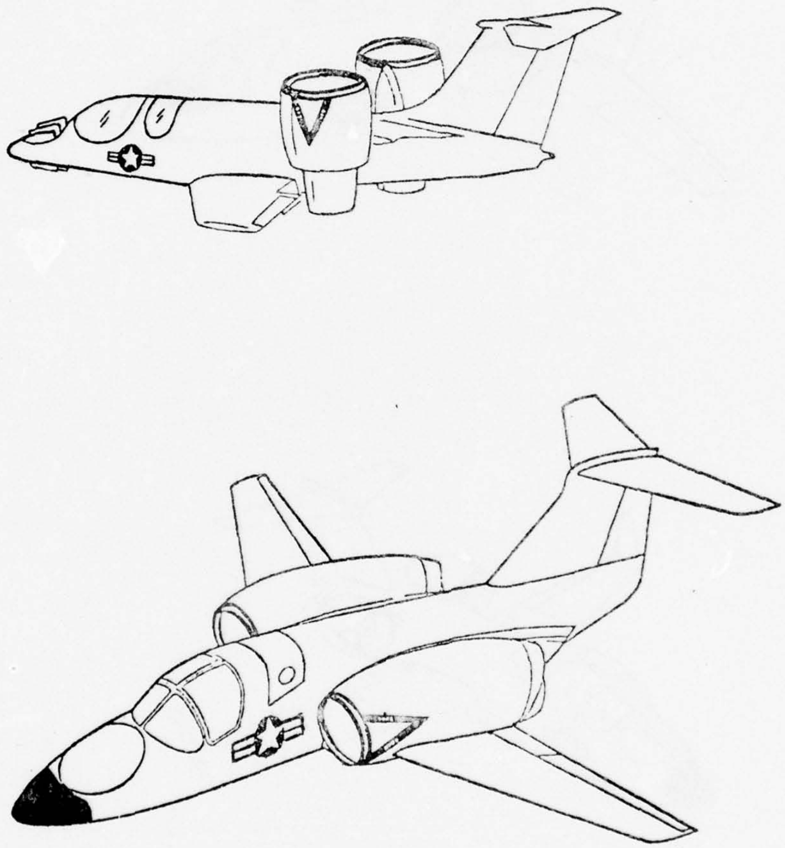

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Figure 2. Representative type A V/STOL Configurations

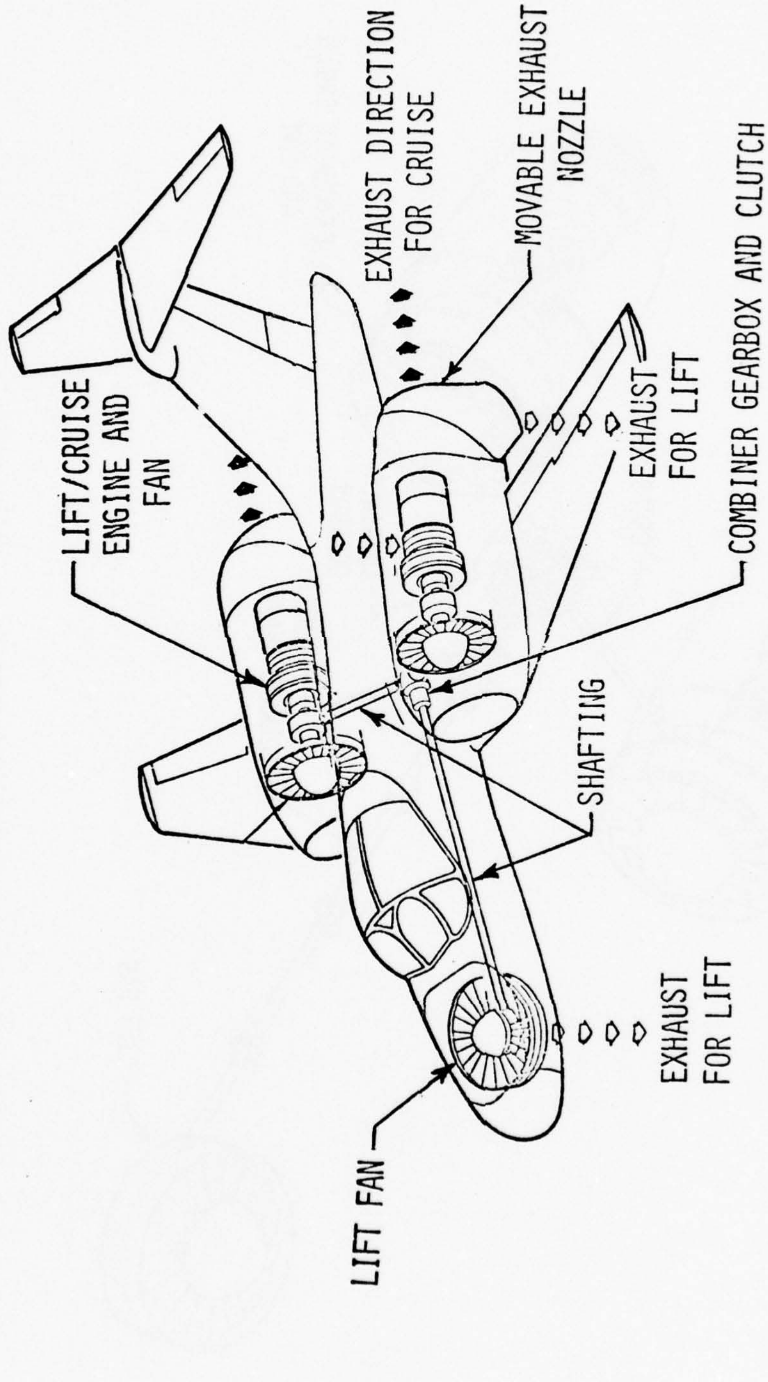

Figure 3. Typical Type A V/STOL Aircraft Configuration (2-Engine, 3-Fan)

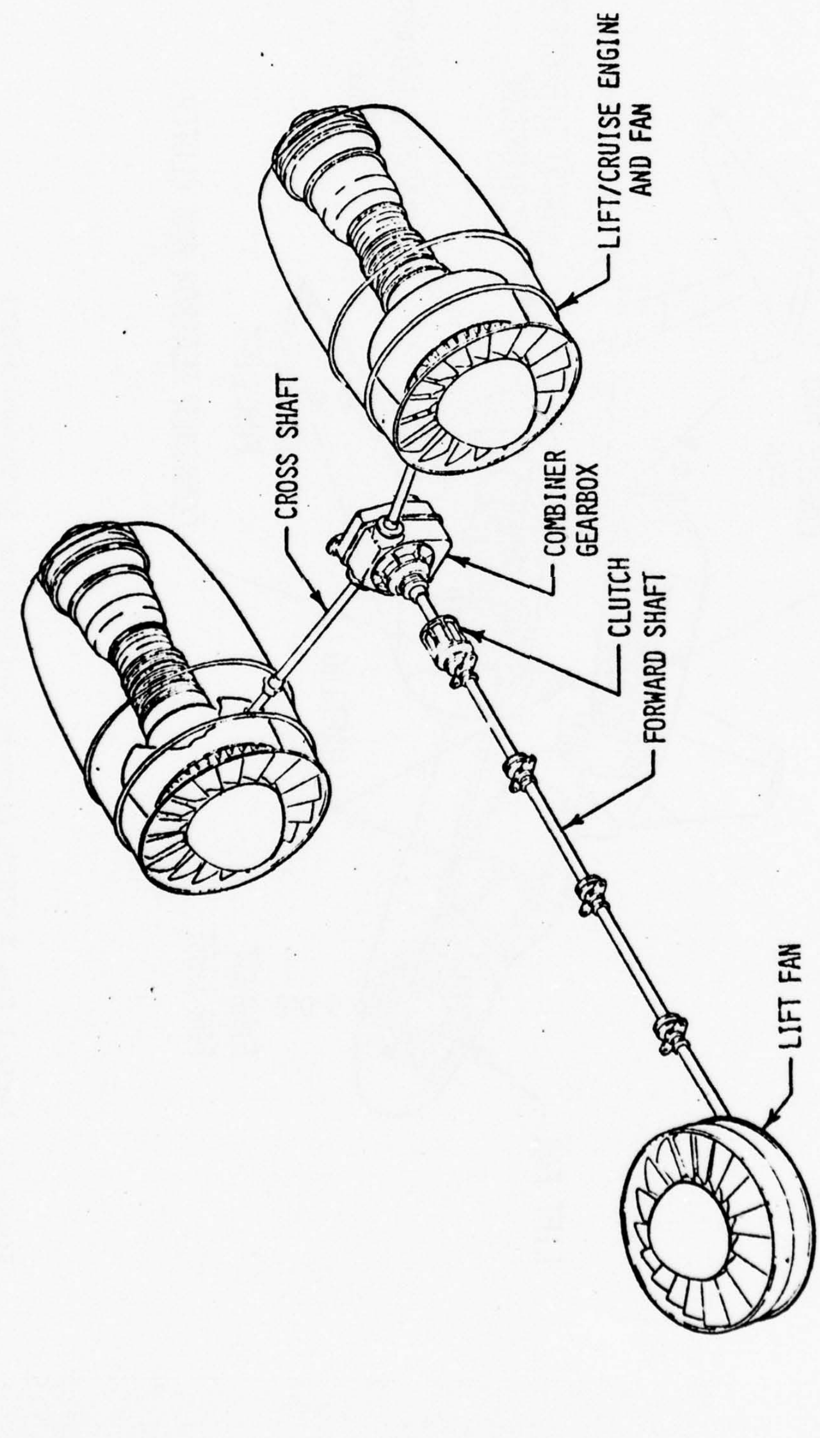

Figure 4. Lift/Cruise Propulsion System (2-Engine/3-Fan Configuration)

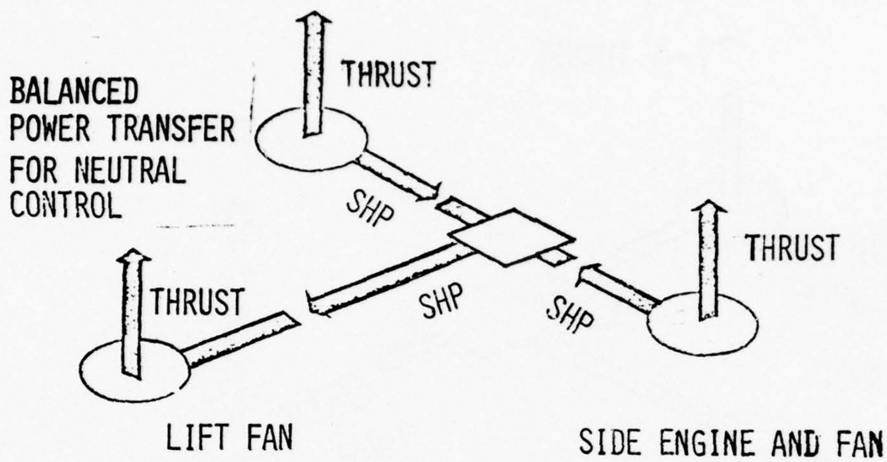

Figure 5. Three-Point Support for Vertical Flight

nozzles are opened rearward (or the engines are rotated to their horizontal positions) so that the side-mounted lift/cruise engines and fans power the aircraft in a conventional manner to provide conventional aircraft performance.

Aircraft Control System

Control of the aircraft in vertical flight and provision for a one-engine-inoperative (OEI) condition are primary considerations in the design of any V/STOL aircraft. For the configuration shown in Figure 3, control is obtained by (1) varying the fan flow with the fan at constant speed, (2) movable vanes in the front fan exhaust, and (3) the independent deflection of the side engine exhaust stream (or rotation of the side engines). Note that the fans become an integral part of the aircraft flight controls. Also, each of these is a quick-response means of control and does not depend on the slower response of engine power or speed changes. For example, in a roll maneuver, the flow of one side fan is increased, that of the other is decreased, and power is simply transferred from one fan to another as illustrated in Figure 6. For a pitch maneuver, the power is transferred from the side fans to the forward lift fan. Other V/STOL aircraft concepts, which may have two, three, or four fans, and use similar controls: variable fan flow, fan exit vanes, and

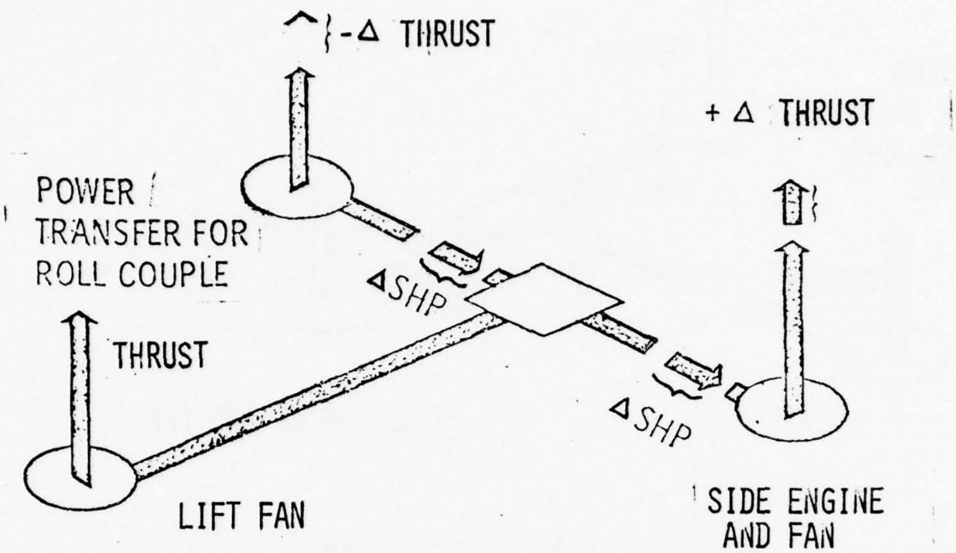

Figure 6. Power Transfer for Thrust Attitude Control

exhaust deflection and/or physical rotation of the fans or the engines.

Provision for Loss of Engine Power

The OEI condition is the most critical one that the V/STOL aircraft faces. Upon the loss of one engine during vertical flight, the remaining engine(s) must immediately take over the job of driving all the fans with a very minimum in loss of propulsive thrust. This is accomplished by the interconnecting shafting, which permits each engine to drive all the fans, and by the use of a redundant control system to insure against failures in a single control system. Each of the engines drive through an overrunning clutch so that an engine, should it stop, is automatically disconnected and does not threaten the entire propulsion system.

The engines can also be so rated that short-time operation at substantially higher power is permissible. In addition, designers are considering a number of techniques that include variable engine flow (variable cycle), thrust augmentation (water alcohol), and variable turbine cooling to enhance this short-time, high-power

capability. If short-time higher OEI power is available, the control system must provide it instantaneously when required and, in addition, must be able to monitor its utilization to prevent its excessive or unauthorized application that could shorten the engine life unnecessarily.

Control System Integration

The separate engine, fan, and aircraft control systems must be integrated into one overall system. Control of the aircraft no longer involves only wing and tail surface movement, but also includes such aspects as fan flow control, engine power response, and, for some concepts, clutching in the forward fan, moving diverter valves, rotating engines, and moving vanes in the fan and engine exhaust streams. All these functions must operate smoothly and with adequate redundancy and self-checks to avoid a loss of control arising from the failure of individual components. The control system should also provide for equal load sharing between engines during normal operation and for take-over by the remaining engine(s) when one engine loses power. Current state-of-the-art, "fly-by-wire" electronic control systems, such as those developed for the Army's Heavy Lift Helicopter (Figure 7), have the ability to do this job, but must, of course, be tailored to the requirements of each indiviual aircraft configuration. Fully electronic controls eliminate the need for conventional mechanical control cables, levers, rods, etc., and provide a lightweight system with greater redundancy than is possible with mechanical controls.

Control System Development

Each of the control system components has already reached the state of development where it can be integrated into an effective aircraft control system. The Heavy Lift Helicopter, for example, had three engines to drive two rotors (Figure 8), similar to some V/STOL configurations. It used an advanced, "fly-by-wire" electronic control system to handle power transfer, load sharing, and the loss of power in the same manner as required for a V/STOL control system.

The selection of control system concepts should run parallel with the current aircraft and propulsion system configuration trade studies. Because the final control system must be tailored to match the selected aircraft configuration, control system development may be delayed somewhat, pending the selection of the final aircraft configuration(s). The simulated propulsion system test suggested under the subsequent "Drive System Development" subheading would provide an excellent vehicle for the early development and demonstration of a prototype control system.

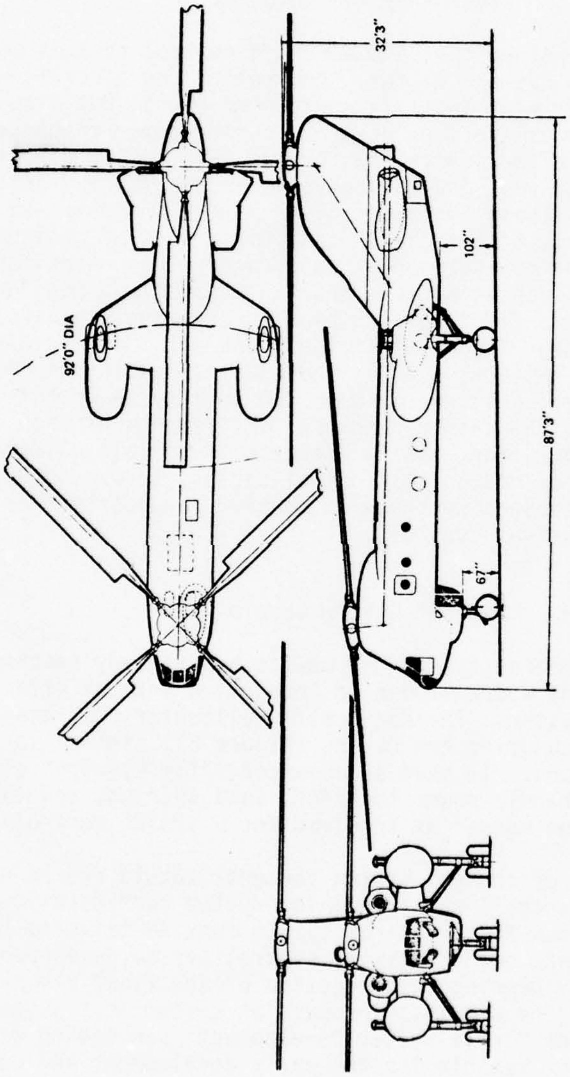

Figure 7. U. S. Army Heavy Lift Helicopter

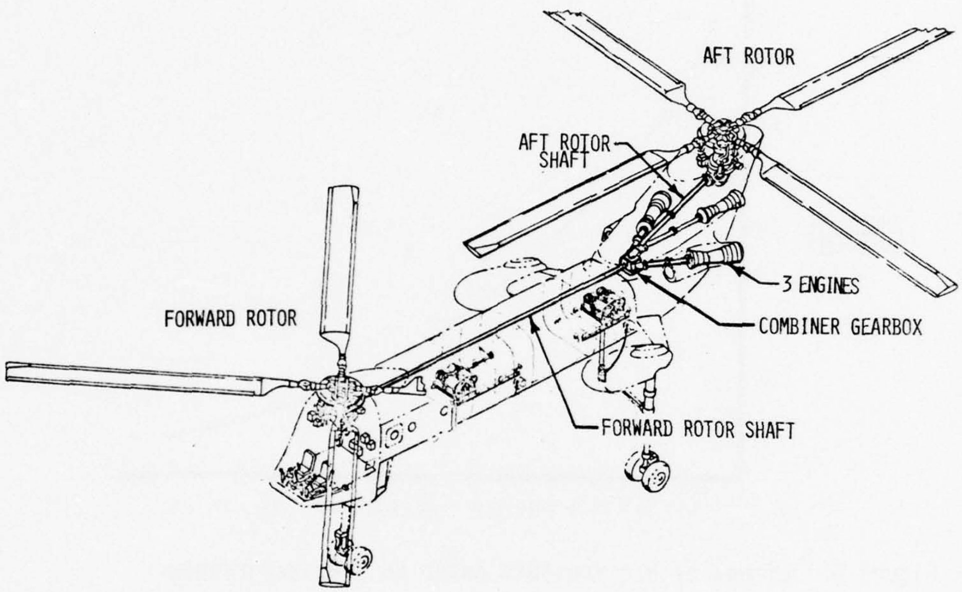

Figure 8. Heavy Lift Helicopter Propulsion System

Fans

Each V/STOL aircraft mission has different requirements-range, payload, desired speed, loiter time, etc. Each of these requirements tends to favor a different facet of fan and engine cycle characteristics. For example, lower fan pressure ratios (higher bypass ratios) result in greater low-speed thrust (greater vertical lift), as reflected by Figure 9, but also in lower efficiency at high speed. The lower fan pressure ratios also bring about the need for larger-diameter fans for a given horsepower, which tends to compromise high-speed flight, and a larger fan flow, which makes turning the side fan exhaust to obtain vertical lift more difficult. Higher fan pressure ratios (lower bypass ratios) tend to favor highspeed flight. Currently, fan pressure ratios in the range of 1.2, 1.3 and 1.4 are being studied for the various aircraft concepts and evaluated against the various missions.

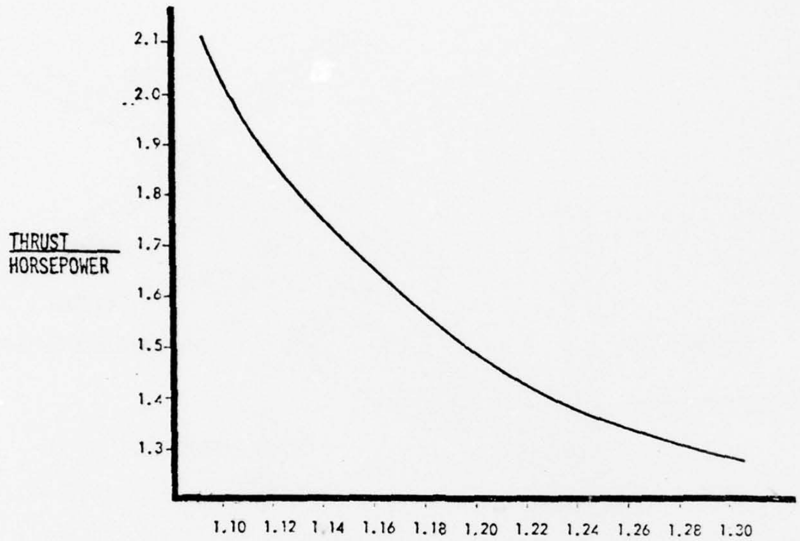

Figure 9. Effect of Fan Pressure Ratio on Thrust/Horsepower

For rapid response, the lift fans must have the ability to vary flow while operating at constant speed. Approximately a ± 25% thrust variation at constant fan speed is required for aircraft vertical control. The criticality of maintaining control of the aircraft in the vertical mode does not allow time for varying the flow by changing the fan speed. Two methods of achieving the required fan flow variation are available: variable inlet guide vanes and variable-pitch blades. Both offer high efficiency and adequate thrust modulation, but both are relatively new developments for today's high-bypass-ratio fans.

Aircraft-Fan Integration

Integration of the lift fans into the aircraft has a strong effect not only on the aircraft configuration but also on vertical as well as conventional aircraft performance. For example, the use of the forward lift fan tends to broaden and flatten the aircraft nose, which could affect the location of nose radar and armament, pilot visibility, and, possibly, cockpit width or height. The forward fan shafting must be placed low enough to clear the cockpit. The side engines must be carefully located with respect to the aircraft center of gravity to provide stable vertical control.

As mentioned previously high fan bypass ratio provides greater thrust per horsepower whereas lower bypass ratio provides greater efficiency and less drag for high-speed flight. Thus, it would be desirable to have high-bypass fans for vertical flight and low-bypass fans for cruise. It is possible to attain both objectives with the 3-fan/2-engine arrangement shown in Figure 3. In the cruise mode, the two engines drive two fans. Cruise bypass ratio is, therefore, the ratio of the airflow for the two fans divided by the airflow for two engines. In the lift mode, the same two engines drive three fans, so that the bypass ratio then becomes the ratio of airflow for three fans divided by the airflow for two engines. Bypass ratio has thus increased 50% for the lift mode. Because thrust/horsepower is a function of bypass ratio, the effect is that of increasing the lift thrust for the same available horsepower. This is an important consideration, particularly for the OEI condition. Other fan/engine arrangements offer the possibility of supercharging the engine, which provides increased engine flow and increased power. Thus, the arrangement of fans and engines in the airframe not only may affect the basic airframe configuration and design but also offers the possibility of providing a variable bypass ratio for increased lift thrust and engine supercharging for increased power and better cruise efficiency. Obviously, the fan arrangement has a strong influence on the aircraft configuration and on mission capability.

Future Fan Development

Two areas of further fan development effort are suggested. First, trade studies are required to select the optimum fan size, pressure ratio, and configuration that will best match the desired multimission characteristics of the aircraft. This is a complex study that is closely tied to the current aircraft/propulsion system optimization studies.[10] Second, further development of the variable guide vane, as well as the variable blade design, is worthwhile. Both designs promise to provide the required flow variation with good efficiency. Further development testing will prove the superiority of one over the other.

Engines

[illegible]

Current Engine Studies

Detroit Diesel Allison Division of General Motors developed an engine of the type under discussion - the XT701 (Figure 10) - for the Army's Heavy Lift Helicopter. In the Heavy Lift Helicopter, the problem of OEI was dealt with in exactly the same manner as discussed for V/STOL aircraft, so that the engine arrangement and requirements were very similar to the V/STOL engine requirements. In the Heavy Lift Helicopter, three engines drove two rotors through overrunning clutches and interconnected shafting (Figure 8). The engine was required to have a short-time high power rating to handle OEI. The control system provided load sharing and "fly-by-wire" operation. The existing XT701 engine produces about 8000 SHP, which makes it a suitable powerplant for early prototype V/STOL aircraft. Most airframe companies are using this engine with the Hamilton Standard variable-pitch fan for their prototype aircraft studies. An advanced, higher rated version of this engine, the Model PD370-24[11], (Figure 11) is being used for operational aircraft studies. This engine drives a variable-pitch fan through an overrunning clutch and reduction gear; it incorporates a bevel gear and cross shaft for power transfer.

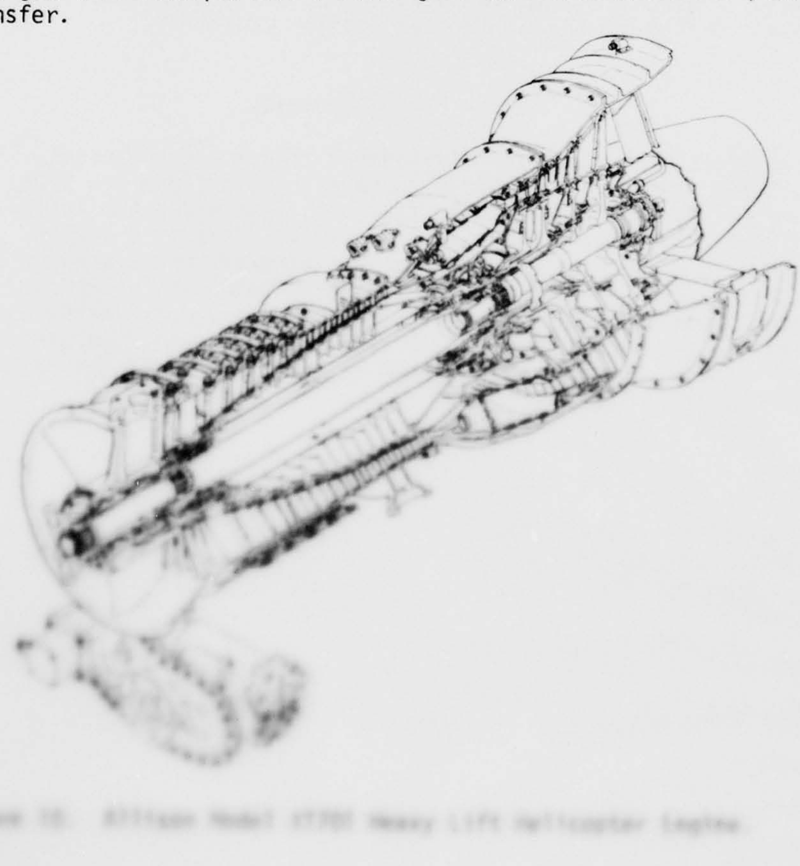

Figure 10. Allison Model XT701 Heavy Lift Helicopter Engine.

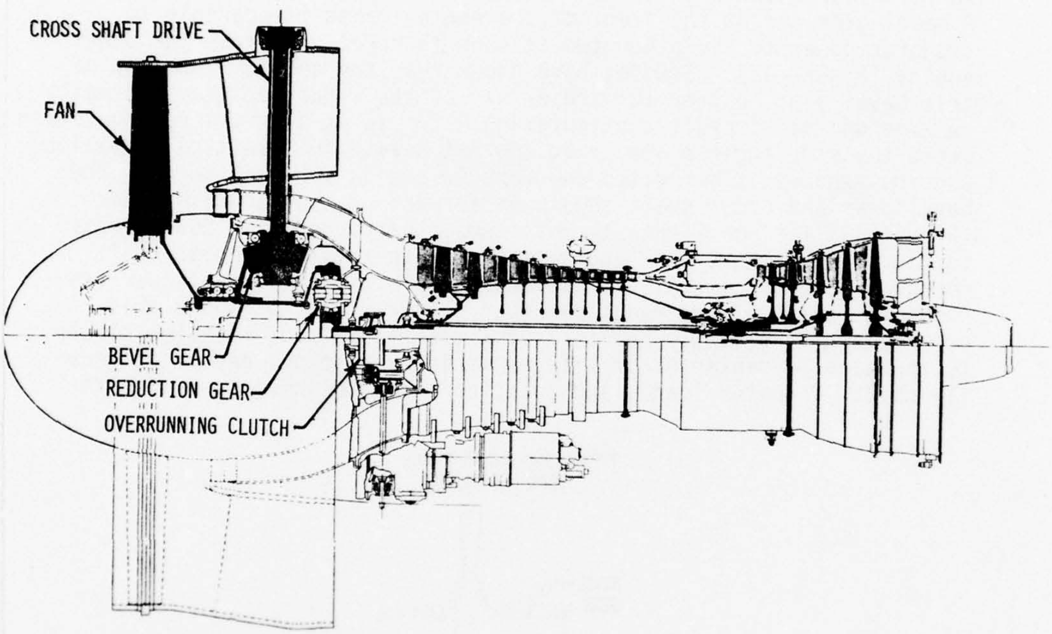

Figure 11. PD370-24 Series Lift/Cruise Engine

Aircraft/Engine Integration

The optimum propulsion system configuration will require careful integration of the engine with the fan and the airframe. In each of the V/STOL concepts, the fan and engine are closely tied to the overall system with respect to mechanical arrangement, control, and performance. For example, the side engines of the two-engine, three-fan configuration (Figure 3) drive the fan through a reduction gear. A bevel gear set at the front of the engine makes it possible to transfer power to the other two fans or to receive it from the other engine (Figure 12). Studies have shown that the optimum location of this bevel gear, either forward or aft of the reduction gear, actually depends on the aircraft configuration.[12] In an aircraft configuration where the side engines are fixed (do not rotate to a vertical position) and the exhaust is deflected downward to obtain vertical thrust, the bevel gear and cross shaft should be forward of the reduction gear (Figure 13) for the lightest, most compact arrangement. For aircraft configurations where the engine must rotate about the cross shaft center line, the best overall arrangement is with the bevel gear and cross shaft aft of the reduction gear (Figure 14). Although this arrangement is slightly heavier, it keeps the axis of rotation closer to the nacelle center of gravity to minimize aircraft center of gravity shift, actuator loads, and rotation time and increases aircraft

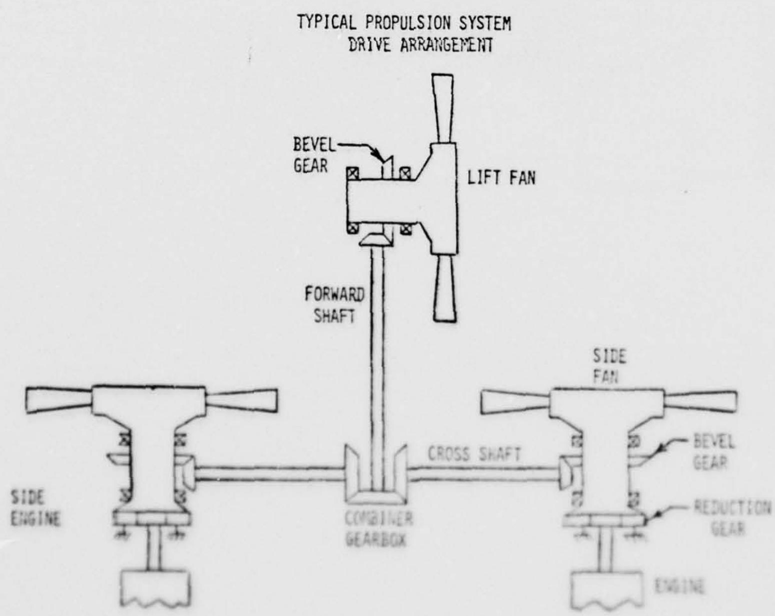

Figure 12. Typical Propulsion System Drive Arrangement

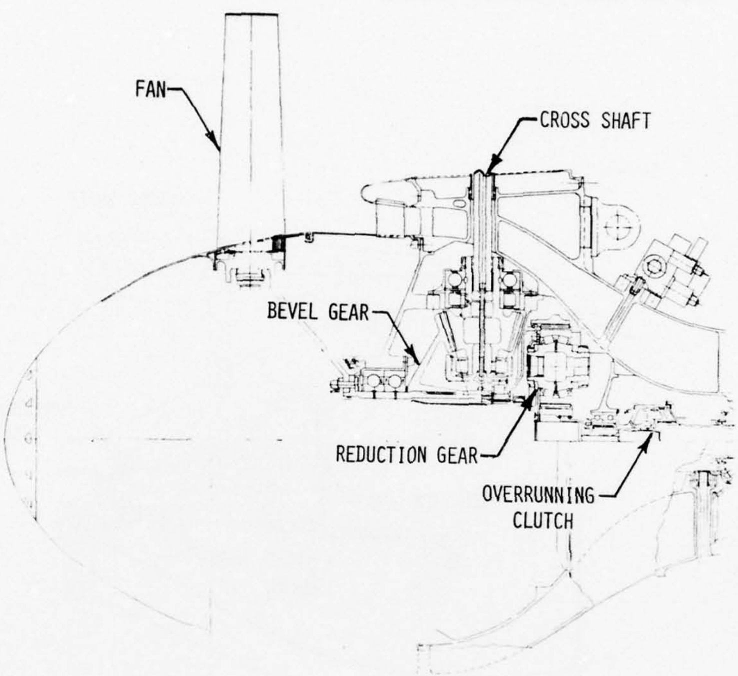

Figure 13. Navy Multimission V/STOL

ground clearance when the nacelle is vertical. This is a case, then, where, despite the heavier engine, the best overall system has been attained. Other mechanical features, such as fan and engine ducting, mounts, and lube oil coolers, must also be optimized for each specific V/STOL aircraft configuration.

Aircraft size, weight, and performance must be tailored to the required mission. The selection of the optimum engine size, cycle, and rating philosophy involves many variables, some of which are dependent not only on the mission but also on the aircraft configuration. For example, adapting a single basic aircraft to both the ASW and Marine Assault missions is a difficult proposition. The ASW mission can be accomplished with a relatively lightweight aircraft, and the OEI vertical landing requirements can be eased by jettisoning stores and possibly permitting a high sink rate (greater

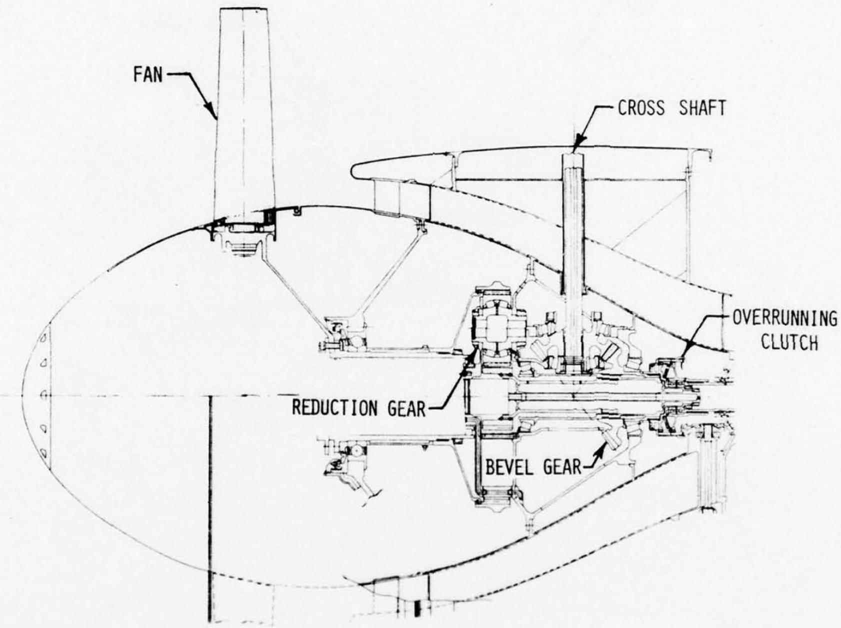

Figure 14. Bevel Gear and Cross Shaft Located Aft of Reduction Gear

loss of thrust) and ejecting the crew. In contrast, the Marine Assault mission requires a greater payload and includes such desirable features as high speed and long range capability at low altitude. Furthermore, the mission requires the absolute assurance that a vertical OEI landing can be completed satisfactorily without the need for ejecting or casting anything overboard, particularly the full load of armed Marines. Consequently, the Marine Assault aircraft calls for a substantial increase in the OEI power requirements.

One way of attacking this problem is the addition of a third engine to the basic aircraft (Figure 15) for the Marine Assault version. This solution keeps the changes in the basic airframe or

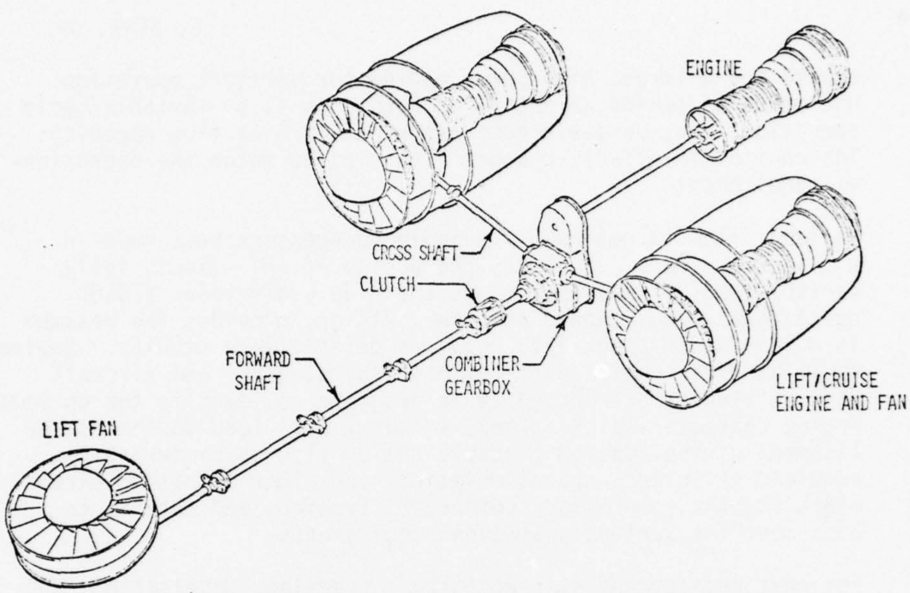

Figure 15. Navy Multimission V/STOL Lift/Cruise Propulsion System 3-Engine/3-Fan Configuration

propulsion system to a minimum and provides the desired multimission capability. It does, of course, require that the airframe and propulsion system concept be capable of accepting a third engine - that is, the integrated airframe/propulsion system design must, from the start, consider both missions and both the two-and three-engine arrangements.

The foregoing is only one typical example of the many perplexities that face the planner of a multimission V/STOL aircraft and mandate an unusually close airframe/engine integration.

Future Engine Development

Several areas of engine development appear to offer a high payoff for the V/STOL application. Each of these begins with current, well developed, high-bypass turbofan technology and tailors it to better meet specific V/STOL requirements.

1. Variable Cycle

The V/STOL missions require a small, efficient engine for

cruise and a large, high-power engine for vertical operation. One means of making an engine do both jobs is by variable cycle operation, i.e., variable compressor and turbine flow capacity. The engine, in effect, changes flow size to match the operational requirement.

Variable flow is not new. Variable compressors have been in service for years. However, the design of an optimum, fully variable engine that best meets the widely divergent V/STOL operational requirements and, in addition, provides the maximum in multimission capability requires detail trade studies. Engine cycle studies, engine/aircraft matching studies, and aircraft mission studies are currently in progress to identify the optimum engine characteristics. They, in turn, will lead to the establishment of the desired pressure ratio, flow characteristics, required efficiency characteristics, and other operating parameters for the compressor, combustor, turbine, and nozzles to best meet the various operating requirements.

The next development step would be the design, fabrication, and test of each component, followed by complete engine testing. These tests will provide verification of the predicted performance characteristics, provide a means for testing the engine control system, and provide an early availability of engines for the operational aircraft development program.

2. Engine Rating and Variable Cooling

The engines (and aircraft) are sized primarily by the OEI vertical requirement. The operable engine must be small and light enough to keep the aircraft gross takeoff weight to a minimum and yet be able to produce the highest possible power during this brief emergency. The extra burst of thrust can be made available through a variety of means including short-time ratings, variable turbine cooling, water-alcohol augmentation, etc.

Various philosophies are prevalent with respect to the question of how the engine should be rated to meet this situation. Some of those who are close to the problem feel that the conventional 30-minute Military rating should not be exceeded because the higher short-time ratings are often used to excess, a practice that results in premature engine damage. The use of a 30-minute Military rating for only the very infrequent OEI condition is an ultraconservative approach. It does, however, tend to be consistent with the requirement for high reliability and low maintenance. At the other extreme are those who feel that a one-time 30-second rating accompanied by substantial turbine damage is more realistic for OEI. In effect, this philosophy tells us to melt down the engine but save the aircraft. Although this approach will produce an aircraft with the lightest

possible gross takeoff weight, such a limited OEI rating probably does not provide enough engine life to permit adequate pilot training to handle this emergency. The best answer may lie somewhere between.

Studies to resolve the best means for obtaining short-time high power without compromising reliability or increasing engine size are indicated. These analytical studies should relate rating philosophy, variable turbine cooling, possible water-alcohol augmentation, and the mission requirements. They must take into serious account the recent service experience in which engine turbine life has not reached its expected levels because of the accumulation of more temperature cycles than expected or, as engines deteriorate, because of the use of the engines at higher than rated turbine temperatures. These studies could be followed by engine tests to verify the conclusions and demonstrate the required life and reliability of the selected system.

3. Weight Reduction

Current studies have confirmed that weight will play a greater role in the finalization of the V/STOL aircraft configuration than in past conventional designs. Aside from paying increased attention to weight in executing the design, the builder will purchase light weight only by incurring the penalties of increased cost, shorter life, and/or lower performance. Therefore, it will take thoroughly planned and executed weight/cost, weight/life, and weight/performance trade studies against the various V/STOL missions to ensure the best compromises in the final design.

Gearboxes, Shafting, and Clutches

Each of the V/STOL concepts utilizes shafting and gearboxes to transfer power during VTOL and/or OEI operation. In addition, the three-fan configurations incorporate a clutch in the forward fan shafting because the forward fan is used only during takeoff and landing. These drive systems incorporate proved state-of-the-art technology similar to that used in current helicopters. For example, bevel gear design parameters (Figure 16) are similar to those used in today's large helicopters. The specific equipment for each design must be tailored to meet the needs of each aircraft configuration. Indeed, the V/STOL propulsion drive system must be so closely integrated into the airframe that it is difficult to think of the V/STOL aircraft design and the propulsion system design as separate entities.

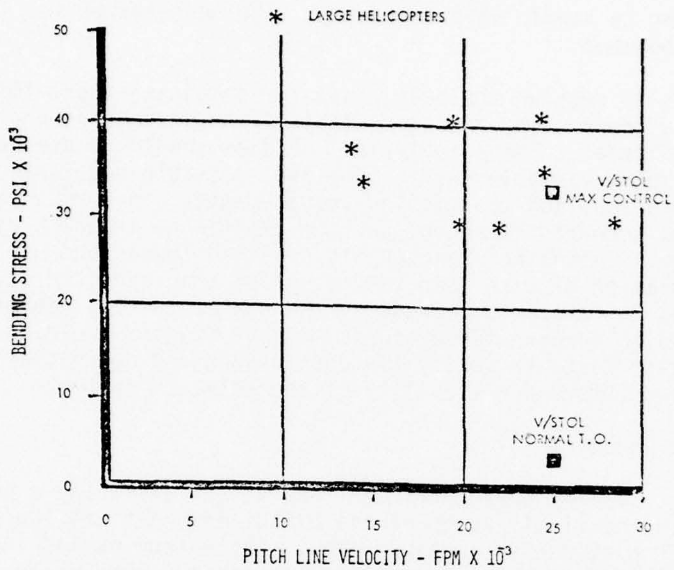

Figure 16. Spiral Bevel Gear Comparison

Drive System Development

The forward fan clutch is the first V/STOL system component to be under full-scale hardware development. In September 1976, the Navy awarded Detroit Diesel Allison a contract to design, fabricate, and test a full-scale, flightweight clutch suitable for V/STOL aircraft use. This clutch engages for vertical flight and disengages for cruise on command from the aircraft control system. For flight safety, it is designed to remain positively engaged while the aircraft is in the vertical mode, so that in case of lube system or other potential failure, aircraft control will not be disrupted. The design is based on Detroit Diesel Allison's extensive experience with industrial clutches, which have similar horsepower requirements, and with aircraft engine clutches, which operated at similar surface speeds.

The design of the clutch has been completed. Component testing has demonstrated the ability of the full-scale clutch plates to operate satisfactorily under full load conditions. Prototype flightweight clutches are being fabricated, and operational testing will start this summer.

As gearbox requirements for the various aircraft configurations become finalized, the development of the various spiral bevel gear

and bearing packages should begin. Each of these right angle drive sets could be designed, fabricated, and rig tested before the final propulsion system design is completed.

An early simulated propulsion test has been suggested as an excellent means of demonstrating the integrated propulsion system. A test of this type, early in the program, would serve to identify those areas where additional development effort was needed and, by the same token, point out other areas where little or no further work was required and thus save a substantial amount of time and money. This full-scale test should include enough of the system to permit demonstrations of power transfer, fan response rates, clutch engagements, and system control under various operating conditions. Existing components, such as the XT701 engines and the clutch from the current Navy development program, could be made available for the test. Other components could be obtained from company development programs in progress to minimize the need for additional special equipment.

SUMMARY

The Navy's subsonic, multimission V/STOL aircraft requires the closest integration of airframe and propulsion system of any new design in the author's experience. This is the result, in large part, of the expanded role of the propulsion system in that it provides not only propulsive force but also primary control of the aircraft in the vertical mode.

The development of the propulsion system is an active program that currently comprises aircraft configuration studies, program planning, and a development project for the front fan clutch. The development of other components is expected to follow in rapid order. Extensive design and trade studies, some of which are already in progress, will be required to ensure an optimum final design that best meets the highly diverse multimission requirements assigned to this new aircraft.

Beyond the Navy's well defined V/STOL requirements, there is a likelihood that a portion of the Army's future helicopter force will turn to an advanced V/STOL aircraft. Furthermore, a recent NASA study shows that V/STOL aircraft similar to those being developed by the Navy offer the possibility of a 10-year saving of $661 million over conventional helicopters for support of future offshore petroleum operations.[13] The possibility of V/STOL for city-center-to-city-center civil transport is an intriguing question for the future. [14]

Clearly, the Navy is stepping forward with a development program that is certain to change the character of Naval air operations, not to mention aircraft support ships. There is also the strong

possibility that this program is the first link in a chain of events that will significantly shape the future of aviation.

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DISCUSSION

CRAGIN: (General Dynamics)

I noticed in your time chart there, you left out drive systems and guidance and control systems.

BEAM:

I left out a lot of things!

CRAGIN:

Well, my question to you is, do you really feel that, in light of the materials and technology that you need in order to develop these high powered gear boxes and the complicated guidance systems that we are going to need between ships and airplanes, you can really get out operational by 1990?

BEAM:

Let me first talk about the gear boxes and drive systems. The designs that we have been working with, to date, have used state-of-the-art gears and shafting; we have used subcritical shafting, greased packed shaft bearings, and spiral bevel gears at stress levels comparable to what is being used in today's heavy helicopters. We do not see anything there that requires any breakthrough at all. It is pretty straightforward, simply scaling up to larger sizes. So, with respect to shafting and gears, I do not really see a problem. There will have to be some development work done to adapt to the particular system and size that is selected, but nothing really new in technology other than minimizing weight as we go along. With respect to control systems, yes, I think a lot of work has to be done there. We are going to have to integrate the propulsion system control system into the aircraft; we are going to have to provide electronic controls with triple redundancy.

CRAGIN:

But you think you can do all that and get it operational by 1990?

BEAM:

Yes, I think it can be done.

CRAGIN:

Okay, I have another question for you. You mentioned the IGV versus variable pitch fan being pretty comparable as far as thrust

response rate was concerned. How do you compare the two systems as far as fan efficiency and surge margin are concerned?

BEAM:

Our tests to date -- we have done subscale tests on the variable inlet and the guide vane, and there have been model tests done slightly subscale on the variable pitch -- show that the efficiencies are quite close. What was the other part of the question?

CRAGIN:

From a given operating point, what effect does the use of variable inlet guide vanes versus variable pitch have on the surge margin of the fan, speaking relative to the control power available?

BEAM:

We found that they are very close, very close. We are looking for thrust variations in the order of 20-25% with the same surge margin on each of them, comparing them on the same basis.

DELANY: (Rockwell International)

To carry on from the previous question a little further, between the variable pitch and the variable guide vanes, in a distorted flow field, do you think that will make any difference? In other words, your efficiency figures are all given assuming it is the same kind of inlet system, but we are talking about distorted systems; will that make any difference?

BEAM:

The lower aspect ratio of the variable pitch fan will be a plus for it with respect to inlet distortion. On the other hand, the variable vane fan will have higher speed and thus lower loading, which will be a plus for it. This is one of the reasons we feel the development of both fans should be carried through development testing before making a selection.

JONGENEEL: (Douglas Aircraft Company)

You mentioned the engine swallowing a lot of seawater. What about all the tiny cooling passages and so on down in the bowels of the engine, which are going to get full of deposits? Do you have a way of coping with that?

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MICROCOPY RESOLUTION TEST CHART
NATIONAL BUREAU OF STANDARDS-1963-A

BEAM:

We will use the centrifuge effect of the compressor to separate the water and take the clean cooling air off the inside of the flow path. That is fairly straightforward. Besides that we simply have to make those passages big enough so that they will not clog, and that is something we have had to deal with, with sand and dust and salt water, as in the past. It is nothing new, but it is going to be a very stringent problem.

JONGENEEL:

I think you are talking about an order of magnitude change.

BEAM:

I suspect we are.

DENNING:

I have a couple of questions. You said you had examined power transfer with variable IGV's as far as the control mode was concerned. In the engine fail case, have you gone down to a situation where you are at 50% of the design power?

BEAM:

Yes, we have. We can go below that. At 50 percent there is no problem.

DENNING:

In the gas generator section, there is always the possibility, when you have a clutch and a gear box, that in the event of an engine accidental disconnect under power, you might have a runaway situation on your power turbine if you have a free power turbine. Looking at our configuration, we have an IP compressor attached to the IP turbine shaft, which, of course, can run with blow off in a steady state condition where there is zero power off-take.

BEAM:

Yes, that is one way of handling it. We had that same situation in the Boeing heavy-lift helicopter, and we handled it there by providing sufficient integrity in the turbine wheels such that we would lose turbine blades before we lost the wheels. In fact, we put a centrifugal trigger in the blades so that they would come off before the wheels burst. There we were using sprag type clutches; here we are using a more reliable type of over-running clutch, a spline type. We are less likely to have a problem with this type

of clutch. But there is that inherent possibility, and it has to be dealt with somehow in the system so that you do not have catastrophic failures to those power turbine wheels.

BERNSTEIN: (Canadair Limited)

Whether you use variable pitch pads or variable IGV's for thrust control, these are obviously primary flight controls in a low speed regime. What requirements so you foresee for redundancy of these systems? Do you have to see the hydraulics duplicated and triplicated?

BEAM:

They have to be prime reliable. Yes, I would expect the hydraulics to be at least duplicated, possibly triplicated. Of course, the control will be at least triple. The designs we have done so far have had duplicate hydraulic actulators and triple redundant controls.

FLIGHT/PROPULSION CONTROL SYSTEMS FOR 1990 APPLICATIONS

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ABSTRACT

Basic research guidelines and a schedule for developing an integrated flight and propulsion control system with Initial Operational Capability (IOC) in 1995 are presented. The value and probability of success of the new ideas is discussed, utilizing the format suggested by the Stanford Research Institute. Some new ideas are discussed that apply to control hardware, software, and system integration that may fit into the predicted schedule of development. A key requirement is identified in each case for meeting the mission objectives in terms of quantified design criteria.

INTRODUCTION

Flight control and propulsion control technology have evolved in a relatively independent manner to satisfy the requirements of conventional aircraft systems. It is generally recognized that a successful verticle takeoff and landing (VTOL) system will require a more cooperative approach to develop effective flight/propulsion control coupling. The first step is mutual recognition of the time schedule for propulsion system development and its impact on control technology programs.

Based on current experience with aircraft engines having limited airframe integration, we can forecast that the design, development, and operational suitability testing of a highly coupled, VTOL aircraft propulsion system will take approximately 11 years. Figure 1 identifies

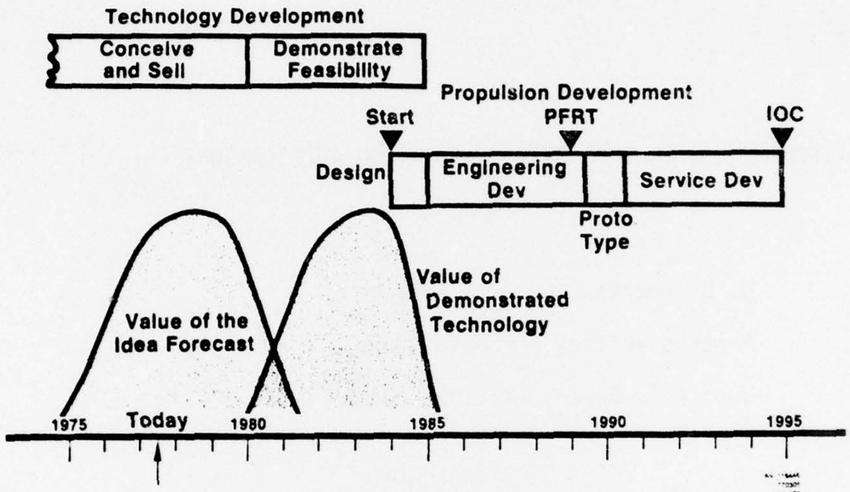

Figure 1. Technology Forecasting for Propulsion Controls

the schedule for the supporting technology, presented in a manner suggested by the Stanford Research Institute.

This schedule recognizes that, prior to design, approximately four years is required to demonstrate an advanced concept and develop the design tools required to use the concept with reasonable confidence. Preceding the demonstration, we find that it takes about four years to identify a new idea, relate it to a future program, and obtain the necessary support for demonstrating its feasibility.

The value of an idea and its successful demonstration is related to the start of propulsion system final design. An idea proposed too early may be inappropriate and, therefore, have relatively low value for the particular application. An idea demonstrated late has low value because incorporation implies redesign and supplementary development. Ideas presented today will probably have maximum value in a system scheduled for IOC in 1995. The technology for systems planned for 1990 IOC should be under test evaluation now.

CONTROLS TECHNOLOGY

The technologies supporting control system evolution draw from a variety of disciplines identified in Figure 2. While some of these disciplines are paced by progress within the aerospace community, most of them are now heavily influenced by the demands of the consumer industries.

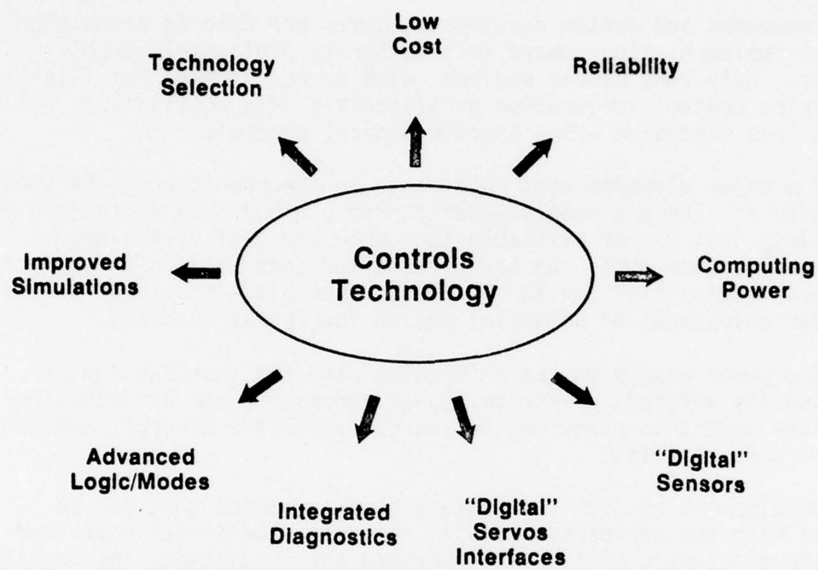

Figure 2. Controls Technology

As automatic controls become more commonplace in the consumer market, industrial research will focus more on that need and will respond less to the special needs of aerospace products. Although some consumer products and techniques will be adaptable to our needs, and we should be alert to this potential, the net effect will be a requirement to expend more research dollars for aerospace specialty items.

Research money alone will not, however, reverse the current trend of specialty industries to ignore or reject the aerospace market. Within these industries, there is a rare consensus between the "managers" and the "innovators" that aerospace products are not worth the trouble. In addition to a low profit margin, the managers see a poor return on the investment of time and limited innovative talent. This reinforces their natural desire to constrain the innovations and react only to the consumer market. The innovators are not stimulated because long range military missions, plans, and products are not visible to them. In addition, their novel or revolutionary ideas are frequently hindered by Military Specifications.

There are many other factors involved in this problem and a solution is not obvious. Research effort should be expended to define new planning, budgeting, and procurement procedures plus new technology management methods that will encourage these specialty item subcontractors to participate in aerospace product development.

Component and system development costs are driving propulsion control implementation toward full authority digital electronic systems. Only very simple engines with no requirement for flight/propulsion control integration or diagnostic data acquisition will remain less expensive using hydromechanical controls.

The major elements that constitute an electronic unit are shown in Figure 3. These elements benefit from a broad base of fabrication technology that is not available to reduce the cost of the hydromechanical counterpart. We have identified some additional supporting technologies that may further reduce the electronic control cost with our assessment of potential design incorporation dates.

The power supply may be eliminated with the introduction of high quality aircraft electrical power generation and distribution. We assume that this power may be shared by all the aircraft and propulsion control units.

Development of high temperature power switches that can be located with the servoactuator will remove a substantial heat load from the electronic unit. With improved box insulation, the cooling problem will be reduced to removing the internal heat generated by the electronic components.

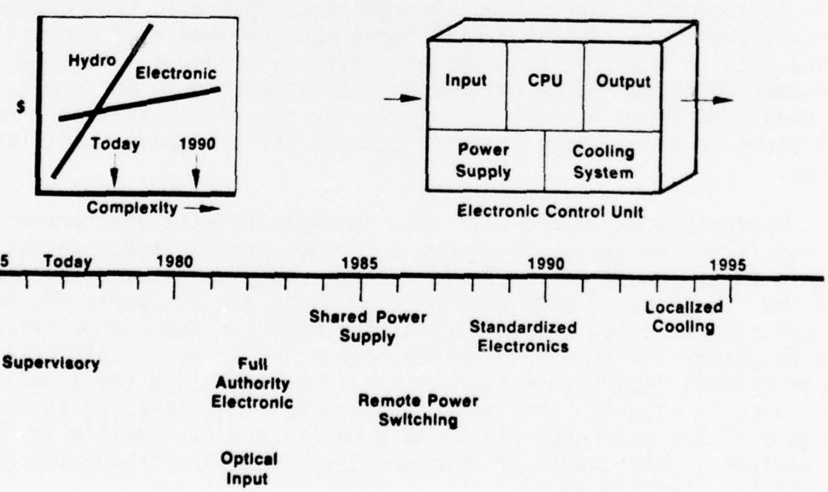

Figure 3. Low Cost.

Local, integrated chip cooling is suggested in Figure 4, wherein the base material of the chip is fabricated to include a thermoelectric cooling junction. The other junction is located outside the box in a convenient part of the fuel system where the excess heat can be safely rejected. While thermoelectric coolers are currently inefficient, the total amount of heat being transferred is small and a practical design should evolve.

By cooling and stabilizing the junction temperatures, the electronic component reliability could be improved. Eliminating the internal thermal planes and air cooled or fuel cooled heat exchangers will allow increased box packing density, and a smaller box will further reduce the external heat load.

Electronic controls today are structured around a multi-chip processor. The cost of the processor will continue to drop as more complex architecture and instruction sets are included on each chip. A dramatic improvement in reliability should also follow with development of a single-chip microprocessor and the associated reduction in external circuit connections. The desirability of using a microprocessor with a large market potential is understood; however, for propulsion control, the ideal microprocessor would contain the elements below:

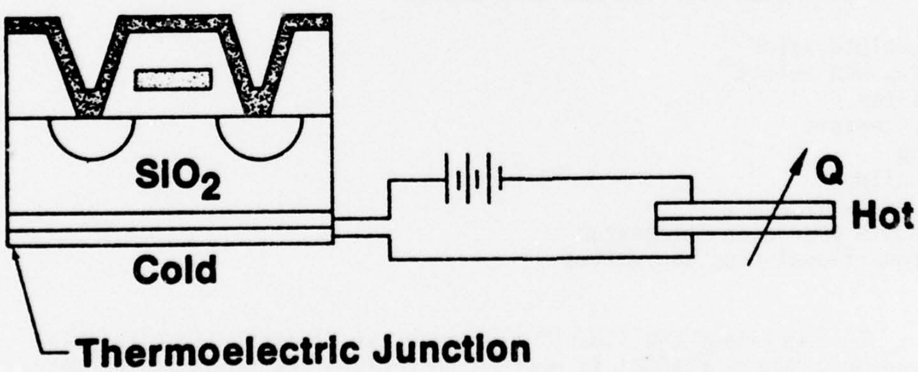

Figure 4. Localized Cooling

Architecture

16 bit words
8 general purpose registers
Register-register arithmetic
Indirect and indexed addressing
Multiple level priority interrupts
Stack system for temporary storage
Floating point arithmetic
Direct memory access
Shared memory capability

Instruction Set (50 instructions)

Load, store
Add, subtract, multiply, divide
Shifts - long, short, left, right, arithmetic, logical, rotate
Tests - Value to value, zero, ± full-scale, overflow
Increments
Logicals - and, or, etc.
Double precision add
Jumps with program counter save

A "super microprocessor" would include the additional instructions:

Arithmetic with overflow and zero protect

Absolute value
Min, Max select
Filter
Hysterisis
Log
Antilog
Curve search
Double precision integrator
Proportional plus derivative

To complement the "ideal" processor, an Electrically Alterable Read Only Memory (EAROM) is desired to replace the Programable Read Only Memory (PROMS) currently in use. This technology will have a substantial impact in expediting and reducing the cost of control development. Flexibility to accommodate propulsion system changes with minimum reliability risk will be possible.

The characteristics of this EAROM should be:

Cycle time (word access) 100 nanoseconds
*Word organization 16 bit X 1000 words
Reads before refresh 10^{13} (20,000 hr)

These microprocessors and memory elements will be an overkill for many subsystem control functions; however, the tendency to use special purpose designs will be reduced if the capacity is available in the microprocessor chip. More complex subsystem controls should be designed to use parallel microprosessors, as shown in Figure 5, to maintain system compatibility.

Digital computers are compatible with binary, pulse rate, or pulse width output signals. The inherent precision of the computer is compromised by analog output transducers. Contemporary electro-hydraulic servovalves utilize electromagnetic transducers between the electronic control and the load actuator. Regulation of these servovalves by pulse rate or pulse width signals is limited by the circuit inductance at high frequency. An alternative approach is suggested in Figure 6.

This concept shows a piezoelectric stack used as the interface transducer that appears as a capacitive load to the output power

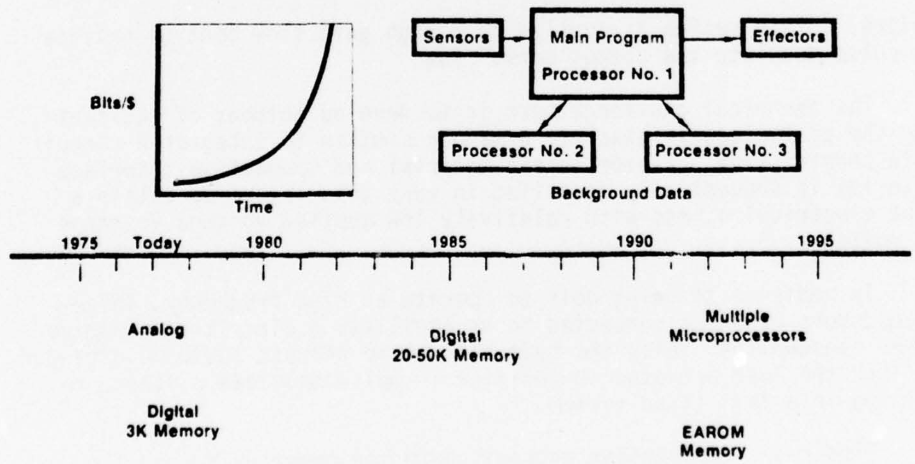

Figure 5. More Computing Power

*(More storage capacity per chip at equal power)

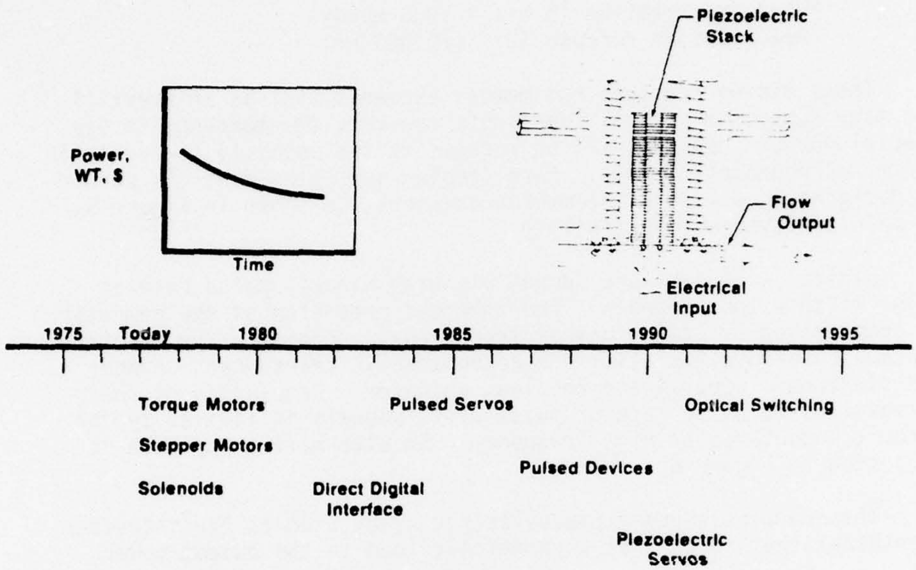

Figure 6. "Digital" Servos

switch. Stack motion is coupled to a high gain flow control orifice to pulse modulate the output servo flow.

The technical challenge here is to develop methods of fabricating the piezoelectric stack in a manner similar to integrated circuit chip construction. Piezoelectric material and conductive interface material is sequentially deposited in very thin layers to obtain a high electrical stress with relatively low applied voltage (perhaps 28 volts).

In addition to being able to operate at high frequency, these transducers might be connected in an oscillating circuit to minimize power consumption. Also the hydraulic servo circuit could be arranged to lock the load actuator in position if pulse modulation stops, resulting in a fail fixed system.

Electronic computation presents both the opportunity and the capacity to interface with a wide variety of transducers; however, classical control design approaches have constrained our logic to that which can use available sensors. Current investigations of the application of modern multivariable control theory are giving us better insight into the benefits of sensing some state variables in the propulsion system that were previously not considered.

While the analysis is incomplete and we have no specific sensor technology to suggest today, this general observation can be made. The new transducers should fundamentally produce a high frequency signal output that varies with the sensed parameter. The "solid state optical" transducer, identified in Figure 7, is visualized as one whose optical transmission frequency is changed by the engine state variable and quantified by the computer input logic.

The first engineering problem for control system design is adequate system definition. Economical analysis and development of flight/propulsion coupled controls will require cooperative team effort between propulsion, airframe and control component suppliers. Comprehensive studies of the VTOL control options will focus on system dynamics, and nonlinear, dynamic models will be required to evaluate design alternatives. Figure 8 identifies the interactive studies required of the team members.

A nonlinear dynamic deck, programmed in "FORTRAN" for use on contemporary general purpose computers with adequate fidelity for VTOL control design, will be expensive to use. We estimate that on our time-shared computer, 2 minutes of real time VTOL transient operation will be "in residence" for approximately 5 eight-hour shifts. A better technique is needed.

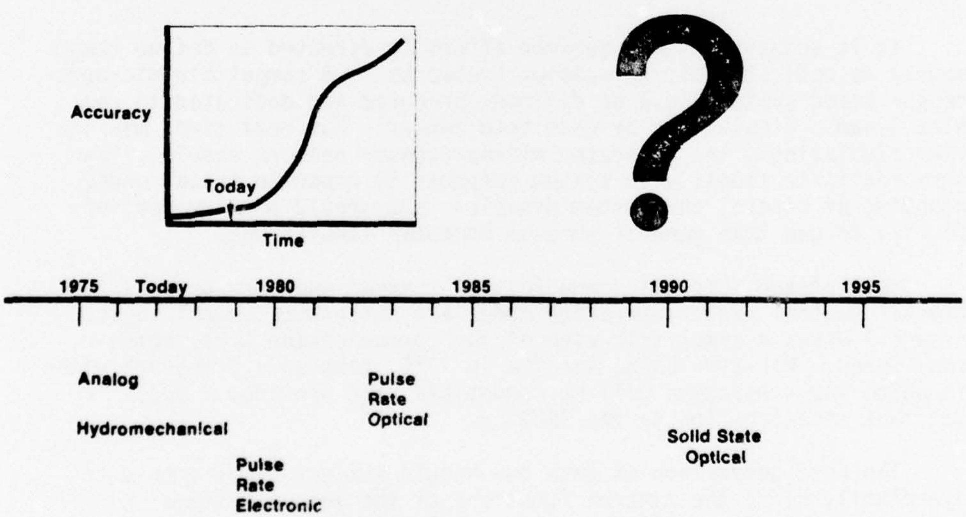

Figure 7. "Digital" Sensors

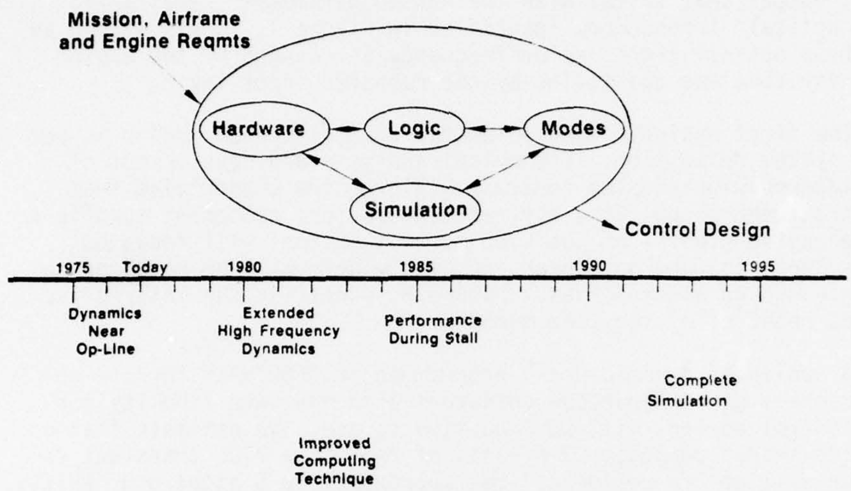

Figure 8. Improved Simulations

It is suggested that research effort be directed to define these models as dedicated microprocessor "networks." A compatible microprocessor based system could be defined, procured and dedicated to complex dynamic simulations by each team member. For real time, man-in-loop simulations, the dedicated microprocessor network should allow more realistic fidelity in system response to expedite mutual understanding of control and system dynamics and should be more cost effective to use than general purpose computer simulations.

Multiplexed, digital communication systems open new options for overall control system operating modes and subsystem control logic. Figure 9 gives a simplistic view of the communication links being considered. MIL-STD-1553A, adopted in 1975, sets up a framework within which the subsystems will be compatible and provides a model for data bus specification in the 1990s.

The next generation of data bus should reflect requirements to dynamically blend the control functions of the weapon system. A typical example for V/STOL operation would tie the radar, flight control, and propulsion control together to complete a landing with minimum power excursions. Another example would couple flight control, propulsion control, and laser tracker to the weapon fire control and optimize aiming precision or target range.

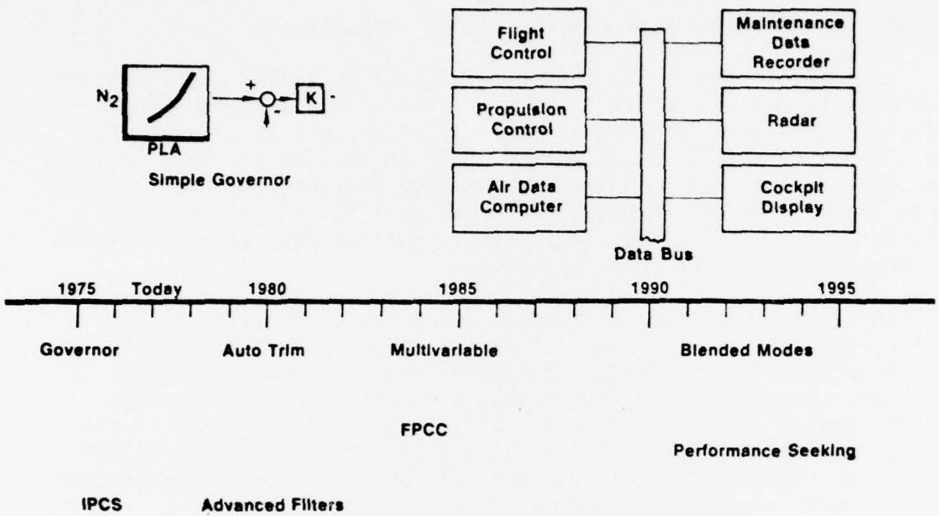

Figure 9. Advanced Logic/Modes

Performance seeking control actions could be supervised by the mission control system. Algorithms stored here could be selected to maximize range, minimize time to target, or maximize flight time. Contributing subsystems (flight control, propulsion control, inertial navigation, and air data computer) could optimize their instantaneous performance while simultaneously observing subsystem limits.

Research to define these blended control modes will require cooperative "team" studies to assure that each subsystem is modeled with adequate fidelity. The studies should derive data transmission rates that support the control performance objectives and will probably indicate optical data transmission is needed for speed and noise immunity.

The major obstacle to universal acceptance of electronic systems has been their relatively high failure rate while operating under severe environmental stress. Today's short haul aircraft use relatively simple engines that are controlled by hydromechanical systems. These systems are currently demonstrating higher reliability than can be expected from electronic systems of equal complexity. As these engines become more complex to satisfy demands for improved performance and response, their control complexity will approach that of current high performance fighter aircraft engines. Experience in controlling these engines shows that electronic systems accommodate the added complexity with less degradation in reliability and, with proper development, can

exceed the reliability of hydromechanical systems.

As shown in Figure 10, both control designs suffer a reliability loss with increased complexity and, unless failure accommodation techniques are used, mission reliability and flight safety will suffer. Conventional aircraft, particularly those with multiple engines, can tolerate propulsion control performance degradation, and for these installations single channel, "fail graceful" electronic controls will be acceptable. Single engine installations may require a simple electronic or hydromechanical backup.

For V/STOL installations, where the propulsion system is part of the aircraft primary flight control, failure accommodation will be more critical, demanding "fail operational" system capability. It is not reasonable to forecast that a single channel electronic control system will have adequate reliability to satisfy the flight safety requirements of V/STOL operations by 1995. Redundant designs similar to those used for flight controls will be necessary, increasing the parts count, cost, and maintenance requirements.

To minimize the impact of propulsion control redundancy on life cycle cost, research effort should be directed to increase the rate at which electronic system reliability improves. Operational experience with electronic controls will accumulate rapidly in the 1980

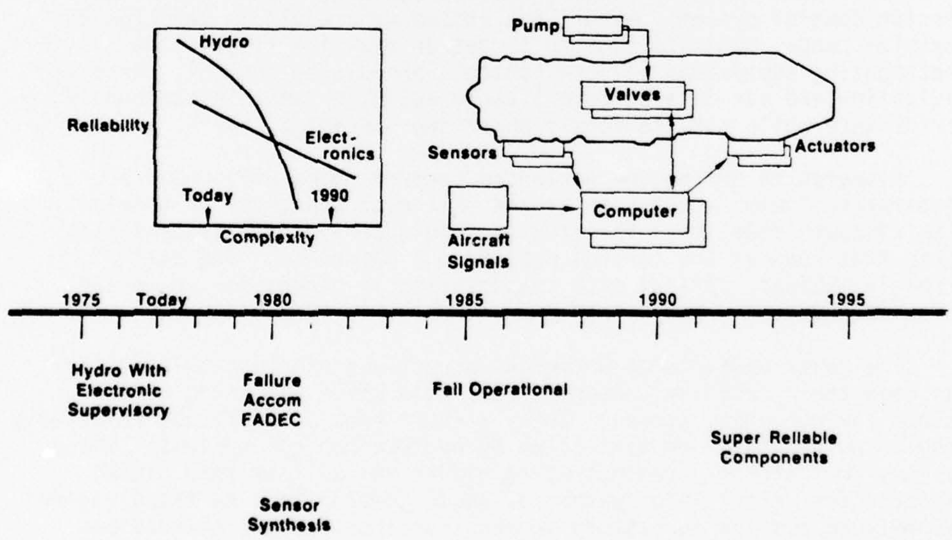

Figure 10. High Reliability

time period and a plan should be formulated to make full use of this experience. The plan should lead to effective failure detection, analysis, information distribution, corrective action, and followup.

Definition of methods for collecting, correlating, and distributing service information may not be as exciting as the invention of solid state electronics, but it is critical to reducing their cost and developing confidence in their use. A comprehensive plan should be formulated now to assure there are no missing links in the data collection and feedback system.

Simultaneous introduction of electronic propulsion controls and on-board fault diagnosis systems will complement the reliability growth plan. Figure 11 shows the electronic control used as the primary data collection interface for the diagnostic system. It is the logical component to do the on-line, real time data correlation because most of the data and diagnostic algorithms necessary for proper control are already there. Error signals, system dynamic response to internal and external stimuli, and system static performance relative to the operating environment are normally available within the control logic.

Integration of the diagnostic data acquisition with the control function and developing it as one system will go a long way toward

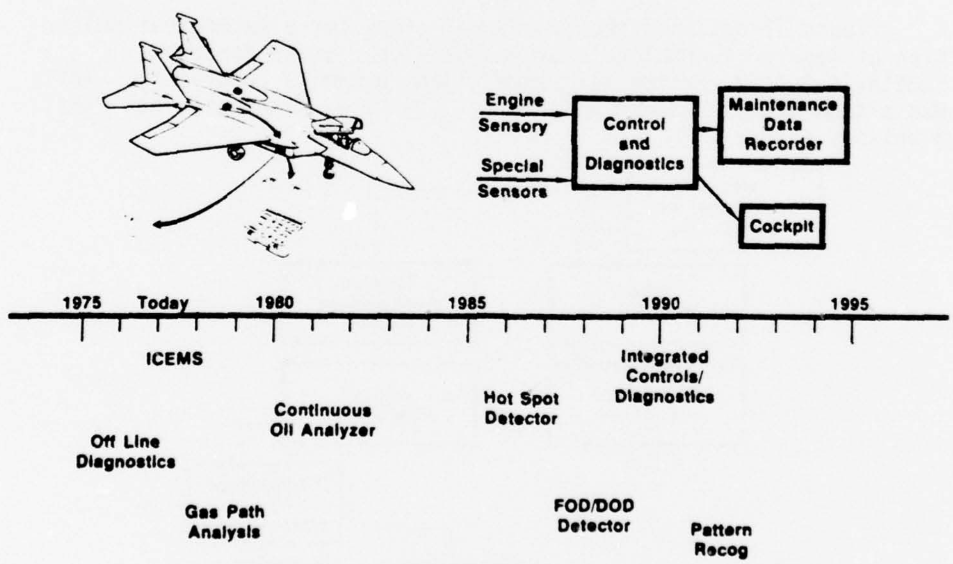

Figure 11. Diagnostics

improving confidence in the diagnostic data and eliminating component removal without cause. The special sensors and signal conditioning methods will be exposed to the entire development sequence and will provide a good link between observations made during development and subsequent in-service observations.

Research programs should address the problem of extracting fault patterns from data the diagnostic system will provide. This will require developing efficient "system identification" algorithms that can characterize a good propulsion system, methods of storing the identified model, and methods of relating subsequent pattern changes in probable faults.

An attempt has been made here to identify some of the control technology that can support the development and perhaps enhance the operational suitability of a V/STOL type aircraft. I hope it is clear that the long lead time for propulsion development makes it necessary to identify and demonstrate this technology very early.

We could provide more effective guidance for research programs if we had better visibility to future mission plans and objectives. While security problems present some difficulty, they can be handled, and the long range benefits are worth the effort.

Figure 12 outlines the procedural steps for a methodical selection of applied technology that was developed under contract F33615-74-C-3042 for the Air Force Flight Dynamics Laboratory. This was a team effort between A.F.F.D.L., Lockheed, Honeywell, and Pratt & Whitney Aircraft.

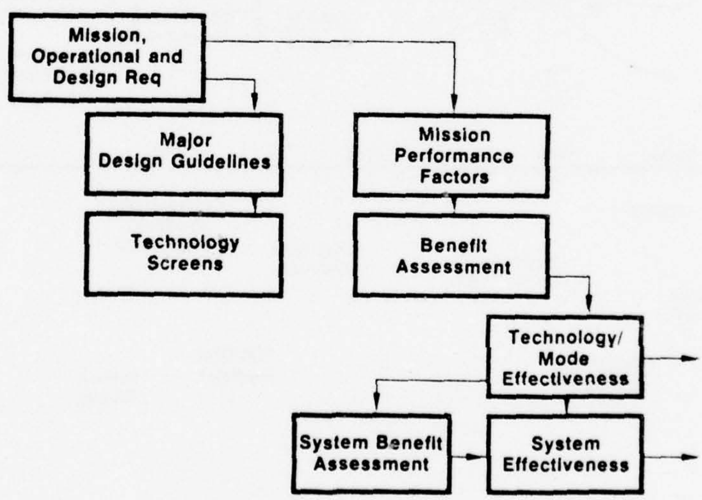

Figure 12. Methodical Technology Selection

This procedure is intended to quantify the relative impact of technology on the cost/effectiveness of a given system. In reality, it is a computerized bookkeeping system that allows rapid comparison of technology payoff.

The procedure does not perform the required studies; it simply identifies what must be done. The first three blocks of information are the ones most difficult to fill. In a sample case we studied, technology screening, benefit assessment, and calculation of cost effectiveness required substantial analysis, but the work was straightforward; however, obtaining realistic mission operational requirements and relative performance weighting factors was a problem, and this information should be made available if future technology forecasts are to be more accurate.

DISCUSSION

KEMPER: (Vought Corporation)

You have made an excellent presentation of the problems of getting where you think we ought to be, particularly with the control requirements which are five years too late based on current schedules on Type A for V/STOL. Most of your stuff comes home in the 1990s. Now is this an idealized situation you are presenting?

EMERSON:

Yes. What I tried to address was the schedule that I know to be real and if I was asked, which I was, to present ideas that are in need of research. I am not trying to identify that they are applicable to the Type A V/STOL or applicable to the Type B V/STOL, because I really don't have that firm a grasp of the schedules for those missions right now today. We are putting in an RFI response for one--everybody is--but I am not sure that schedule is going to hold, and I don't know that anybody is. Furthermore, I just ignored that (V/STOL-A Schedule) and backed out what I know to be the necessary time for demonstrating and selling an idea, so that if we have the ideas today, they would see initial operation in 1995.

KEMPER:

That was very good. That was an excellent way to present it.

PROPULSION RESEARCH REQUIREMENTS FOR POWERED LIFT AIRCRAFT

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ABSTRACT

This paper suggests research areas offering the greatest return for a research investment in terms of our ability to achieve optimum aircraft configurations. Emphasis is on improving the means for examining potential design concepts. Research areas which offer the largest potential payoff are identified as numerical fluid mechanics, testing methods, and propulsion simulation and control. Detailed research objectives in each of these areas are suggested, and the use of the resulting technology to improve the design process is discussed.

LIST OF ABBREVIATIONS AND SYMBOLS

<u>Symbol</u>

B.L.	Boundary layer
C.R.	Contraction ratio
CTOL	Conventional takeoff and landing
DAIS	Digital avionics information system
EPCS	Electronic propulsion control system
FADEC	Full authority digital engine control
IPCS	Integrated propulsion control system
LCC	Life cycle cost
LDV	Laser Doppler velocimeter
L/D	Length/diameter
Mth	Theoretical one-dimensional Mach number
MTBF	Mean time between failures
N-S	Navier-Stokes
P_A	Ambient static pressure

P_{tF}	Fan stream total pressure
P_{tP}	Primary stream total pressure
R	Radius
R_o	Duct radius
S.O.A.	State-of-the-art
STOL	Short takeoff and landing
T_t	Total temperature
T_{tF}	Fan stream total temperature
T_{tP}	Primary stream total temperature
V	Velocity
VAXIAL	Axial velocity component
VIP	Ideal primary velocity
V/STOL	Vertical/Short takeoff and landing
VTOL	Vertical takeoff and landing
W	Velocity component in the streamwise direction
α	Angle-of-attack

INTRODUCTION

Several aspects of propulsion system integration for new STOL and V/STOL aircraft need increased emphasis if we, the technical community, are to achieve anything close to optimum aircraft configurations in the next decade. For example, assume that one has a new airplane concept, of great promise, which must be evaluated. How is this evaluation to be accomplished without building and flying an entire airplane or without building and testing a large number of scale models and then overcoming a difficult scaling problem? Experience has shown that all too often, particularly in V/STOL aircraft, a concept will look great on paper; but, as the hardware evolves, performance is eroded, and weight creeps up until the usefulness of the aircraft has disappeared and the concept is dropped.

The purpose of this paper is to suggest research areas offering the greatest return for a research investment in terms of our ability to achieve optimum configurations. The system aspects of the problem will not be addressed. Research objectives must be defined that are general and not tied to a specific aircraft concept and that will significantly contribute to our ability to design better V/STOL aircraft. The actual concept is important, but we are discussing the path to define the best concept and, finally, the best configuration. Emphasis must therefore be concentrated on providing a means for examining potential concepts other than through expensive testing, which has been shown in the past to be misleading and which has a good probability of missing the "best" concept.

One difficulty is that propulsion integration for V/STOL aircraft is more complex than for CTOL aircraft. This increased complexity, following present design practices, necessitates very extensive parametric model and full scale tests to evolve a final practical design.

For example, a typical CTOL airplane with pod mounted nacelles is shown in Figure 1. The nacelle is usually placed away from the flowfield of the wing to avoid airplane propulsion system interactions; therefore, the flow through and around the nacelle is reasonably well behaved and understood. Contrast this installation with those of the YC-14 STOL demonstrator, Figure 2, a V/STOL aircraft with rotating nacelles and nose fan, Figure 3, and a highly integrated STOL transport, Figure 4. These powered lift aircraft differ from conventional aircraft in four important areas.

1. They are generally propulsion dominated systems; therefore engine integration is a major concern.

2. Wing aerodynamics are strongly affected by the propulsion system.

3. Inlets and nozzles are required to operate effectively over a much broader envelope than is necessary for CTOL aircraft.

4. Weight of the propulsion system is of critical importance.

A common characteristic of most V/STOL aircraft is a strong interaction between the propulsion system and aircraft flowfields. These interacting flows are in general very complex and three-dimensional. Present design procedure is based on an experimental simulation of these flows and the creation of an empirical data base through parametric testing. It is our belief that analysis can replace testing, to some extent at least, in the design process. However, our ability to analyze these flows needs to be improved. Another important characteristic is that a significant fraction of the aircraft's lift and control is provided by the propulsion system in low speed flight through thrust management.

Three research areas which we suggest would offer a large return on investment in terms of our ability to achieve optimum aircraft configurations are as follows:

1. Numerical fluid mechanics

2. Improved testing methods

3. Propulsion simulation and control

In the sections which follow, we suggest research in each of these areas and discuss how technology resulting from this be used to improve the design process.

Figure 1. Typical CTOL Airplane

Figure 2. YC-14 STOL Demonstrator

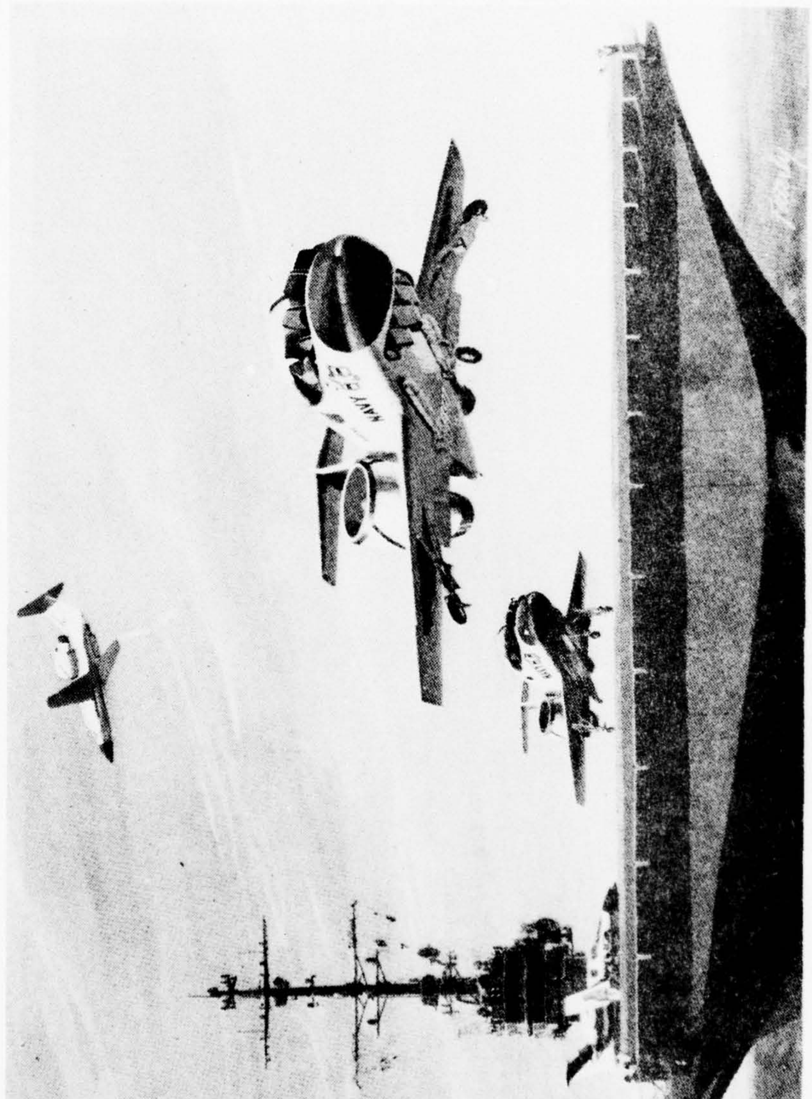


Figure 3. V/STOL Airplane with Rotating Engines

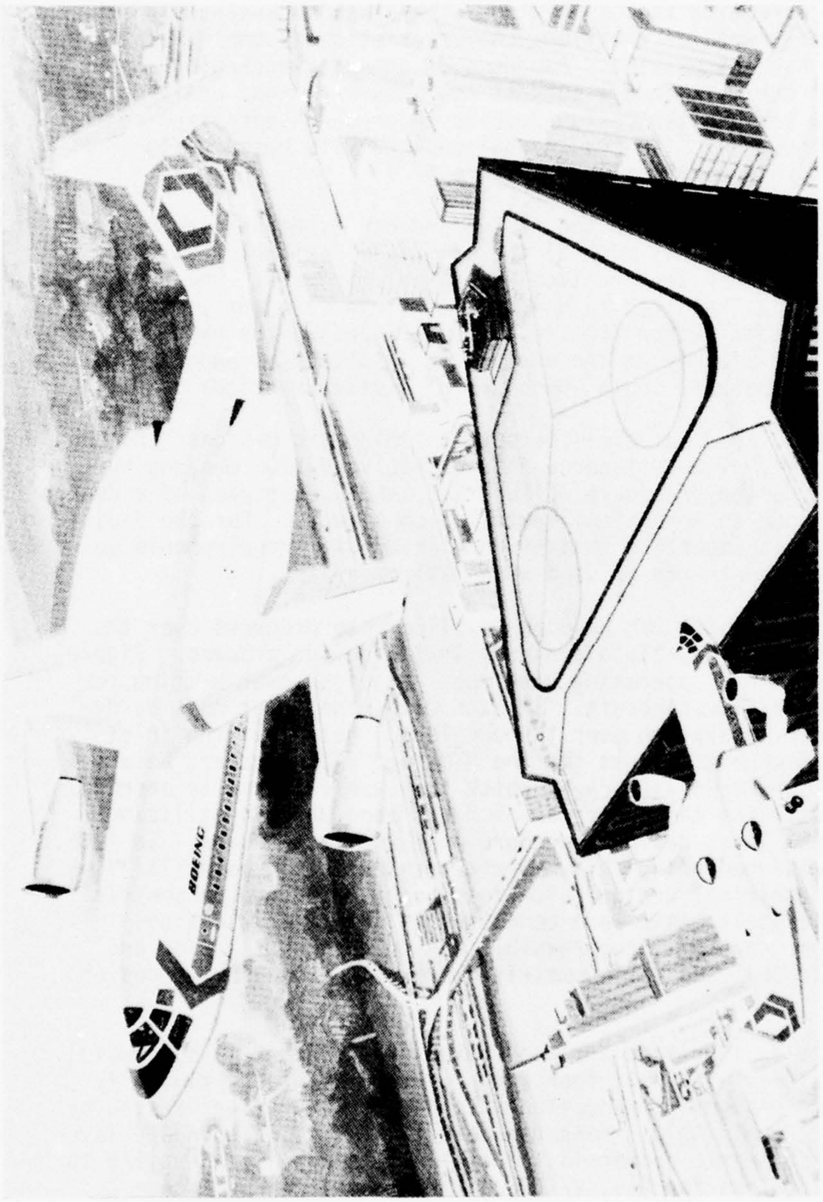


Figure 4. STOL Transport Airplane

NUMERICAL FLUID MECHANICS

Analysis has traditionally been used as an aid to testing, not as a primary screening tool. This was natural, since analysis which could accurately predict the flows characteristic of complex V/STOL installation has not existed. Advances in computer technology and analysis over the last decade suggest now, however, that analysis could replace test in the design process to an ever-increasing degree over the next decade. The traditional and analysis based design processes are shown schematically in Figure 5.

Use of flow analysis in the design process is emphasized because experience has shown that much of the presently available analysis is not being utilized in the practical design of hardware. The highly specialized skills required to develop the fluid flow analysis usually precluded that same person from really understanding the needs of the hardware design in applying the analysis. This lack of understanding often leads to analysis tools which are of little practical value.

My comments are thus aimed at both the aircraft designer and the fluid dynamicist. The designer wants innovative V/STOL designs but, using current design procedure, often finds that advantages of a design may slip away due to nonoptimum installation effects. For the fluid dynamicist, it is important that he understand user requirements so that new developments can be used with maximum effect.

The subsonic inlet can be used to illustrate progress over the last decade in applying fluid analysis in the design process. Figure 6 shows typical inlet operating envelopes at high power setting for CTOL, STOL and V/STOL,aircraft. In each case, the inlet must be designed to avoid separation over the envelope. Note that the inlet design task is more difficult for the STOL and VTOL aircraft because of the higher angles-of-attack at which they are expected to operate. Figure 7 illustrates the progress which has been made in utilizing analysis in the inlet design procedure over the last decade. In 1965, inlets were designed using axisymmetric incompressible potential flow analysis and empirical design guidelines; very extensive parametric model and full scale tests were conducted to arrive at the final inlet configurations. By 1973, compressible subsonic potential flow and boundary layer analyses (axisymmetric) were being used in the design process.

Even with an increased angle-of-attack requirement, a successful inlet was designed with only four wind tunnel model tests to verify and refine the design -- a reduction in model configurations tested by a factor of ten. In 1976, transonic potential flow and boundary layer analyses (axisymmetric geometries at angle-of-attack) were applied to the design of inlets for Navy V/STOL aircraft as shown in Figure 8. It is clear that the design of this inlet using 1965 capability would

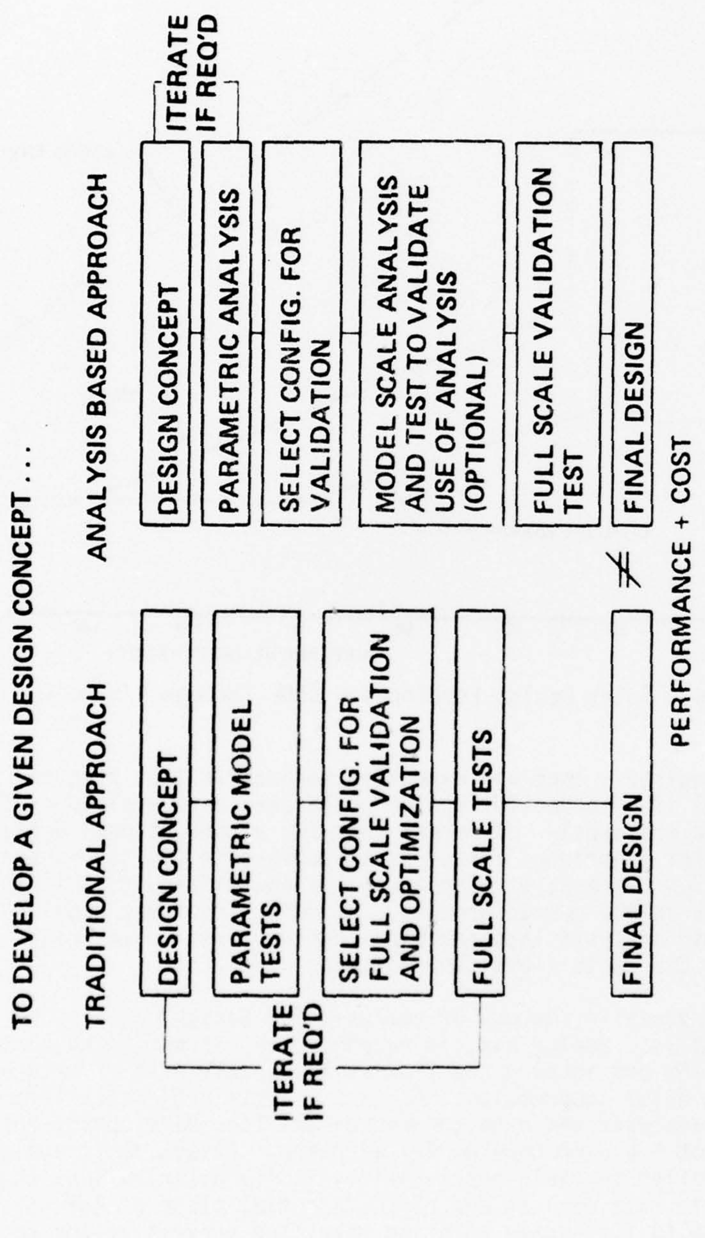


Figure 5. Design Procedures for Propulsion Installations

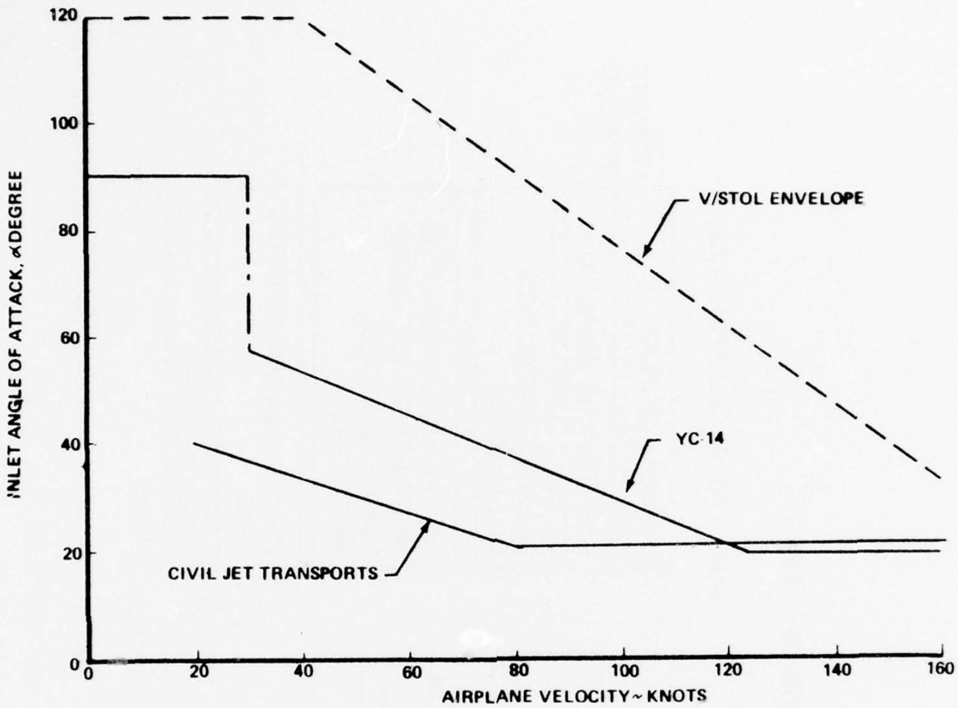


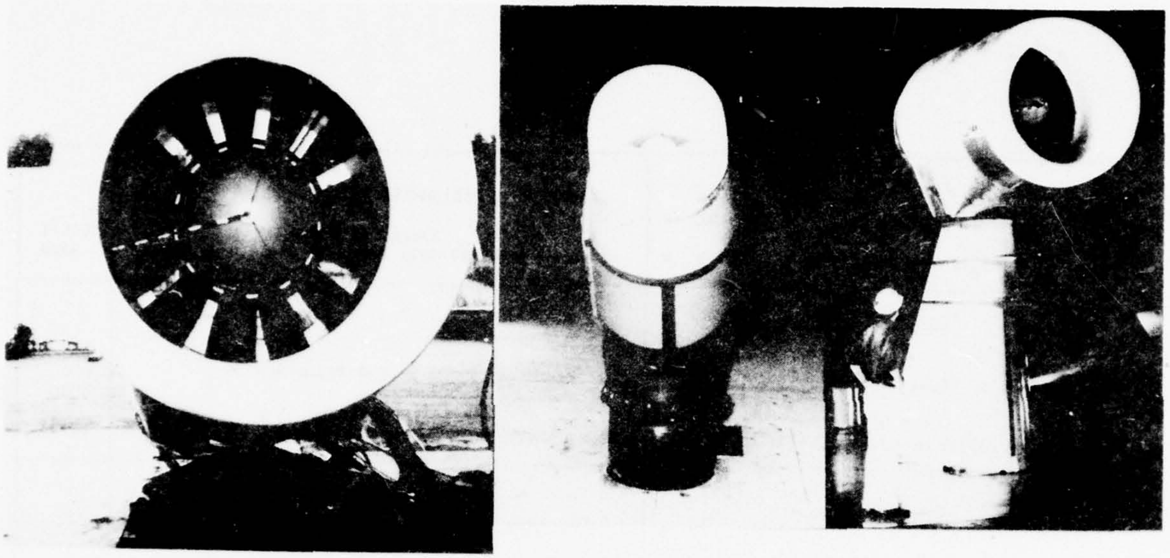
Figure 6. Inlet Design Envelopes - CTOL Through V/STOL

have required a long and expensive parametric test program. It is doubtful if a successful design would ever have evolved, as the lines required are subtly different than for a conventional design. While this inlet experience illustrates progress in the use of analysis in the design process, it also taught us how difficult it is to apply analysis in the design process. As much effort was required to make the fluid analysis into practical design tools as was required to develop the basic fluid flow computer codes.

Progress in the use of analysis for design can also be illustrated for nozzles. Boeing has, in recent years, investigated various devices for mixing and shaping the flow at the nozzle exit of a turbofan engine for jet noise suppression. As part of this project, axisymmetric mixing analysis was used to predict the flow development through the nozzle of the JT8D powered 727 airplane. Axisymmetric analysis, however, failed to yield nozzle exit velocity distributions that agreed well with test data at one circumferential plane at the nozzle exit. This led to full-scale rotating rake flow surveys at the nozzle exit

A/P PROGRAM		INLET DEVELOPMENT PROGRAM	
MODEL	TECHNOL. DATE	ANALYTICAL DESIGN METHOD	DEVELOPMENT TESTS
747	1965	• ONE-DIMENSIONAL, VISCOUS FLOW. • AXISYMMETRIC POTENTIAL FLOW (INCOMPRESSIBLE) • EMPIRICAL DESIGN RULES: L/D, M_{TH}, C.R., LIP SHAPE, ETC.	• EXTENSIVE PARAMETRIC MODEL TEST • FULL-SCALE BOILER-PLATE TESTS • A/P GROUND AND FLIGHT TESTS
YC-14	1973	• SUBSONIC POTENTIAL FLOW COUPLED WITH B.L. PROGRAM • EMPIRICAL DESIGN RULES	• 4 INLET MODELS TESTED TO SELECT CONTRACTION RATIO (C.R.) • FULL-SCALE GROUND RIG TESTS (FLIGHT HARDWARE) • A/P GROUND AND FLIGHT TESTS
NAVY V/STOL	1976	• 3-D POTENTIAL FLOW AXISYMMETRIC AT ANGLE-OF-ATTACK • TRANSONIC AXISYMMETRIC FLOW COUPLED WITH B.L.	• 3 MODELS TESTED TO REFINE DESIGN • FULL-SCALE TEST WITH Q-FAN IN AMES 40 x 80 WIND TUNNEL

Figure 7: Inlet Development Paced by Analytical Capability



ADVANCED ASYMMETRIC INLET DESIGNED FOR HIGH ANGLE-OF-ATTACK CAPABILITY

INLET PERFORMANCE VERIFIED ON 1/4 SCALE INLET MODEL IN 9' X 9' WIND TUNNEL.

FULL SCALE TESTING IN 40' X 80' TUNNEL

ACCURATE ANALYTICAL PREDICTIONS OF INLET FLOW SEPARATION DEMONSTRATED ON BOTH 1/4 SCALE AND FULL SCALE INLETS.

Figure 8. V/STOL Inlet Development

which showed a strong flow asymmetry not expected as a result of the model test program. As illustrated in Figure 9, 3-D analysis was then used to predict the flow development through the nozzle. Note that the analysis successfully predicted the asymmetry of the flow at the nozzle exit and explained the asymmetry as an interaction between the swirling primary flow and the turbine support struts. The turbine supports struts are shown in Figure 10 in a photograph of the aft end of the engine, up the nozzle. Although extensive model scale tests had been conducted, the swirl of the primary flow in the nozzle was not simulated in these tests, and the asymmetry of the nozzle exit flow was not present model scale. At the beginning of a test program, one may not understand which elements may have a major influence on the flow development. It is typically difficult to justify a sufficiently complex model test program to accurately simulate the full scale flow. Once the importance of the primary swirl to the downstream flow development was understood, it could be accounted for in subsequent experimental and analytical investigations in support of the jet suppressor development project.

As part of the Boeing investigation of jet noise suppression, the 3-D viscous analysis (1) was used to conduct a parametric study to identify critical geometric design parameters for a lobed internal mixing device, as shown in Figures 11 and 12. While the analytical simulation of the full-scale flow shows some need for further analysis development, analytical results were at least as good as a model scale test in the full scale flow. The analysis also yielded a good simulation of the model scale flow (not shown).

The cost to prepare the input data and to run the analysis was \$500 to \$1,000 -- the cost of the model scale test was \$30K to \$50K. These results strongly suggest the trend toward the analysis based design process, illustrated earlier in Figure 5.

Fluid Flow Problems of Powered Lift Aircraft

In discussing the application of numerical methods to the design of powered lift aircraft, I will use as an example a USB aircraft because this is a current activity at Boeing. It is also appropriate because it includes many of the problems which must be solved in predicting the flow field associated with powered lift aircraft. The flow field (Fig. 13) can be divided into two main parts, one dominated by viscous effect and one largely inviscid. The viscous region is particularly difficult when compared with that of a conventional aircraft. Starting from inside the tailpipe at the engine turbine exit, although geometry is axisymmetric, the flow is highly turbulent and strongly 3-D. This is due in part to residual swirl in the flow exiting the turbine and in part to the interaction between the flow and engine mounting struts. As the flow moves downstream the hot primary stream starts to mix with the cooler fan stream. This flow is then forced

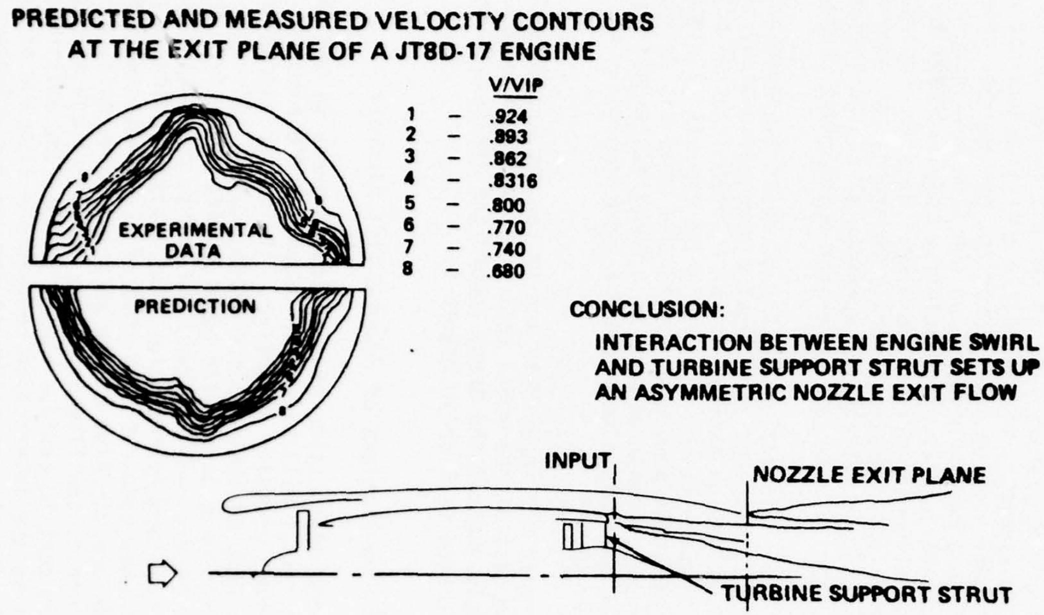


Figure 9. Examples of 3-D Mixing Analysis for Confluent Fan Engine Nozzle Flows

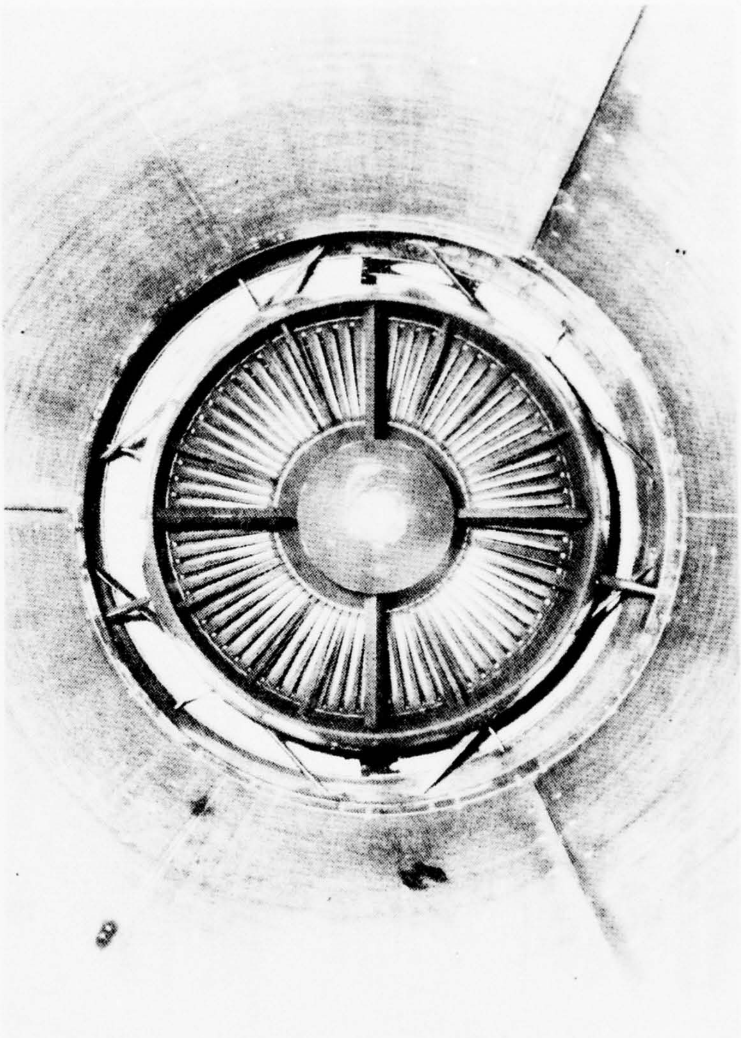


Figure 10. Aft End of the Engine

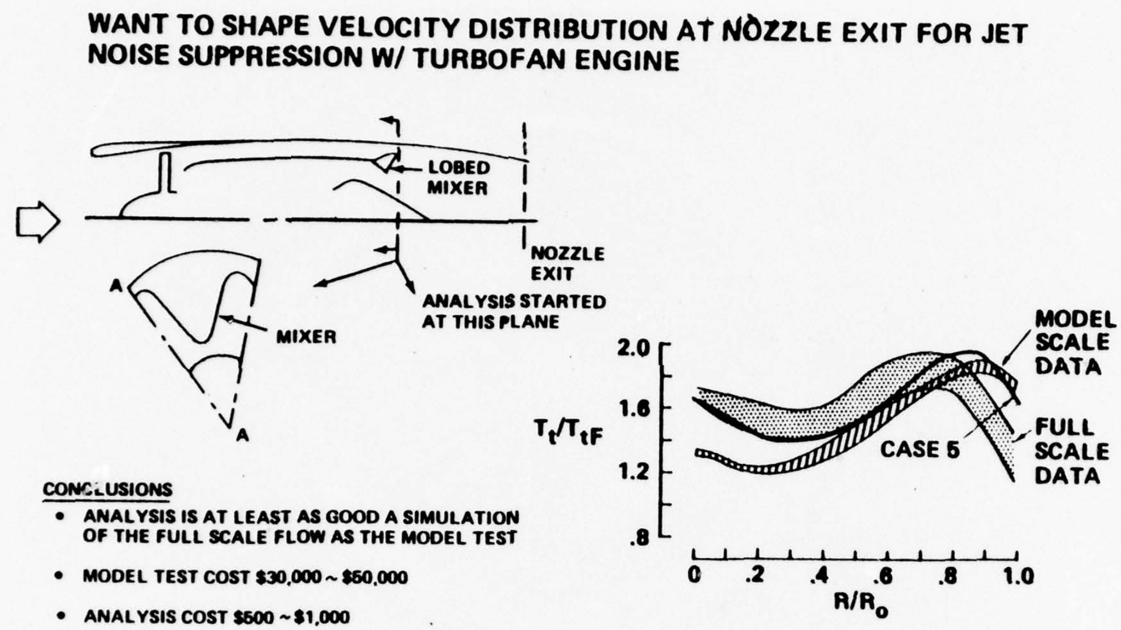


Figure 11. Example · 3-D Analysis for Lobed Mixers

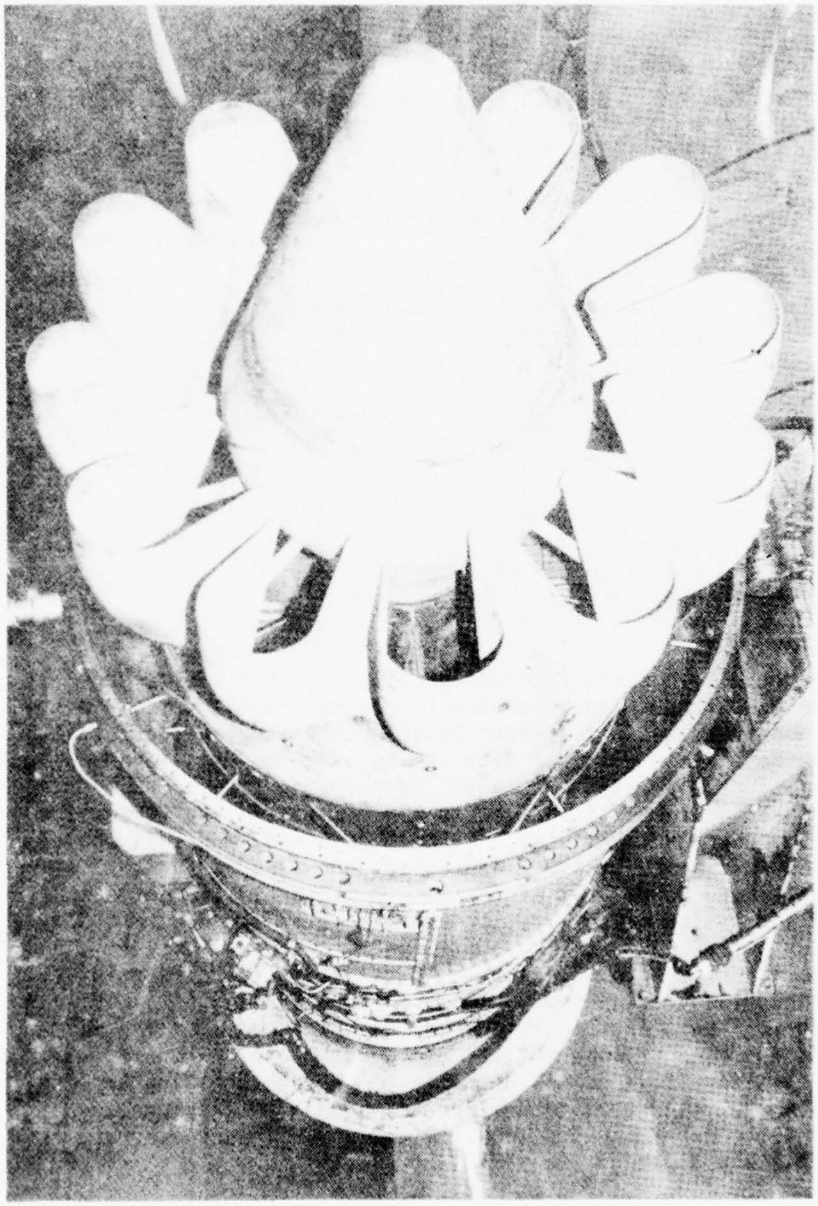


Figure 12. Full Scale Forced Mixer

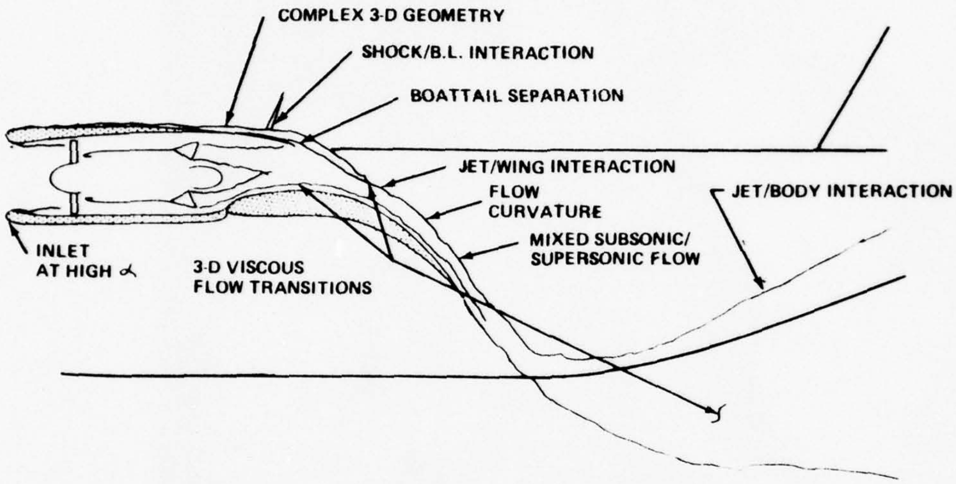


Figure 13. Fluid Flow Problems of Propulsion Powered Lift Installations

through a duct with a changing cross section. This distorts and strains the turbulence and generates a complex secondary flow in the cross plane of the duct. After being accelerated by the nozzle, the jet emerges to mix with the flow coming over the nacelle and wing, which may include separation regions. On the wing side, a 3-D boundary layer starts to develop. This flow is now complicated by streamline curvature and divergence effects. At high power setting, this whole region is criss-crossed by shock and expansion waves to further complicate the flow. Downstream of the wing, the flow continues as a curved 3-D jet which may or may not interact with the aircraft body.

A major problem with powered lift aircraft is the strong interaction between the viscous and inviscid regions of the flow. This is illustrated in Figure 14 from reference (2) which compares the lift/drag characteristics of powered lift and conventional aircraft. On powered lift aircraft, it is not possible to analyze the wing and the nacelle separately and then to design for minimum interference. Because of this, our ability for detailed analysis of such aircraft is almost completely lacking.

Whether or not it will ever be possible to obtain solution of the full time-dependent Navier-Stokes equations for problems of practical interest is a debatable question. There is no question, however, that for many years to come, we must settle for much less detailed analysis.

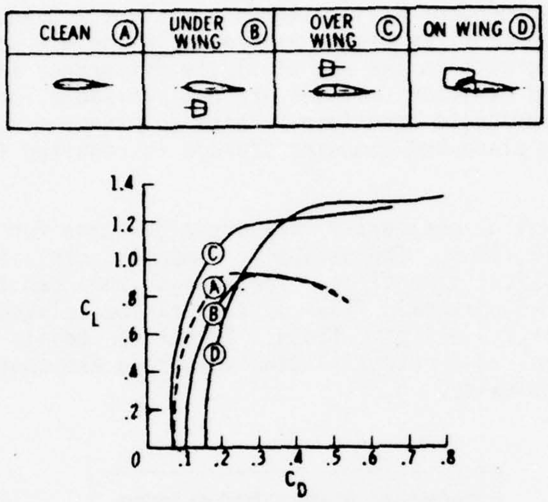


Figure 14. Lift/Drag Characteristics of Powered Lift and Conventional Aircraft

Because of this, most current work is concentrated on solutions of the time averaged equations. At first consideration, this may not appear to be a serious problem. After all, we are concerned primarily with the mean flow, and even if we had solutions for full time-dependent N-S equations, the first thing we would do is to time average them. The problem is that the time averaged equations contain second order correlations. Equations can be derived for these correlations, but these equations contain additional correlations, so there are always more unknowns than equations. This is the turbulence closure problem, and it is the mathematical consequence of the loss of information which occurs when the Navier-Stokes equations are time averaged. The second order correlations known as the Reynolds stresses, which appear in the mean flow equations, appear as gradients, but since the Reynolds stresses themselves are generally small compared with the total pressure, they can only affect the flow in regions where the gradients are high. Therefore, much of the flow can be considered as inviscid. Unfortunately, these high gradient regions are often the critical regions, and the accuracy and utility of the solution as a whole often depends on the accuracy with which the flow in these regions can be predicted. Hence, the importance of turbulence models.

The different levels of complexity of turbulent flows are illustrated in Figure 15. These range from 2-D parabolic flows to solutions of the full time-dependent Navier-Stokes equations. Since methods for solving 2-D parabolic flows became available in the mid-sixties, there has been a rapid growth in the use of finite difference methods, and a number of fast and efficient methods are now available for solving coupled sets of parabolic equations. These solutions march downstream from the starting plane, and computer storage is required for only one plane of data.

The next level of complexity includes algorithms for solving 3-D parabolic/elliptic flows. The parabolic dimension can either be time or one of the physical dimensions. These algorithms can therefore be used to predict 3-D parabolic flows or they can be relaxed in time to give solutions for 2-D elliptic flows. Turbulence models are required in all these cases. 3-D potential flow solutions are approximately equivalent in complexity.

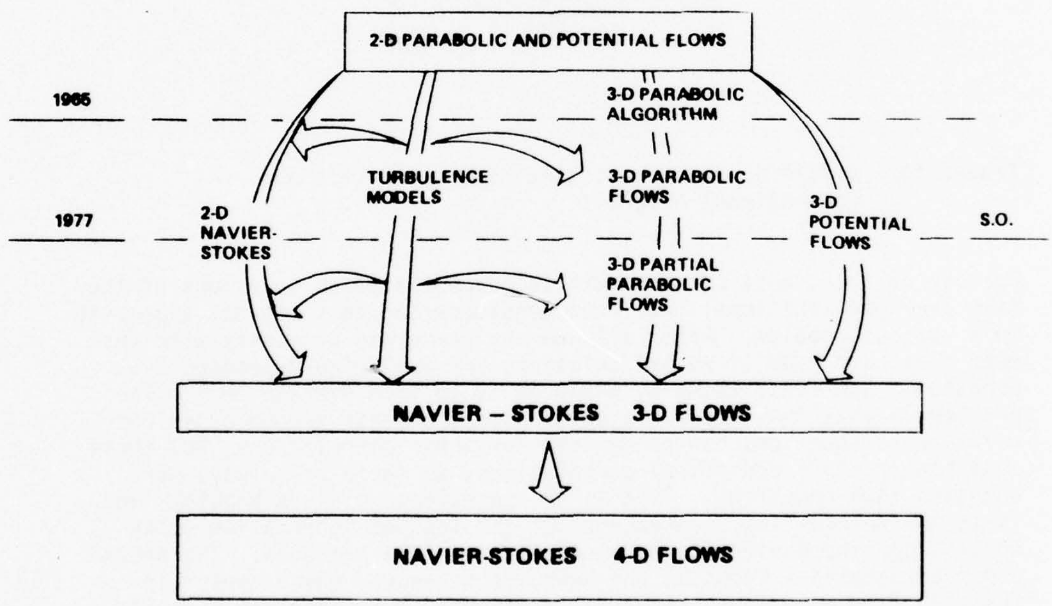


Figure 15. Current S.O.A. - Numerical Fluid Mechanics

The next step is a big one. Solutions for the full time averaged Navier-Stokes equations require numerical algorithms valid for 3-D hyperbolic/ elliptic flows. The major problem here is that full 3-D computer storage is required for all variables. With present computers, solutions of this type are possible only for selected flows. If there is no separation in the main flow direction, a simpler procedure can be used. If the 3-D parabolic solution is iterated and the pressure is updated at each iteration, it is possible to relax toward an elliptic solution. Such algorithms are called partially-parabolic or semi-elliptic. Turbulence models are also required in these cases. The principle advantage of this procedure is that full 3-D storage is required only for the pressure.

Solutions for the full time-dependent Navier-Stokes equations are so far beyond the capacity of present computers that this type will not be discussed in detail here.

Overall Objective

The overall research objective is thus to develop methods to compute the flows, and aerodynamic and propulsive performance characteristic of powered lift aircraft for the selection of designs. This requires fully 3-D analysis of mixed subsonic and supersonic flows, including all of the relevant viscous flow processes. Testing will still be required, but to validate the analysis procedure rather than to provide an empirical data base for configuration selection.

The transition from a test-based to an analysis based design process will not happen suddenly. Each analysis development can be used to improve the design process as it is available, and it is essential that the analysts work with the designers each step of the way to improve the design process. This incorporation of new analyses, as they become available, into the design process was illustrated above in the inlet example.

Overall Approach

In developing a research plan, it is important that this overall objective be kept clearly in mind and that potential problem areas be identified early and, if possible, anticipated. One of the most difficult of these problems is the coordination of the various parts of the research effort. The overall success of such a research plan frequently depends more on such coordination than on success in the specialized research areas. For example, it is important to insure that the numerical solution algorithm and the turbulence model are compatible and consistent with the capacity of the available computers. An over simplistic turbulence model can severely limit the utility of a

design tool, while an unduly complex turbulence model may be useless if its inclusion in the program requires a storage capacity which is greater than that available. It might seem that such considerations would be obvious, but in practice this does not seem to be the case. There are surprisingly few groups working on numerical fluid mechanics that successfully balance the competing requirements of the various parts of the work.

I believe that a "broad front strategy" advocated here will generally be more successful than a series approach, where one portion of the problem is solved before the other portions are considered. With present computer limitations, the development of the design tools of maximum utility require too fine a balance between complexity and generality to offer any reasonable hope of success for a development effort that proceeds in a piecemeal fashion.

The analysis elements which must be coordinated for optimum progress toward the overall objective are as follows:

- Numerical and computational mesh generation
- Turbulence modeling
- Computing capacity
- Data handling and display
- Analysis validation.

Computing Capacity

Analysis to be developed and incorporated into the design process in the next decade should reflect the growth in computing capacity and speed to be expected. Analysis developed with today's computing speed and storage in mind as constraints will not fully utilize the available computing resources when the analysis is finally ready to be applied five or ten years in the future. While the analysts cannot do much about the computer resources to be available to him in the future, he should be cognizant of what is now and what is likely to be available to him.

Figure 16 is a summary of where we are likely to be in computing speed and storage. The new large, fast computers such as the CRAY and CDC Star offer the potential that fully elliptic solutions of the Navier-Stokes equations, at least in local flow regions, may soon be feasible. Current NASA planning is to make available large computer resources through Government research centers in much the same way as large wind tunnel facilities are handled today.

Numerical and Computational Mesh Generation

Given that the model to represent turbulence for the flow physics

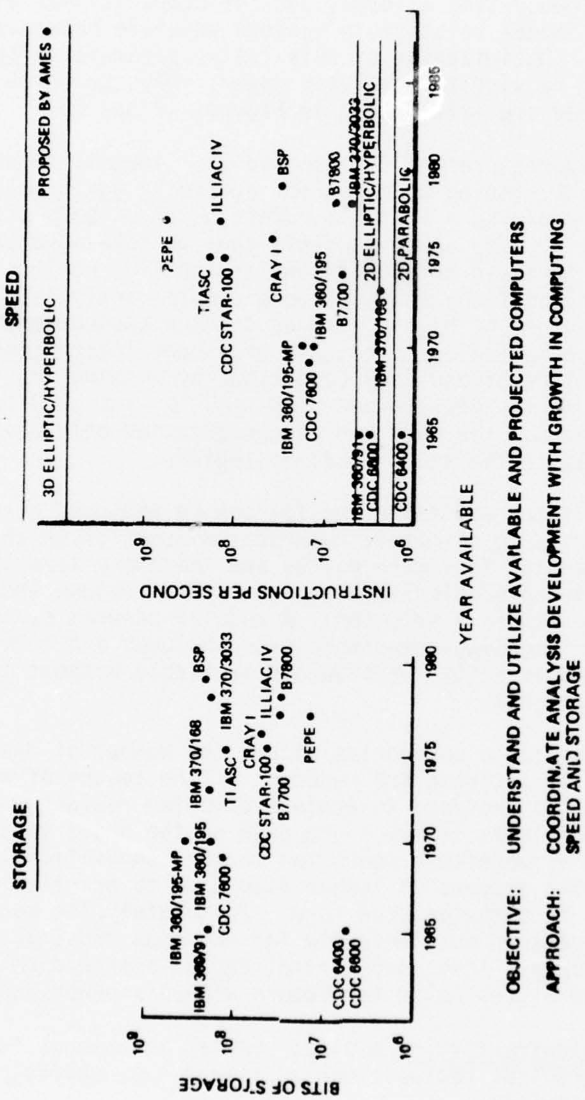


Figure 16. Computing Capacity

of interest is of adequate inherent accuracy, the power to analyze physical phenomena with computer techniques is limited by the computer speed of processing instructions, the amount of storage available, the computer residency time allowed, and the computational efficiency of the numerical model relative to various possible numerical model formulations. Optimization of this latter parameter offers great potential for raising the analysis power. Ways by which this may be accomplished are illustrated in Figures 17 and 18.

These figures present the powered lift installations from the viewpoint of developing numerics for analysis, again only as an example. This could be a V/STOL aircraft or an aftbody problem for a fighter as well. The chart suggests that we take advantage of the parabolic or inviscid nature of some of the flow regions to reduce our computing requirements. The more complex analysis then would only be used in regions of highly viscous transonic/supersonic 3-D flow. Also shown schematically is the use of a body-fitted mesh employed for easy handling of boundary conditions by eliminating irregularly shaped cells in the transformed plane and for improved computational efficiency because the mesh can be concentrated only where it is needed, such as in the steep gradient regions.

Local fitting and filtering (3) can be employed at shocks and plume boundaries to eliminate numerical dispersion, which allows higher order schemes with less mesh points and therefore less computing time. Covergence acceleration can be used to reduce the number of iterations to obtain a solution. A modular program strategy can be used to avoid recoding components over and over and to allow modules to be replaced as better ones become available, without disrupting the overall program.

In all of these categories, either the number of descrete points of the field of interest are reduced or the amount of work at each discrete point is reduced to achieve computer residency time reduction. Incredible as it may sound, each of the above tactics may yield orders of magnitude reduction in the computational costs in practice at the expense of longer flow time to practically implement the analyses in computer code form. Fortunately, the manpower time consumption goes up only modestly for enormous reductions in computational costs. Thus, substantial emphasis should be placed upon the six areas listed below for future analysis power improvements.

- Discontinuity fitting employs special techniques for treating steep gradient regions such as slip lines, shocks, and contact discontinuities.

- Patching techniques involve applying powerful expensive analysis only where needed. Cheaper methods are applied elsewhere.

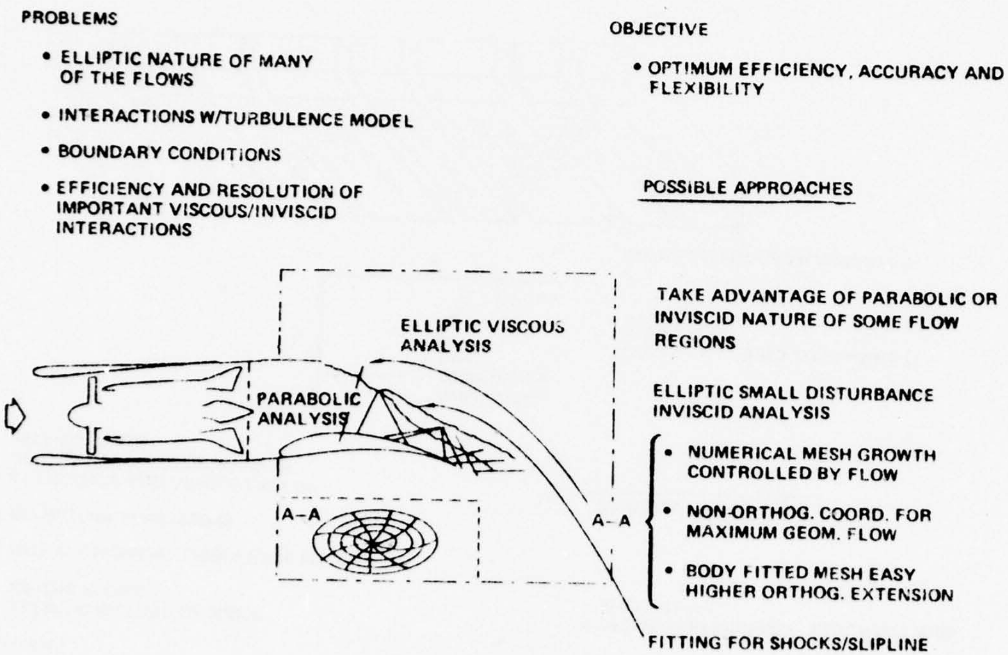


Figure 17. Numerics

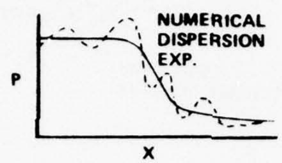
PROBLEMS

- ELLIPTIC NATURE OF MANY OF THE FLOWS
- INTERACTIONS W/TURBULENCE MODEL
- BOUNDARY CONDITIONS
- EFFICIENCY AND RESOLUTION OF IMPORTANT VISCOUS/INVISCID INTERACTIONS

OBJECTIVE

- OPTIMUM EFFICIENCY, ACCURACY AND FLEXIBILITY

POSSIBLE APPROACHES



LOCAL FILTERING TO ELIMINATE DISPERSION

MODULAR PROGRAM STRATEGY

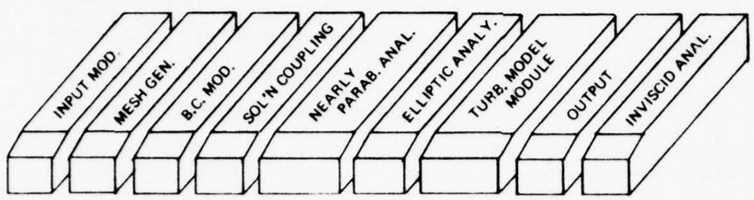


Figure 18. Numerics

- Convergence acceleration rate tactics may be used for reducing the number of iterations required to obtain the desired solution.

- Filtering techniques may be used to reduce the mesh required to resolve steep gradients which are not conveniently amenable to discontinuity fitting.

- Optimization of mesh gradation relative to the flow features of interest may be used.

- Higher order accuracy may be used to reduce mesh requirements.

Figure 19 is an example of a 3-D parabolic marching procedure with a two equation 2-D turbulence model used to predict the flow through a "D" nozzle. The calculation shows the use of a numerical mesh generator shown in the upper left. The right figure shows a contour map of the axial velocity at the exit of this nozzle. The lower left is a plot of the cross flow vectors showing the three-dimensionality of the flow and perhaps is the most important part. This is the result of a duct transition from a round engine exit through an "S" type duct to a D shape at the nozzle exit. This gives an indication of the true thrust from the nozzle and a start for the analysis of the flow as it continues over the wing and flap in a powered lift aircraft. What we would like is to have the ability to continue through the complete flap analysis. A point is that in this case, the internal character of the flow has a major influence on the conditions at the nozzle exit and therefore has a significant effect on the downstream flow development.

Turbulence Modeling

Recall that a turbulence model is necessary to supply the information lost when the Navier-Stokes equations are time averaged. Good modeling is available for most 2-D flows; however, modeling of the more complex 3-D flows is really just beginning. The current state-of-the-art, possible approaches to improved models, and modeling objectives are illustrated in Figure 20.

Phenomenological turbulence models have traditionally been devices used by experimentalists to organize and classify large quantities of experimental data. A turbulence model, as its name implies, is a mathematical model of a physical process which, of necessity, must be based on a clear understanding of the basic physics involved. However, as turbulence models have increased in complexity, a widening rift has developed between experimentalists and those attempting to develop improved turbulence models -- a rift which, if allowed to continue, is certain to retard future programs.

In view of this, a classification system for complex flows,

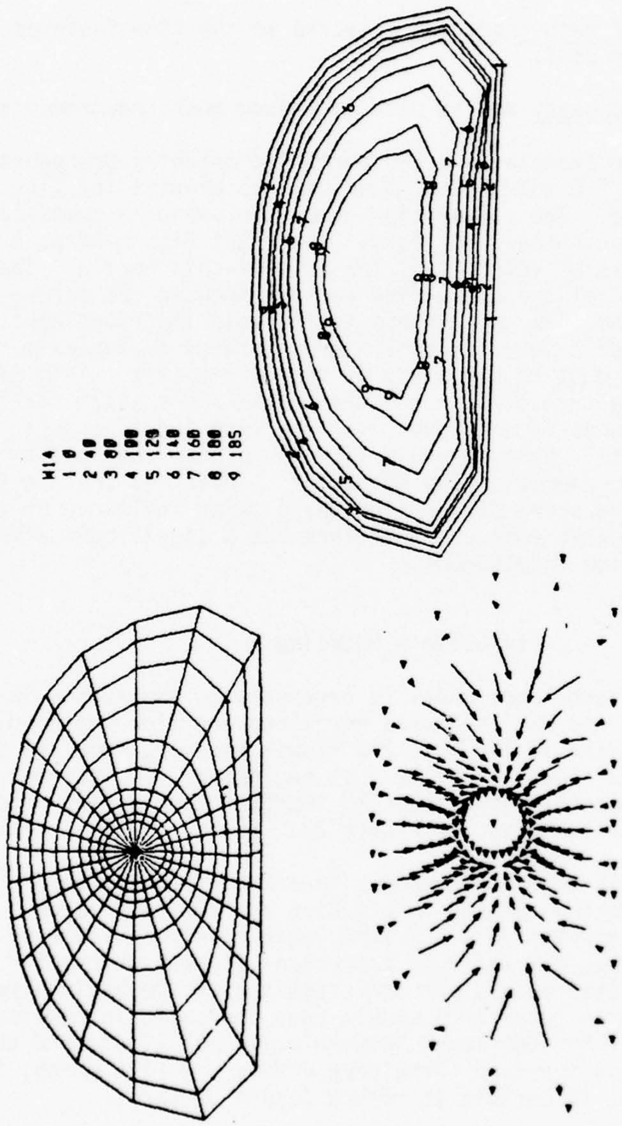


Figure 19. "D" Nozzle Analysis

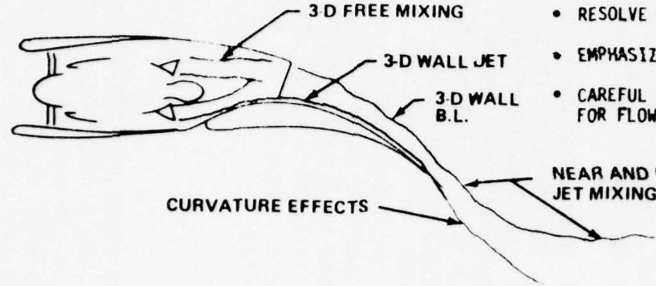
PROBLEMS AND S.O.A.

- SOL'N OF "FULL" N.S. EQ.'S IS BEYOND THE CURRENT AND PROJECTED COMPUTING CAPACITY FOR PROBLEMS OF INTEREST
- WHEN N.S. EQ'S ARE TIME AV'GED, INFORMATION IS LOST
- TURB. MODEL ALLOWS SOL'N OF TIME AV'GED N.S. EQ'S THRU SUPPLYING "LOST" INFORMATION
- GOOD MODELS AVAILABLE FOR MOST 2-D FLOWS – MODELING OF 3-D FLOWS HAS JUST STARTED

OBJECTIVES

- SIMPLICITY
- COMPATIBILITY WITH NUMERICS
- ACCURATE FOR A WIDE RANGE OF FLOWS

TURBULENT FLOWS WHICH MUST BE MODELED INCLUDE . . .



POSSIBLE APPROACHES

- USE BRADSHAW CLASSIFICATION OF 3-D TURBULENT INTERACTIONS
- RESOLVE 2-D/AXISYMMETRIC MODELING PROBLEM
- EMPHASIZE MODEL SIMPLICITY
- CAREFUL EXPERIMENTAL VALIDATION OF MODELS FOR FLOWS TO BE PREDICTED

Figure 20. Turbulence Modeling

proposed by Bradshaw (4) and developed by him in a series of papers, must be regarded as one of the most important and interesting developments in turbulence research in recent years. Such a classification provides a blueprint for future development in turbulence models and is an indispensable guide for the experimentalist seeking to unravel the mysteries of complex 3-D flows.

Space here will not permit a detailed discussion of this work or comparison of individual turbulence models, but there are a few topics which should be discussed. The first is the difficulty of predicting axisymmetric and planar flows using the same turbulence model. A resolution of this problem is a basic prerequisite for the accurate prediction of general 3-D flows. There is also a need for relatively simple turbulence models capable of predicting complex flows, including flows in which more than one of the Reynolds stress components is significant. It is probably inevitable that as turbulence models grow in generality, they will also grow in complexity. For the present, however, the degree of complexity which can be handled is limited by the storage capacity of present computers.

Figure 21 gives a typical example of today's turbulence modeling capability for complex flows. The example is from the 12 lobe internal mixer presented earlier, where good total temperature distribution comparisons were shown. The detailed flow picture, however, shows the cells more elliptical than the test data, but the peak temperature levels are accurate. The difference occurs because a turbulence model, with constants optimized for planar flows, was used. If the far downstream flow characteristics were important, it would be necessary to adjust the constants in the model. Mother Nature wants the peak total temperature regions to become almost circular, but the turbulence model used caused the cells to retain their elliptical shape. This implies further development of the turbulence model is required.

Data Handling and Display

Vast quantities of information must be manipulated if the information available from the analysis is to be useful. It is important to make the analysis yield results similar to that produced from tests, such as velocity profiles, pressure and temperature. The data available from the analysis must produce useful design information quickly and with a minimum of manpower. We feel computers dedicated to data handling and display are a good solution to these problems. Special software can be developed for data display to make this as fast and easy as possible.

The use of a computer dedicated to display information from a 3-D viscous analysis of the flow development through a 727 nozzle is illustrated in Figure 22. This flow, discussed above, was assumed

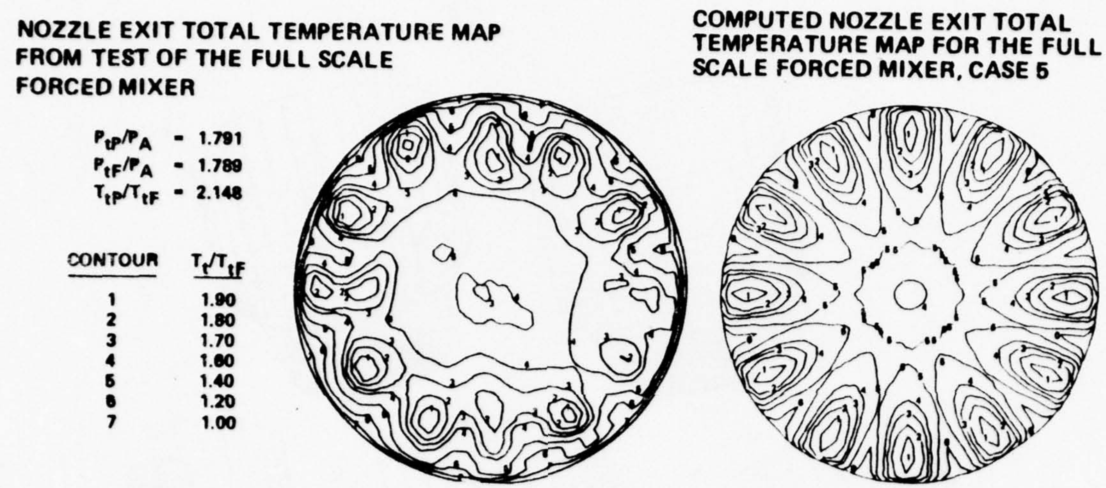


Figure 21. Total Temperature Distribution at Exit Plane of a Turbofan Engine Nozzle

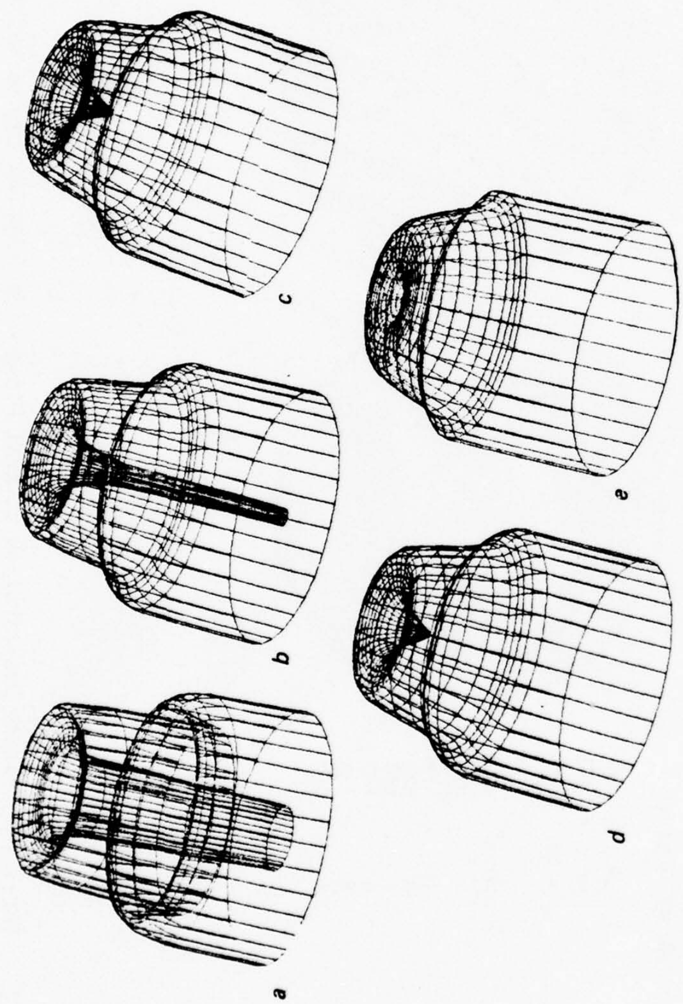


Figure 22. Axial Velocity for 727 Tailpipe with Standard Splitter

to be axisymmetric for purposes of this illustration. Given are the 3-D contour plots of axial velocity as the flow develops through the nozzle. Note the gradual mixing out of the wake from the turbine centerbody as the flow approaches the nozzle exit.

Analysis Validation

To be successful the above work must be accompanied by a series of well-planned and professionally executed "bench mark" experiments to validate the analysis for the complex flows of interest. This experimental work will be discussed in more detail in the next section. The point to be made here is that turbulence models and numerics cannot be developed in isolation. A lack of experimental data, like the poor, will probably be always with us, but at present there is a grave shortage of detailed experimental data for 3-D configurations of practical importance. To quote just two examples, there is no published, detailed experimental data for the 3-D boundary layer in inlets at angle of attack, and the detailed experimental data available for 3-D jets is limited to simple nozzle geometries such as elliptic or rectangular shapes.

IMPROVED TESTING METHODS

In 1973, Professor Kline from Stanford chaired a committee to recommend critical experiments for the NASA Langley Turbulent Free Shear Layer Conference(s). This committee came up with two classifications of experiments.

1. Technology Tests. The objective of these tests is parametric data base for configuration selection. These tests are characterized by the measurement of gross performance parameters (thrust, drag, recovery, etc.), a lack of detailed flow measurements, and a poor simulation of the flow in detail.

2. Test to Validate of Support Analyses. The objective of these tests is to provide information for analytical modeling of the flow or for use in the validation of a flow analysis. These tests are characterized by simple, isolated flow processes, careful control of initial and boundary conditions, and detailed measurement of mean and turbulent flow properties.

Unfortunately, it is seldom possible to combine both objectives in the same experiment. Attempts to do this generally result in experiments that satisfy neither objective.

Technology Tests

Most applied experimental work differs from experiments to support analysis development in that the primary objective is the measurement of gross performance parameters such as thrust, drag, pressure recovery, etc. Detailed flow measurements are seldom taken. This is partly due to the cost involved, most often because such measurements are either impractical or impossible for the flow being studied. However, as aircraft become more sophisticated and more complex, improved test techniques which provide a more accurate simulation of the full scale flow are becoming necessary. This includes the use of turbo-powered nacelles, hot, blown nacelles, and the use of non-intrusive instrumentation, LDV's, and laser holographic interferometry on a production basis. As numerical analysis tools are improving, this applied experimental work is being used more and more to validate the analysis, which is then used for parametric studies. In spite of this trend, it is important to remember that there is a basic difference in philosophy between the two types of experimental works.

Tests to Validate or Support Analyses

Even when the mean flow in turbulent flow is 2-D, the turbulence is always 3-D, so that three-dimensionality is not necessarily, in itself, a major complication. Because of this, many of the basic features encountered in complex 3-D flows can be studied in simpler 2-D geometries. This is not only possible, it is often desirable if the feature being studied is to be isolated from other phenomena. There are of course, certain flow phenomena which are inherently 3-D, 3-D wall boundary layers for example. To be of use in the development of improved turbulence models, these experiments must be carefully planned and professionally executed. This is generally expensive both in time and money. Very few such experiments exist, even for 2-D flows; for 3-D flows, we are faced with a major problem. It seems unlikely that experimental data of the required quality and quantity for these flows are going to be generated without a well-coordinated long range plan. Because of this, Boeing has recently suggested that NASA undertake the planning and execution of a series of Benchmark experiments. The objective of this work should be to generate detailed experimental data for complex 3-D flows to be used to evaluate the performance of current turbulence models and to provide the basic information required for future development.

The development of non-intrusive and flow-mapping test techniques that can be used model or full scale to supply detailed mean and turbulent flow information is also very desirable. Experimental techniques which should be further developed are as follows:

- LDV
- Laser holography
- Hot film/hot wire
- More use of translating/rotating P_t, T_t rakes

An example of the use of laser holographic interferometry to provide a detailed mapping of the density field at the exit of a complex tube nozzle geometry is given in Figure 23.

PROPULSION SIMULATION AND CONTROL

Rising fuel costs and increasingly stringent aircraft mission requirements will provide increasing incentive for optimizing aircraft performance in a wide variety of environmental conditions and operating scenarios. This situation, in an era of powerful and economical microprocessor-based computational capability, is expected to lead to functional integration of aircraft avionics, flight control, and propulsion control systems. This integration is particularly important for V/STOL aircraft because of the relatively complex propulsion/flight control interactions. The development of digital systems to achieve the highest overall efficiency of an engine/airframe combination will require propulsion research in areas that range from thrust modulation techniques for flight control purposes, with designs of highly reliable digital hardware and software, to the adaptation of modern optimal control theory to the special needs of propulsion systems.

Substantial benefits may be realized from early identification of the research needed and the establishment of goals. The existing state, as represented by current programs and recent research must be evaluated in light of requirements projected for the 1980-1990 time frame, as illustrated by Figure 24. The areas that we feel require in-depth exploration are indicated in the figure.

Increasingly stringent aircraft and mission requirements have led to a proliferation of electronic "black boxes" that have become a nightmare in terms of procurement cost, reliability, and maintenance cost. Since the requirements are going to get worse instead of better, we must seek relief through the only avenue open, integration of controls and information systems. Controls integration is just now becoming feasible as a result of developments in digital data busses and electronic microprocessors. The ground work for the application of integrated digital controls has already been laid by concurrent research and development programs in propulsion and flight controls as shown in Figure 25.

Included are fly-by-wire and control configured vehicle testing by NASA and the Air Force on F-8, F-4, and B-52 aircraft. These programs and others have demonstrated the feasibility and advantages

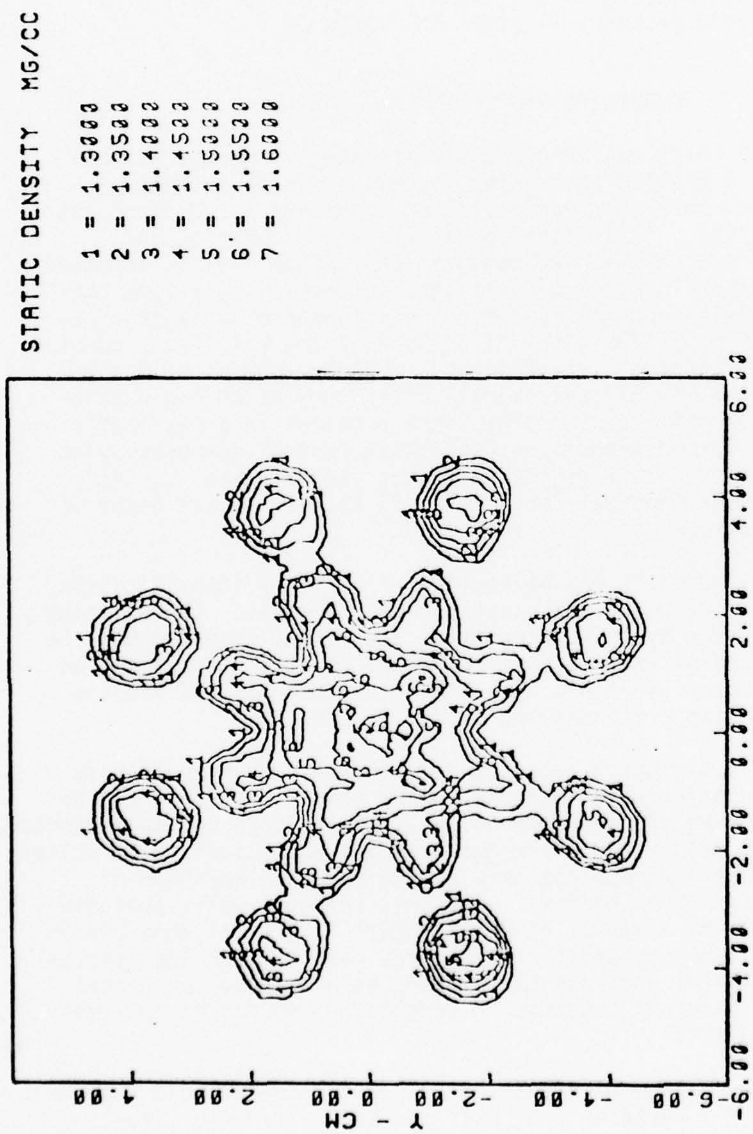
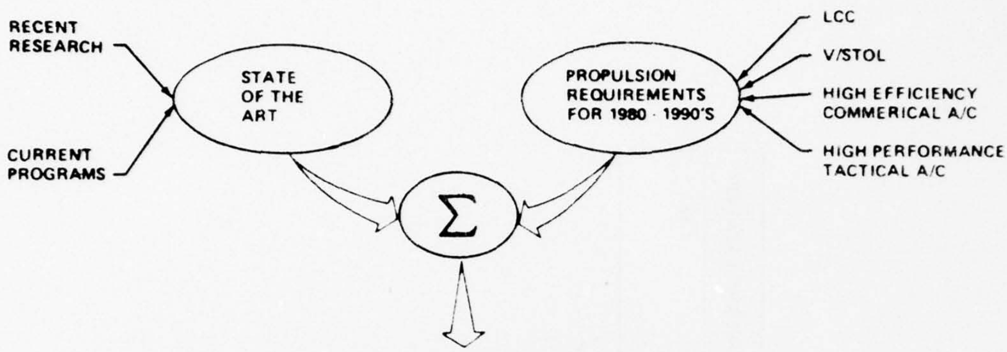


Figure 23. Eight Tube Nozzle, Scalloped Core



REQUIRED PROPULSION RESEARCH

- **THRUST MODULATING/VECTORING TECHNIQUES**
- **CONTROL SYSTEM DESIGN FOR RELIABILITY**
- **CONTROL SOFTWARE DEVELOPMENT AND TEST TECHNIQUES**
- **MULTI-VARIABLE OPTIMAL CONTROL THEORY DEVELOPMENT AND APPLICATION**

Figure 24. Identifying Research Requirements

of electronic flight controls. Direct application of this technology has been used in the development of the F-16, F-14, and YC-14 aircraft. With the ground work laid, we now need to tie the elements together into a viable operational system. The development of improved electronics is a significant task that is being pursued in many areas. The efficient application of the technology is also a big task that must be addressed by propulsion and controls people. Areas of major effort are summarized in Figure 26.

Perhaps the greatest impediment to the adoption of digital controls is concern over system reliability. Despite significant advancements made in micro-circuit reliability in recent years, the MTBF of most systems developed to date is only marginally acceptable for operational aircraft applications. Additional research is necessary to improve the reliability of digital hardware. Potential areas of investigation include improved digital circuit design and hardware packaging techniques to reduce the effects of temperature and vibration. Also, improved testing methods, such as accelerated stress testing, should be applied to screen out potentially faulty micro-circuits more quickly and accurately. The reliability situation is summarized in Figure 27.

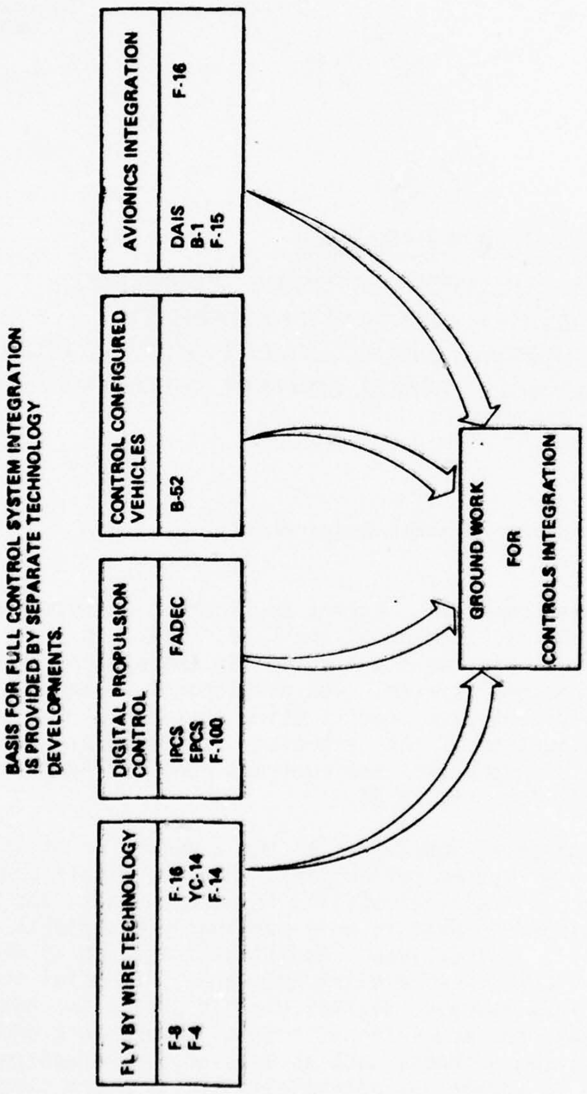


Figure 25. Groundwork for Controls Integration

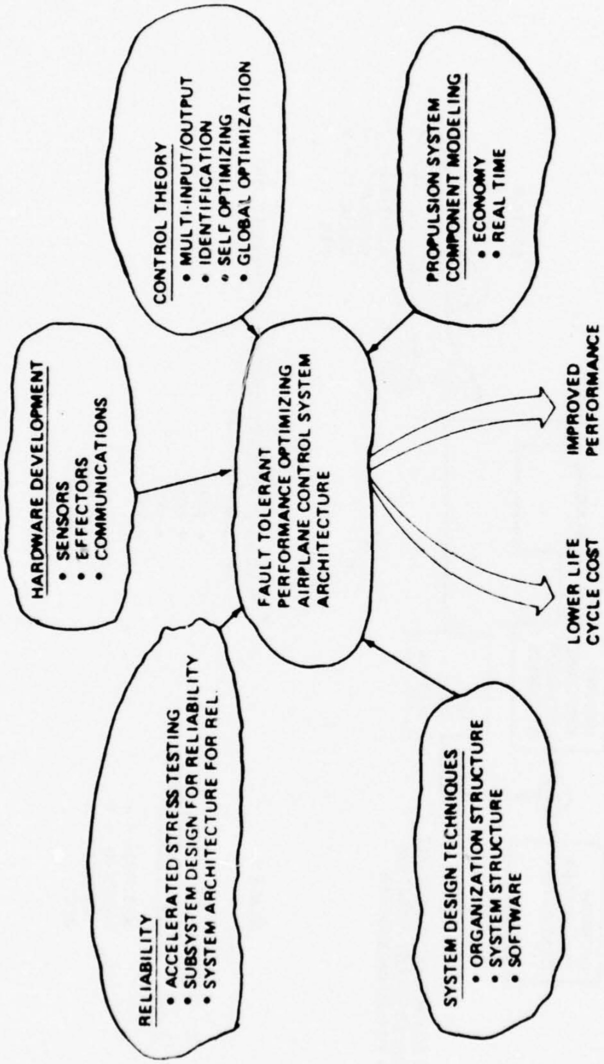


Figure 26. Importance of Control System Research

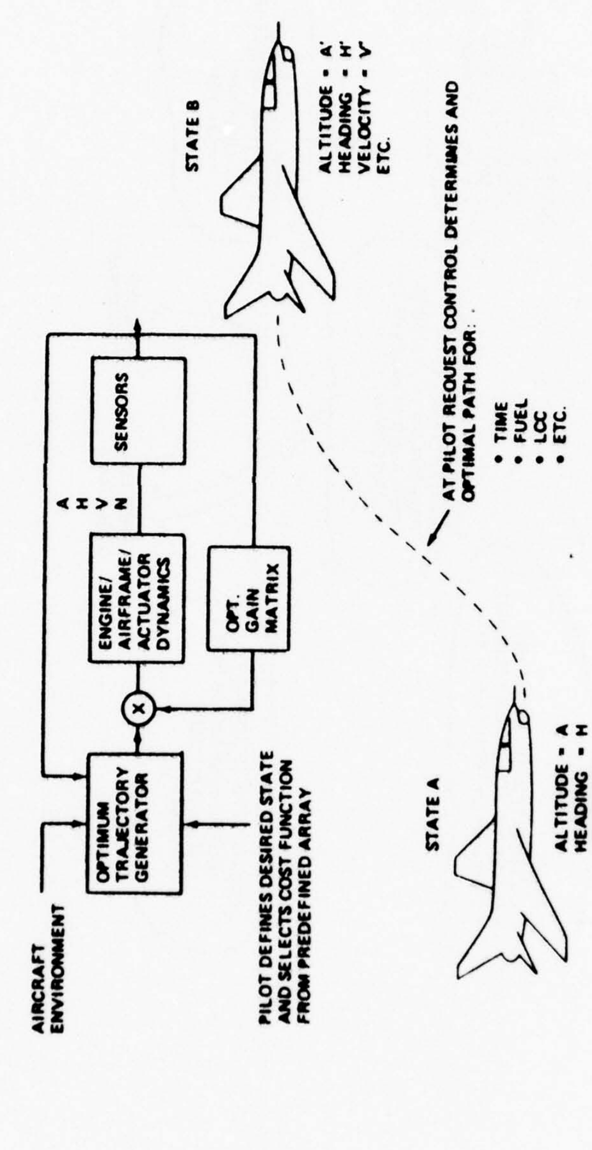


Figure 27. Improved Digital Electronics Reliability

Figure 28 indicates that as the computational capability of digital electronics continues to expand, the options available to the controls engineer will increase. The classical control design techniques, which involve the evaluation of each control loop on an individual basis, were adequate for older simpler engines, but such techniques will prove increasingly cumbersome and time-consuming if applied to variable cycle engines. The advent of digital electronics capable of matrix manipulation has cleared the way for adoption of multi-variable control techniques which offer potential improved performance. The linear quadratic regulator (LQR) is one specific area of modern control theory which is actively being investigated for application to engine control. This work needs to be expanded, together with examination of additional areas of modern control theory.

Although simulation and wind tunnel testing permit verification of many of the concepts outlined above, only testing of complete systems will generate the confidence required to proceed to flight status. Extensive ground testing of complete systems using iron bird test rigs, such as that shown in Figure 29, can achieve many of the goals of a flight test program at a much lower cost.

An extensive real-time simulation that interfaces with the test rig, the aircraft control system, and crew systems is required to achieve maximum utility from the iron bird rig. Figure 30 shows the block diagram for a real-time simulation that might be used in conjunction with an iron bird test rig to evaluate control laws and check out software and hardware.

Closing Remarks

In conclusion, I would like to summarize the key features of the research areas discussed above. Numerical fluid mechanics can yield the most significant advances in improving the propulsion system installation design process. Continued emphasis on the development of analytical tools will result in replacing parametric testing with parametric analysis. This will reduce the cost of the design process by orders of magnitude. A broad front strategy is advocated for the development of computational analyses. By following this approach, developments in numerics and turbulence modeling will be coordinated with advances in computing capacity and data handling and display, such that optimum progress will be made toward developing analytical tools for designing powered lift aircraft. Improved testing methods need to be developed with the idea of using testing to identify crucial flow problems of powered lift aircraft and for the validation of analyses. The increasing cost of technology testing to provide design information has necessitated the emphasis placed on numerical analysis. Testing that leads to a better understanding of the fluid dynamic phenomena is needed for improving analytical tools.

OBJECTIVE: **40,000 HRS MTBF/10% OF ENGINE LCC**

APPROACH:

CIRCUIT DESIGN AND PACKAGING

- PURSUE GOOD DESIGN PRACTICES
- OPTIMIZE DESIGN FOR RELIABILITY
- STUDY PRIOR EXPERIENCE FOR DON'TS
- DEVELOP IMPROVED QUALITY CONTROL METHODS
- ACQUIRE APPROPRIATE AIRCRAFT ENVIRONMENTAL DATA
- INCREASED APPLICATION OF AUTOMATED DESIGN ANALYSIS

SCREENING PROCEDURES

REDUCE TEST TIME TO IDENTIFY FAULTY DEVICES AND ESTABLISH MTBF OF "MAIN POPULATION"

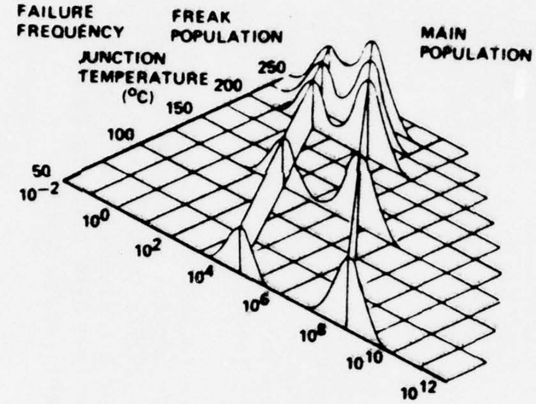


Figure 28. Long Term Goals on Control Theory

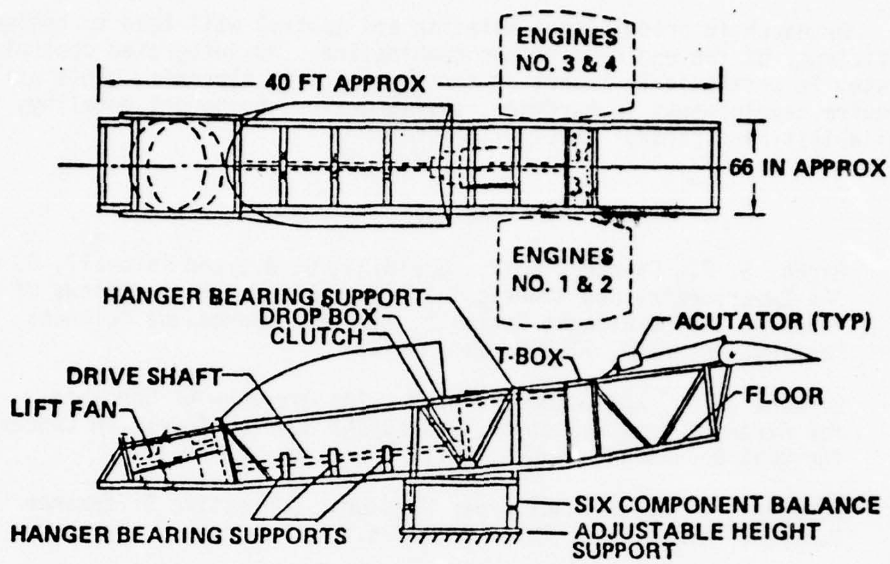


Figure 29. Iron Bird Test Rig

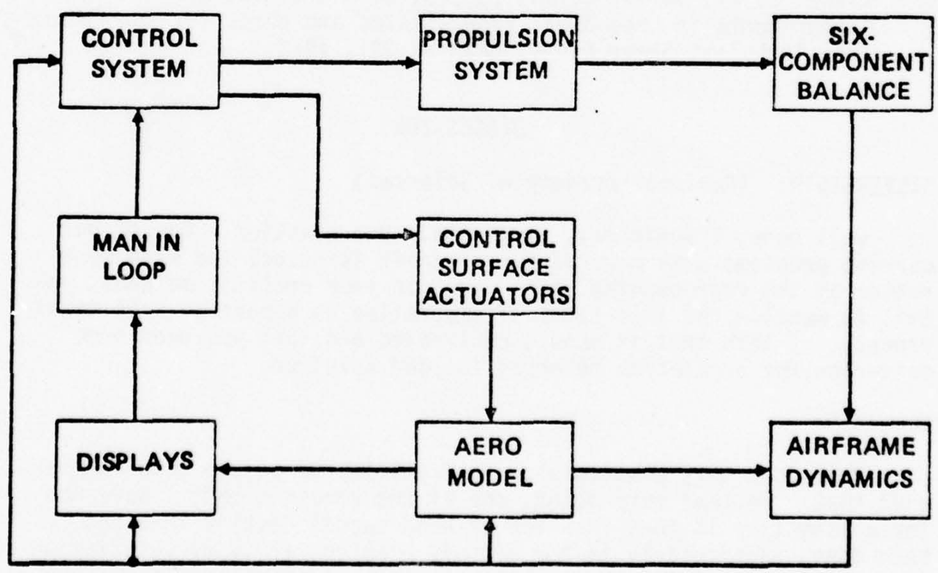


Figure 30. Simulation Block Diagram

Research in propulsion simulation and control will lead to optimal efficiency of the engine/airframe combination. An integrated control system is particularly important for powered lift aircraft. This will require developments in hardware, system design, component modeling, reliability, and control theory.

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DISCUSSION

SILVERSTEIN: (National Academy of Sciences)

Well done, I would say, very good. One question. One of our current problems with many of our aircraft is associated with separation at the rear causing base drag. In your analytic methods, you fail to mention the importance of separation as a part of that total process. I thin that it should be treated and that you need some criterion for separation to apply to your solution.

WELLIVER:

That is a very good point. In fact, let me go one step further than that. On that very point, one of the concerns that I have had for a long time is that in a lot of wind tunnel testing that has been done, particularly in the aftbody problem, it is my observation that the experimentor fails to establish what the initial conditions are. You will have a difficult time validating any analytical tool based on available data due to the experimentalist's failure to establish the relationship between the aftbody separation and the imposed

initial conditions. In reality, if you really want to comprehend some of the 3-D problems associated with aftbody separation, I believe you are better off spending a fair amount of money trying to understand the governing physical phenomena for the purpose of designing analytical tools. This becomes much more meaningful than just running a series of parametric tests. The basic elliptic nature of the flow in the separated region requires solution of the full Navier-Stokes equations, as I indicated in the presentation. Thus, I think it is very critical to understanding the aftbody problem, that we pursue a parallel approach with analytical modeling and wind tunnel experimentation. The analytical model must inherently treat the separation phenomena, as I have attempted to describe in my presentation. In addition, we must have assurance that the experimentor performs the testing with the proper simulation and control of the flow parameters that are affecting the interaction being studied.

PLATZER: (U.S. Naval Postgraduate School)

Can you tell me a little bit about the transonic inlet problem? Is your solution obtained with a Murman-Cole type relaxation method, or are you using the Lieblein-Stockman correction?

WELLIVER:

Well, it is based on the full transonic equation, and it is a finite difference algorithm. There is a pretty good description of that published in the literature. It is an AIAA paper presented by Ted Reyhner at the 1976 Aerospace Sciences Conference in Los Angeles, California.

HILL (Grumman Aerospace Corporation)

Being in the field of experimental research, I would like to point out that you left out one big slot in the middle of your two parts in the future. And that is, experimental investigation to basically find out what is going on. You had a very good example of that in your own results, in that you are computing an axisymmetric problem, but the real flow was not coming out that way. If that class of experiment is not done, there is going to be a whole series of nice computations of things that are not what exists.

WELLIVER:

I agree fully, as I stressed in the presentation. There are inadequacies in the computer models to simulate accurately certain physical processes. In particular, I emphasized through example that the turbulence model employed for the axisymmetric flow caused the overall computed total temperature contour shapes to disagree with measurements. This indicates a turbulence model deficiency which must be remedied by further experimental flow analysis to define the physics

and concurrent turbulence model refinement to accurately reproduce the physical processes. In addition, as I mentioned, the advent of LDV and laser holography techniques renders ever more important the future of the analytical/experimental partnership.

HEISER: (Arnold Engineering Development Center)

Would you care to comment on how the work at NASA-Ames ties up with the computational work you are describing?

WELLIVER:

Okay, I will comment on that! I think that, as a lot of people know, I have a very high respect for Ames in particular for their fluid mechanics work. For years they have done a good job. I think one of the areas is, if I were pushing Ames, as I was last night, I would push them in the area of trying to make the programs more user-oriented and cover a broader front, rather than some of the specific areas they are working on. It is hard to talk in generalities, I guess, but, for example, I think that turbulence modeling needs a broader, more general focus, and some other areas of the total analysis based design process have not gotten attention, to my knowledge. The basic numerical algorithms have received most attention, and that is one of the points I was making. It is very popular to work the code, but the usefulness of a code depends on the extent to which the governing physics have been modeled and how user-oriented the code is. And as I was pointing out in the discussion here this morning, people have run the full time-dependent, three-dimensional Navier-Stokes equations, we know that, and they are operational. That is not particularly pioneering. You can take a computer like the Cray I and, with that glorious program, you can analyze one cubic centimeter. Now what are you going to do with that? And that is part of the issue.

COMPUTATIONAL METHODS IN PREDICTING AIRFRAME PROPULSION SYSTEM INTERACTION

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ABSTRACT

With presently available computational methods and computers, complete solution of the total flow field about short-haul aircraft cannot be obtained in sufficient detail for use in engine-airframe integration. A more realistic approach appears to proceed along a path wherein components of the flow about aircraft are computed and integration effects are predicted as a result of matching these solutions along appropriate boundaries. This paper describes several computational techniques that are currently available for computing fuselage and inlet flows and typical results are presented.

Three different types of computational methods for forebody flow fields are discussed: (1) a shock-capturing code for supersonic flow, (2) a Navier-Stokes code for supersonic flow, and (3) a Navier-Stokes code for transonic flow. All of these methods are applicable to three-dimensional configurations.

Two different types of computational methods for inlets are discussed (1) a shock capturing code for supersonic flow and (2) a time dependent method for transonic and supersonic flows. The first method is capable of treating three-dimensional flows, while the latter is applicable only to two-dimensional flows.

Schemes for matching forebody and inlet flow field solutions are discussed.

INTRODUCTION

Historically, neither the aerodynamicist nor the propulsion

system designers would admit to the existence of airframe-propulsion system interaction. When the aircraft being designed were either simple or single point designs, such interaction effects were not large and could be justifiably ignored. With the added design complexity of modern aircraft and the trend toward designs that perform well in several different flight modes, airframe-propulsion system interactions cannot be ignored. Almost by definition, aircraft that are designed to take off vertically, make a transition to horizontal flight, and have the capability of transonic or supersonic flight will be aircraft whose performance is controlled by interaction effects.

Most VTOL research aircraft have been developed through extensive wind tunnel test programs. Little design information has been generated that is sufficiently general to serve as a basis for the design of the next generation of aircraft. To increase the design base will require either more extensive and systematic wind tunnel testing, an increasingly costly approach, or increased reliance upon verified computational methods. Although the fluid flow problems that are encountered in airframe propulsion system integration are very complex, numerical techniques are being developed and computers designed that will be able to analyze component flows with some confidence.

The purpose of this paper is to describe some of the techniques that are available or currently under development at Ames Research Center to analyze the interaction between the basic airframe and an inlet, primarily at present, of a fighter-type aircraft. The material is not intended to be comprehensive, but rather, representative of current developments in this rather restrictive interaction problem. A comprehensive description of all methods applicable to the wide variety of interaction effects on typical airframe configurations would be very useful to the VTOL/Short Haul community, but this task will have to await the attention of an individual having a more pedagogical bent. Three primary subjects are discussed in this paper: (1) methods for obtaining the flow about isolated forebodies, (2) methods for obtaining the flow about and through isolated inlets, and (3) a discussion of coupling concepts for these two flows.

COMPUTATIONAL METHODS

Before describing the details of several computational methods for analyzing either forebody or inlet flows or both, some general comments should be made. The overall objective underlying the development of any computational method is to generate a tool that can be used for predictive design. For the work described herein, the driving objective is to develop computational tools that can be used to predict airframe/propulsion system interactions for V/STOL aircraft. It is being tacitly assumed that we do not have, or will not have in

the foreseeable future, adequate computational capacity to compute, with a single program, the flow about a complete aircraft with sufficient detail to define interaction effects. In some cases, as will be pointed out later, some of the computational algorithms for obtaining such solutions have been or could, with a reasonable effort, be developed. The primary limitations, in spite of some claims to the contrary, are and will continue to be computer hardware speed and capacity. Until adequate computers are developed, the only option is to develop techniques which can analyze aircraft components separately and determine interaction effects by interacting the boundary conditions of the separate solutions. This so-called hybrid approach allows individual programs designed for specialized tasks to be used. Such an approach should permit discrete elements of the flowfield to be analyzed in detail without taxing machine storage requirements, as is the situation associated with detailed analysis of a complete flow system with one program. In addition, this approach will allow analysis of complex flow fields with existing computer hardware.

<u>Forebody Flow Field Methods</u> - Four different methods capable of computing forebody flow fields will be discussed. The first two are limited to supersonic flows, while the last two are applicable for transonic and subsonic flows.

<u>Shock-Capturing Technique</u> - Several recent papers, Ref. 1 and 3, have reported methods wherein the Euler equations are cast in conservative form and solved throughout a supersonic flow field using finite difference techniques. Due to their ability to define, with no special logic, shock waves in a supersonic flow, these methods are broadly labeled shock-capturing techniques. Discrete shocks can be fitted into the computational mesh, either as outer computational boundaries, Ref. 1, or floated in the computation mesh, as described in Ref. 2 and 3. As shown in Fig. 1, this type of computational method proceeds in a space marching procedure once the flow is defined along some starting line. The three-dimensional Euler equations are solved using, for the most part, MacCormack's second-order accurate differencing algorithm. The present version of the code, described in Ref. 1 and currently in use at Ames Research Center, incorporates a general geometry description package which utilizes parametric cubics to fit the body contours. With this generalization of the geometry description, arbitrary cross sections, representative of the fuselage of modern fighter aircraft, can be described and input into the computer code. In most cases, proper representation of the fuselage is more time-consuming in both engineering man-hours and computer time than the actual aerodynamic computation. An example of a fitted fuselage contour is shown in Fig. 2.

Some comments should be made regarding the limitations of the code. As presently programmed, it will treat three-dimensional supersonic flows with a discrete bow shock wave. Since the technique

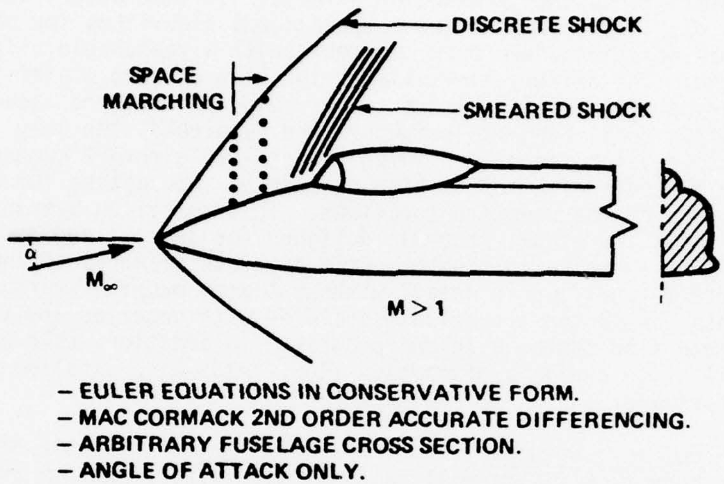


Figure 1. Shock-Capturing Technique for Fuselage

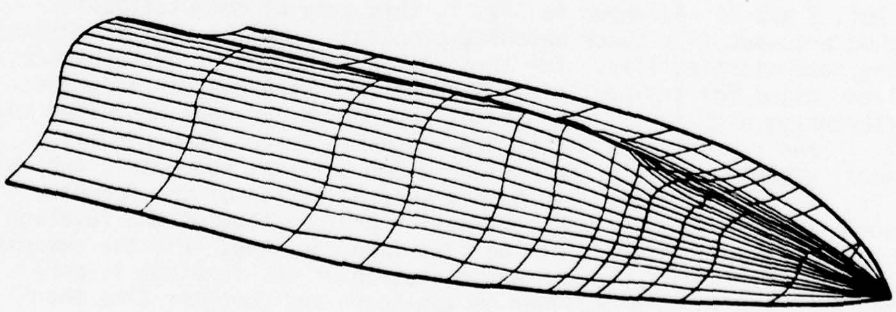
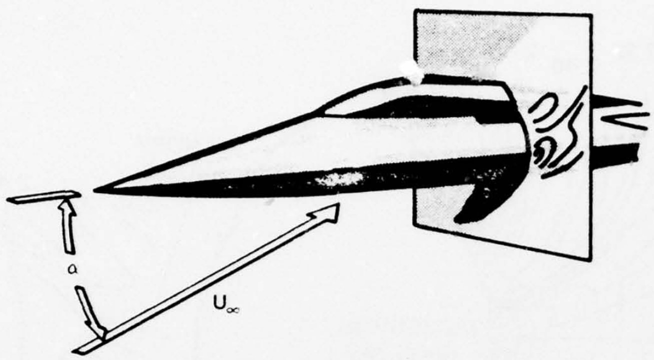


Figure 2. Body Fitted Contours



- TWO-DIMENSIONAL NAVIER-STOKES EQUATIONS
- TIME DEPENDENT IN CROSS-FLOW PLANES
- ESTIMATION OF BOUNDARY LAYER
- LIMITED CROSS-SECTION DESCRIPTION: $M_\infty \geqslant 2$

Figure 3. Navier-Stokes Technique-Hypersonic Equivalence Procedure

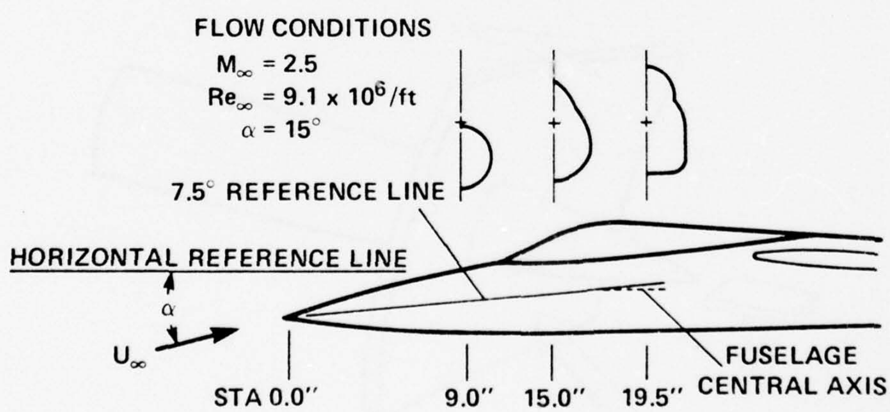


Figure 4. Fuselage Configuration

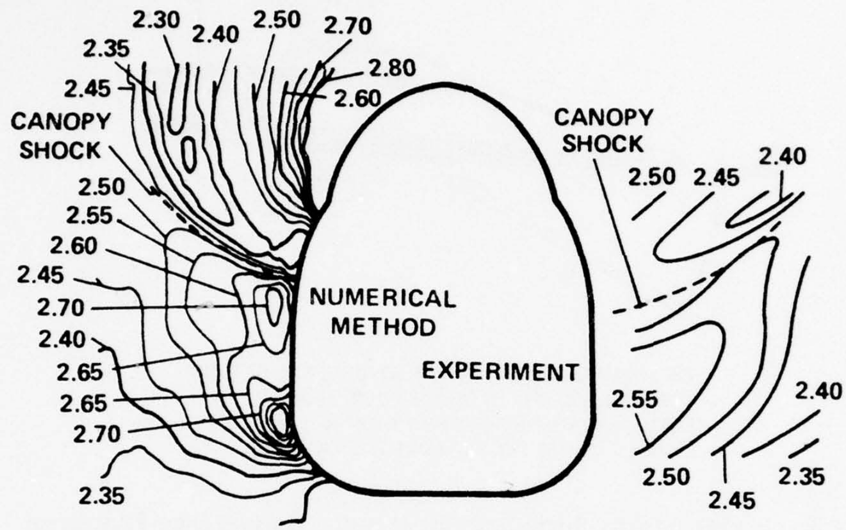


Figure 5. Comparison of Navier-Stokes Solution with Experimental data for M = 2.5, α = 15°

CROSS-SECTIONAL ITERATION

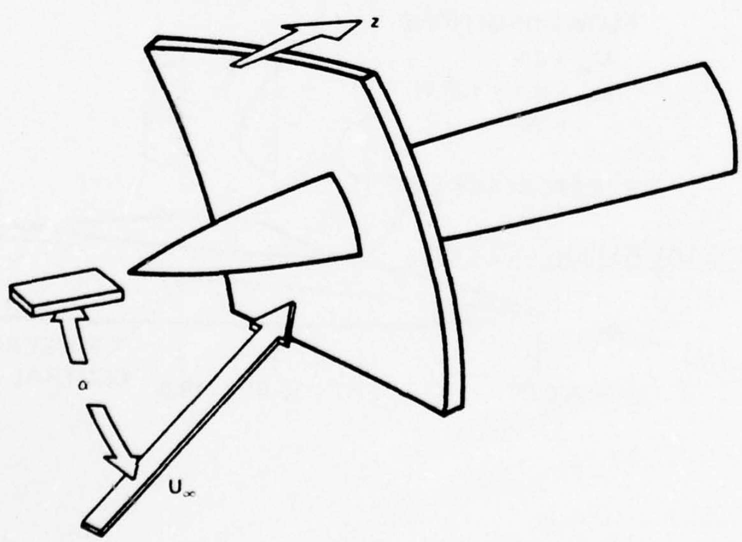


Figure 6. Navier-Stokes Technique - Alternating Direction Explicit

obtains a numerical solution of the hyperbolic equations of motion for supersonic flow, the flow must be everywhere supersonic. If the flow becomes subsonic, a real step size to advance the solution cannot be obtained and the technique fails. A scheme could be inserted that would solve for small regions of subsonic flow in a supersonic stream, but this has not been incorporated into the existing code. Thus, the present code is limited to supersonic flows.

Navier-Stokes Solution (Hypersonic Equivalence Principle) - This technique has existed for some time (Ref. 4) but is still the only technique available whereby simultaneous solution of the laminar boundary layer and inviscid flow field can be obtained for supersonic flow. The foundation of this approach is the equivalence between the three-dimensional hypersonic flow about a slender body and a two-dimensional time dependent flow. As shown in Fig. 3, the solution proceeds by computing the flow in two-dimensional planes wherein the body cross section changes with time--proportional to the distance from the origin divided by the free-stream velocity. The full Navier-Stokes equations for a laminar flow are solved in each cross sectional plane. A comparison of a numerical solution for the typical fighter configuration, shown in Fig. 4 with experimental data (taken from Ref. 4), is made in Fig. 5. The gross features of the flow are reproduced by the numerical method. However, significant differences in detail exist between the numerical solution, and the data, particularly in regard to the canopy shock wave. These differences are probably due to a combination of shortcomings of the basic numerical method, i.e., the hypersonic equivalence principle approximation, the exclusion of axial viscous terms, and the lack of a turbulence model in the numerical method since the boundary layer in the experimental data was turbulent. A further limitation that currently exists with this numerical method is lack of generality in the code for describing the fuselage cross section. As shown in Fig. 4, the fuselage cross section is described by a combination of circles and straight lines.

Navier-Stokes Solution (Alternating Direction, Explicit Method) - A new computational technique for calculating the flow about fuselages of arbitrary cross section is being developed on contract to NASA-Ames Research Center by Numerical Continuum Mechanics, Inc. This technique will include a turbulence model and will be applicable to subsonic and transonic as well as supersonic speeds. A generalized geometry package will be included that can fit arbitrarily shaped fuselages.

An iterative method will be used to solve the time averaged Navier-Stokes equations. As shown in Fig. 6, the first cross sectional iteration begins with a zeroth iterative approximate solution. The elliptic derivatives in the axial, Z, direction are approximated from the zeroth iterate. Explicit differencing is then used in the cross sectional plane. The next iteration in the meridonal plane uses the

results of the first iteration to evaluate the elliptic derivatives in the marching direction, now the meridonal angle, Φ, see Fig. 7. This iteration between alternating directions continues until a converged solution is obtained. Preliminary results for the flow in a centrifugal compressor indicate that a converged solution can be obtained in two or three iterations and that flow details such as separation are defined by the technique. It is estimated that approximately one hour of CDC 7600 CPU time will be required for a solution.

Paneling Methods.- Paneling methods, see for example Ref. 5, can be used to solve for subsonic and slender body supersonic flow wherein the flow field can be described by the linearized equations of motion. Within the range of applicability for the method, the flow about complete configurations can be defined. However, for detailed calculations such as those often required for integration effects, it is doubtful that adequate resolution can be obtained. Perhaps the greatest value for interaction problems lies in the use of paneling methods to define the outer boundary conditions for more detailed fine-mesh solutions of the full equations of motion, e.g., the Euler or the Navier-Stokes equations, or for determining starting conditions for more exact solutions.

Inlet Flow Field Methods.- Two different methods capable of calculating inlet flow fields will be discussed. Both techniques solve the inviscid equations of motion, one for only supersonic flows and the second for both subsonic and supersonic flows.

Shock Capturing Technique - Recently, a technique has been developed which can analyze the supersonic flow in an axisymmetric inlet at angle of attack see (Ref. 6). As shown in Fig. 8, this technique solves the Euler equations in conservative form using MacCormack's second-order accurate finite difference algorithm. The computational domain is partitioned as shown in Fig. 8, and the flow defined throughout the inlet, as long as the flow remains everywhere supersonic. A comparison of the flow in a typical supersonic inlet, Ref. 7, at $M = 2.65$ and two angles of attack, 0 and 3 degrees, is shown in Fig. 9a and b. At $\alpha = 0$, the flow in this inlet is nearly isentropic, with very weak shock waves. At $\alpha = 3$ degrees, stronger shock waves are produced, particularly on the leeward side of the centerbody, as noted by the sharp increase in pressure.

The inviscid method has been combined with the turbulent boundary layer program of Ref. 8 to design a mass removal system. With an appropriate mass removal system to control boundary layer separation, combined viscous inviscid solutions can be obtained by running the inviscid code over the original contours corrected by the boundary layer displacement thickness. A comparison of a combined solution at $\alpha = 0$, $M = 2.65$ with the original inviscid solution is shown in Fig. 10. Only small differences in the pressure distribution are apparent between the inviscid and combined solutions. This is due, for the main part, to the necessity for essentially removing the boundary layer in the throat region to obtain a combined solution.

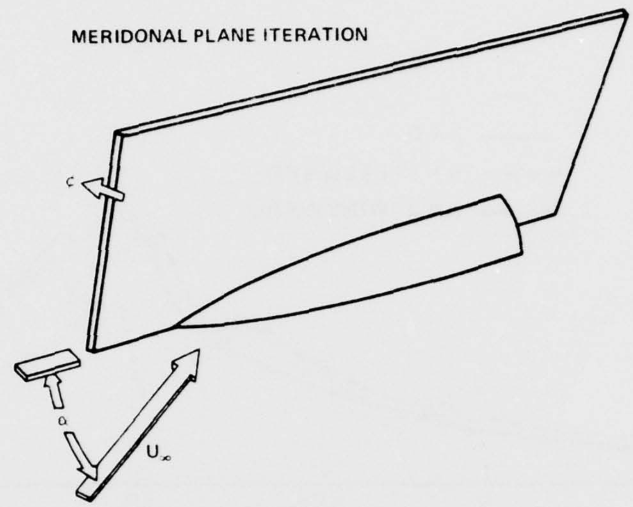
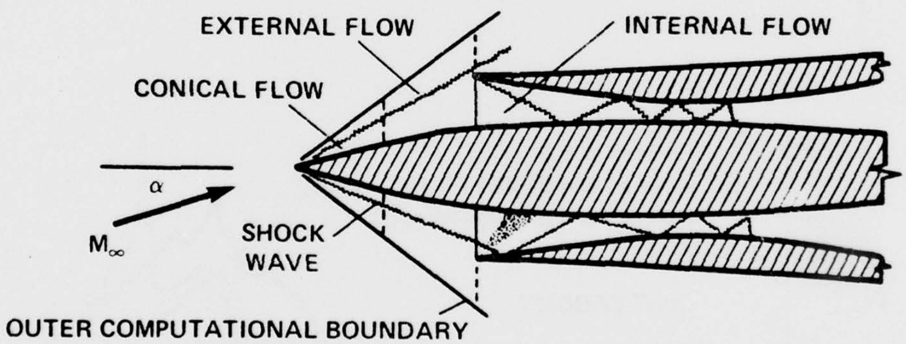


Figure 7. Navier-Stokes Technique



- EULER EQUATIONS IN CONSERVATIVE FORM
- MAC CORMACK 2ND ORDER ACCURATE DIFFERENCING
- LIMITATIONS
  - AXISYMMETRIC BODIES
  - $M > 1$ EVERYWHERE
  - SMEARED SHOCK WAVES

Figure 8. Shock-Capturing Technique for Axisymmetric Supersonic Inlets

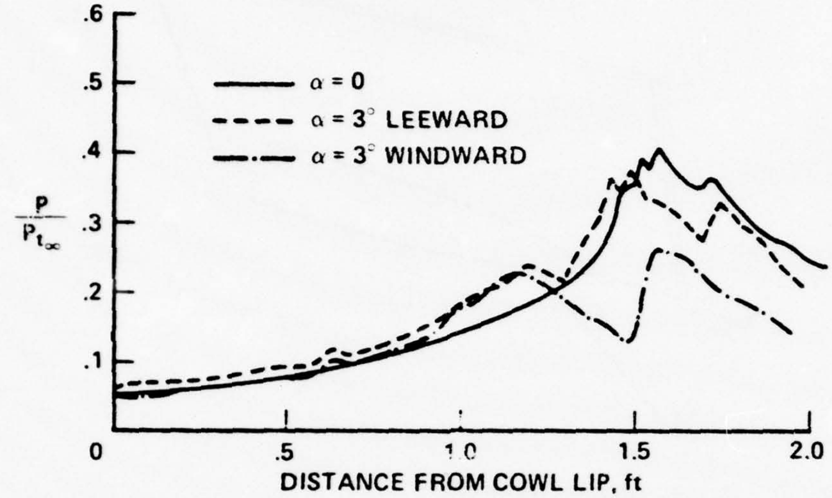


Figure 9a. Comparisons of Cowl Pressure Distributions for $\alpha = 0$ and $3°$ at $M_{\infty} = 2.65$

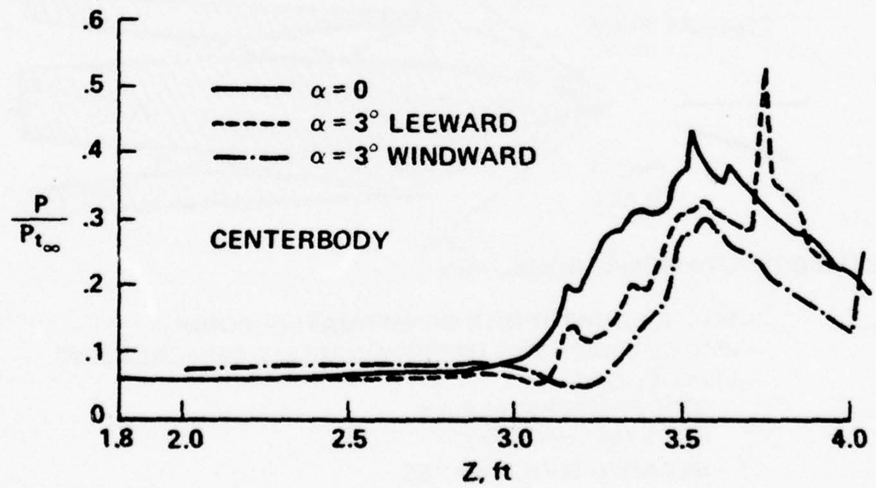


Figure 9b. Comparison of Centerbody Pressure Distributions for $\alpha = 0$ and $3°$ at $M_{\infty} = 2.65$

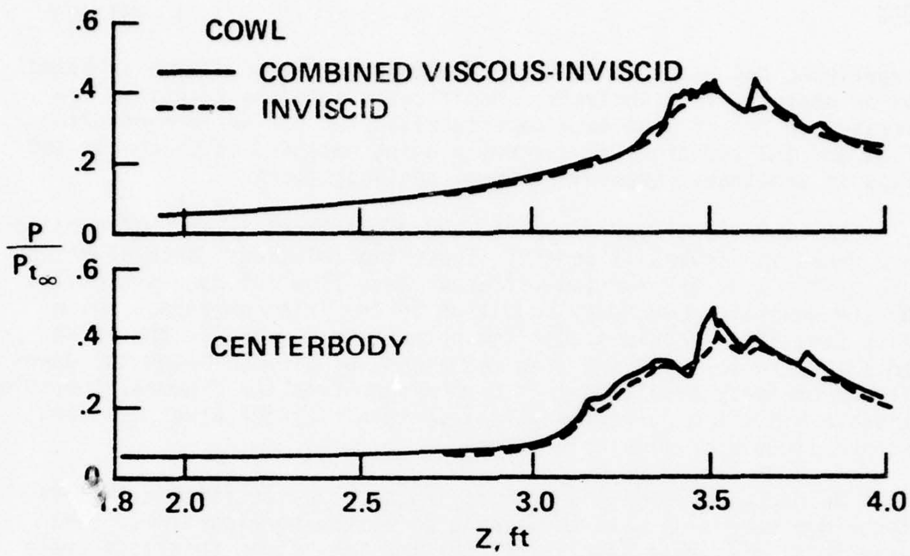


Figure 10. Comparison of Pressure Distributions at $\alpha = 0°$ and $M_\infty = 2.65$

In addition to the other limitations discussed earlier for forebody solutions using shock-capturing techniques, the restriction of supersonic flow everywhere raises particular problems for internal flows. As the Mach in the throat approaches one, the characteristic pressure oscillations that a shock capturing technique produces near shock waves often results in erroneous predictions of subsonic flow near strong shock waves. Some of these oscillations and pressure overshoots are due to the differencing algorithms in the technique. Presently, higher order differencing algorithms are being investigated to determine if they can reduce the amplitude of the oscillations, often at the expense of discrete definition of the shock waves. If higher order algorithms are not effective in eliminating erroneous predictions of subsonic flow, then elliptic-solver techniques will have to be imbedded in the overall hyperbodies flow solution to treat small, and perhaps unimportant, subsonic regions. Further, modifications are being incorporated into the code to allow inlet flow solutions with nonuniform entering flow, as would be produced by a fuselage forebody.

Time Dependent Technique - This technique for solving subsonic or supersonic two-dimensional (planar or axisymmetric) flows has been described in Ref. 9. The computer code has been set up to solve the integral form of the time-dependent Euler equations with any of several numerical differencing algorithms. Most of the current

experience has been gained using the Godunov method, which is based on an unsteady flow analysis. Modifications to the technique described in Ref. 8 have been made to allow for non-uniform entering flow, and the technique is currently being extended to calculate the flow in arbitrary three-dimensional configurations.

Two solutions* for a typical two-dimensional inlet configuration are shown in Figures 11 and 12. These two solutions, both of which are at M = 2.0, are for two different mass flow ratios, as dictated by the downstream boundary condition in the inlet passage. A low mass flow ratio requires that the normal shock wave for the inlet move forward to spill the required amount of flow to match the downstream boundary condition. As can be seen from the figures, the solutions exhibit the correct physical feature of shock wave position, stream lines and velocity vectors.

Perhaps the largest degree of inaccuracy associated with this technique has to do with the finite differencing algorithms. The effects of different algorithms upon the flow field solutions are presently being investigated. A question closely coupled with accuracy is how much computer time is required for a solution that is sufficiently accurate for engineering purposes. The solutions shown here required on the order of 10 minutes of CDC 7600 computing time for the Godunov differencing method. An assessment of computing time versus accuracy will be made as other differencing algorithms are incorporated into the code.

Coupling Concepts - Individual forebody or inlet solutions are of little use for interference effects unless they can be coupled together in some fashion. For completely supersonic flows and some transonic flows, coupling is a matter of bookkeeping, i.e., using the output of one solution as the input and boundary conditions of the other. However, for subsonic flow where upstream effects are possible, the inlet flow could possibly affect the forebody flow; thus under such conditions an iterative approach will have to be employed. As suggested in Figure 13, overlapping computational domains will be required with interaction effects resulting from the imposition of boundary conditions along common boundaries.

A postulated iterative scheme would proceed as follows. The first step would be to obtain a solution for an isolated fuselage in the fuselage domain. Output of this solution would be used to define the boundary conditions for an isolated inlet solution in the inlet domain. It should be noted that the inlet and fuselage domains are three dimensional, and the inner side of the inlet domain would

*The author is indebted to F. P. Kirkland of General Dynamics, Fort Worth Division for these solutions.

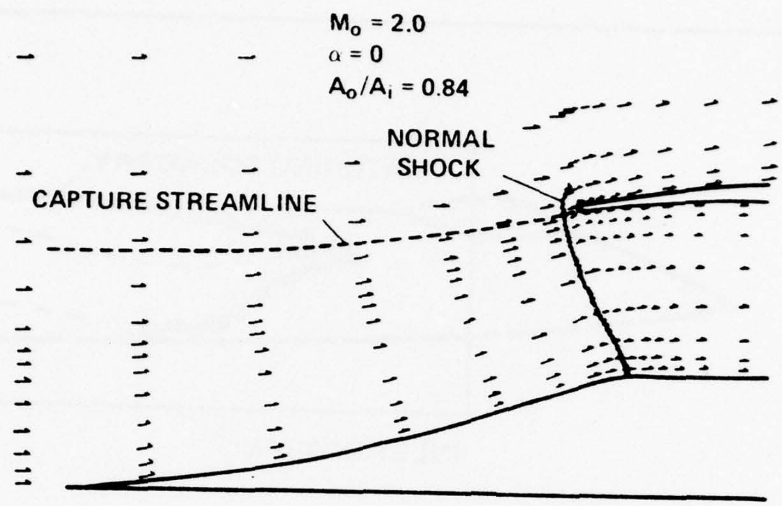


Figure 11. Flow Field Vectors for High Mass Flow Ratio

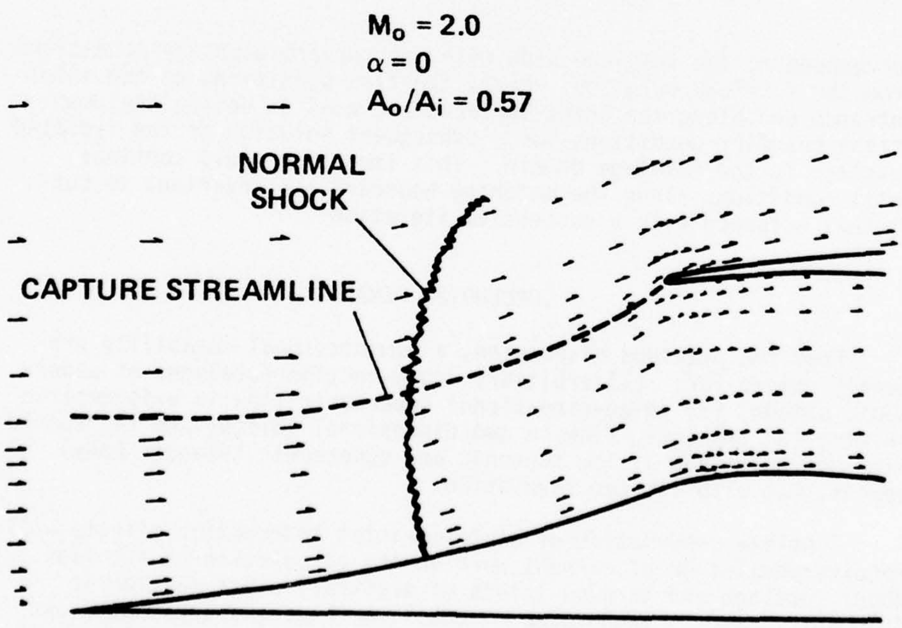


Figure 12. Flow Field Vectors for Low Mass Flow Ratio

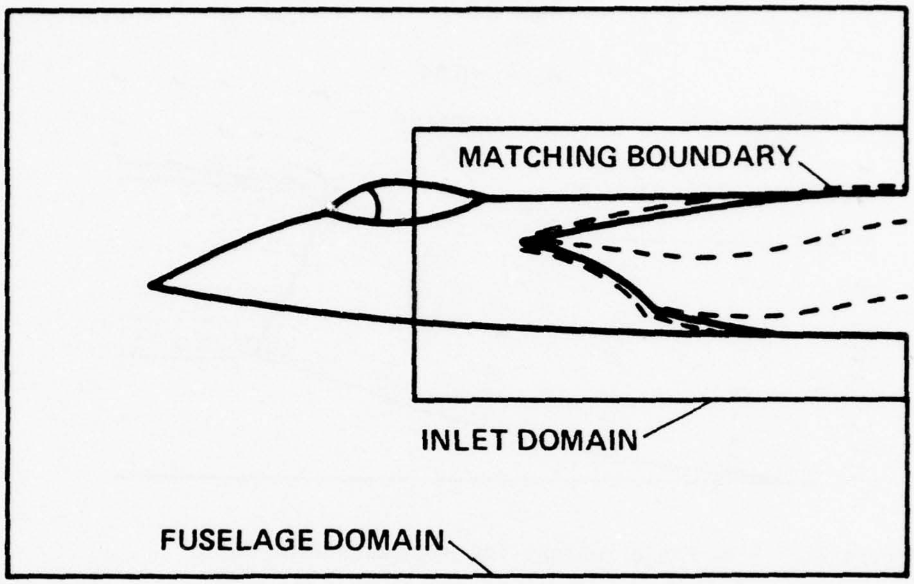


Figure 13. Coupling Concepts

correspond to the fuselage side with appropriate boundary conditions from the fuselage solution. Next, the flow conditions at the inlet entrance and along the inlet surfaces are used to define the downstream boundary conditions for a subsequent solution of the isolated fuselage in the fuselage domain. This iteration would continue until conditions along the matching boundary are invariant to sufficient accuracy with a successive iteration.

CONCLUDING REMARKS

From the previous discussion, a computational capability presently exists for: (1) arbitrary cross-section fuselages at supersonic speeds; (2) three-dimensional supersonic flow in axisymmetric inlets; (3) transonic flow in two-dimensional inlets; and (4) complete aircraft solutions at low subsonic and supersonic (slender body) speeds, but with limited resolution.

Complete computation of airframe-inlet interaction effects will require completion of current work on the calculation of the flow about fuselage and through inlets of arbitrary cross section at transonic speeds. Techniques of coupling these two types of flows where large upstream (or elliptic) effects are present will have to be developed. Coupling methods for completely supersonic flows are currently being developed.

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DISCUSSION

Welliver: (Boeing Military Airplane Division)

I have two questions. One is, in your hypersonic program that you are talking about, you said that the free-stream Mach number had to be greater than 2. Is that because you need an attachment, or what is the reason?

PRESLEY:

Practically; of course, that approximation is good only at hypersonic speeds, so M - 2 represents a lower value. Some people say you can stretch the approximation, but the basic equations of the technique are not applicable at low supersonic speeds. That is the primary reason; plus, the shock wave must be attached. You could run a detached shock wave by combining with a blunt body starting solution, but we are not doing that right now.

WELLIVER:

The other question, in your inlet program where you were talking about the bleed, I did not understand, do you include bleed in the program now? And does it run automatically with bleed, or do you patch that in somehow, in terms of defining boundary layer effects?

PRESLEY:

I should have made that point clear. We are not coupling together viscous and inviscid solutions automatically. We obtain a shock capturing solution, the same as Sieberg and Hickok do, obtain a method of characteristics solution to get an inviscid pressure distribution. We then input that pressure distribution into a turbulent boundary layer program, calculate the boundary layer with a chosen bleed distribution, and determine displacement thickness. We then add that displacement thickness on to the original contours and input these corrected contours into the inviscid code and recalculate the inviscid flow. We presently do this entire process in the computer by writing files. It was initially questionable whether or not we could take the displacement thickness distribution straight out of the boundary layer code, add it to the original contours without any artistic manipulation by an engineer and obtain a subsequent solution. However, the technique works. As a general rule, for finely <u>tuned</u> axisymmetric inlets where throat Mach numbers of 1.4 are being approached, you have essentially to remove all of the boundary layer. This means, of course, you drive the displacement thickness to zero.

HEISER: (Arnold Engineering Development Center)

I understand the physics of the problem better than I understand the administrative situation! What steps are being taken by NASA to make this material available to people who don't have access to the Illiac IV or have not participated in the development of the procedures

PRESLEY:

Well, anybody who wants a copy of our 3-dimensional inlet solution can have it now. Everything I am describing for you now is

computer codes running on a 7600. None of this work involves the Illiac.-4. Essentially, anyone who has a CDC 7600 can use these same codes, and he will suffer about a 4-fold or 5-fold increase in computer time to get the solution. The real difficulty arises when, for some of these codes, you must have 7600 class machines to have the storage and the computational speed to obtain a solution in a reasonable time. For a typical solution of a fuselage forebody at transonic speeds, we are estimating that about an hour of 7600 time will be required. Eventually, Ames is going to have to bite the bullet (and I don't think they have really done it, but perhaps Dick Petersen can answer these questions) on outside use of our computational capability. We are going to have to face the questions of making our computers, our programs, and our program maintenance capability available to outside users either on a fee basis or on a cooperative basis where, within a reasonable length of time, the results can be released and go to public domain. Currently, if anyone wants a 7600 code, we will provide a deck and do our best to tell you how to use the code. However, we are not in the business of documenting and maintaining these codes other than to write up brief user manuals on how to use the codes. The administrative structure is not set up yet, as I understand, on how to make Ames' total computational capability available to the public.

GOETHERT: (The University of Tennessee Space Institute)

For some flight regimes, the range of large angles of attack is particularly important, as we know. My question is, to what extent is your method developed to calculate the flow field on parts of the aircraft--on the forebody, particularly, I think at high angles of attack, 40 degrees or so, where you get highly separated flow? Do you see any prospect for such flows, such as around forebodies where you don't have symmetrical solutions, but you have multiple stability when the vortex moves to one side of the body?

PRESLEY:

The only thing we are working on that can give us information on vortex separation is a Navier-Stokes code. We are hopeful that the transonic Navier-Stokes code using explicit technique will reproduce the right physics of that problem. We are trying it right now for a missile forebody which we will be testing in a transonic tunnel at a very high angle of attack. We know already, from compressor solutions that were done for the Lewis Research Center, that the code does a very good job predicting separated flows at fairly low effective angles of attack. The other problem you get into, and we will have a lot of problems, is that no Navier-Stokes code is any better than the physics that we put into it. The understanding of turbulence modeling and separated flows is at a very primitive stage right now.

GOETHERT:

The specific problem which I am referring to is not only that you have an oscillating vortex from one side to the other, but really an asymmetric steady state condition which might jump from one side to the other unpredictably.

PRESLEY:

We will try to solve that type of flow with the Navier-Stokes technique. We have no major effort going on in non-Navier-Stokes techniques that require more physical insight to solve these kind of problems.

LOTH: (West Virginia University)

Do I understand you include a normal shock wave in your computational method as the last shock?

PRESLEY:

I did not want to give the impression that the shock wave I was describing was a normal shock wave; it is just a very strong oblique shock wave. The shock-capturing technique cannot capture a normal shock wave, since you cannot treat subsonic flows with this particular computational scheme. To expand upon your question, proper resolution of shock waves is one of the problems with the shock-capturing technique. Across several mesh points, the Rankine-Hugoniot jump conditions for a shock wave are reproduced. However, entropy never enters into the shock capturing calculations. It is there implicitly by the fact that you have the right pressure density jump across the shock wave, so you can calculate what it is. The real problem arises in determining what the entropy is on the walls, and that is the largest uncertainty with the shock-capturing technique. You essentially have to hold entropy constant on the walls as a boundary condition.

SECTION III

SPECIAL PROBLEMS

EFFECTS OF OVER-THE-WING PYLON-MOUNTED ENGINES ON TRANSPORT AIRPLANE PERFORMANCE

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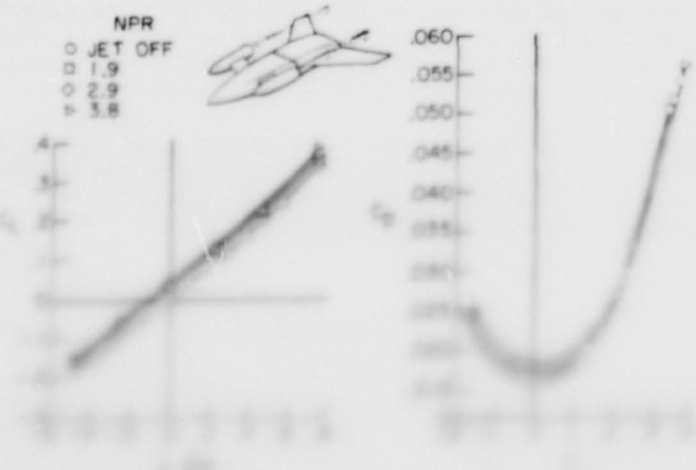
ABSTRACT

Increasingly stringent requirements for fuel economy and noise reduction are forcing the airplane designer to consider unconventional locations for the engines on future transport airplane configurations. An engine location that shows promise for reducing noise and increasing airplane performance is over-the-wing on pylons mounted on the upper-wing-surface. This paper shows with theoretical calculations and experimental data the effects of jet exhaust from over-the-wing pylon-mounted nacelles on the lift and drag characteristics of transport-type wing-body configurations. The variation of these jet-induced interference effects with free-stream Mach number, nozzle exit total-pressure ratio, and nozzle location is discussed.

A technique for minimizing the installation penalties associated with installing the nacelles on the airframe is described. The results of using this technique of streamline contouring the nacelle and pylon on the interference lift, drag, and critical Mach number are shown.

engine nacelle forward and above the wing, a potentially quieter airplane will result from the shielding effect of the wing. Improved take-off and landing performance can also result if the jet exhaust is deflected onto the wing upper surface and flap system, producing substantial increases in maximum lift in a manner similar to upper-surface-blown nacelle configurations. Since the engine-nacelle is mounted relatively high above the wing, however, scrubbing drag penalties associated with the jet exhaust can be minimized or eliminated at cruise and climb conditions.

The interferences due to installing the over-the-wing propulsion system on the airframe can be broken into two components: those interferences caused by the jet exhaust flow and those interferences caused by the nacelles and pylons. An investigation reported in Reference 1 has shown that the jet exhaust flow from over-the-wing nacelles can reduce the drag-due-to-lift and increase the lift of the wing and fuselage of an airplane configuration. (See Figure 1 for typical results of this investigation.) These effects of the jet exhaust flow on wing-body lift and drag increased with increasing ratio of nozzle total pressure to free-stream static pressure. The results of Reference 1 also show an increase in zero lift drag with jet total pressure ratio. This phenomenon caused by the jet scrubbing the wing occurred because the nacelle locations of Reference 1 were all relatively close to the wing. However, the favorable effects of jet blowing on drag-due-to-lift were of



sufficient magnitude to produce a net improvement in drag at lift conditions representative of climb and cruise.

Experimental investigations and analytical calculations have shown that installing the nacelles and pylons on the airplane configuration can produce large unfavorable effects on airplane lift and drag. Reference 4, for example, indicates that these unfavorable effects may be larger for the over-the-wing nacelle location than for the conventional under-the-wing nacelle location. It has been suggested (Ref. 3) that these unfavorable interference effects may be minimized by contouring the nacelles and pylons to make them essentially invisible to the flow about the wing and fuselage. As a result, the nacelles and pylons could be installed for only their skin friction drag.

The purpose of the present paper is twofold. Since Reference 1 was published, there have been additional investigations, both experimental and analytical, of the effects of the jet exhaust from over-the-wing nacelles. (See References 4 through 8, for example). The present paper will, therefore, summarize the results of several of these recent investigations which show the effects of the jet exhaust flow from over-the-wing engines on airplane wing-body lift and drag characteristics at conditions representative of the take-off, landing, climb, and cruise. These results and the analytical methods of References 2 and 3 will be used to show, in particular, the effects of nozzle total pressure ratio, free-stream Mach number, and nacelle location on the jet-induced interference lift and drag. Secondly, the present paper will show the effects of nacelle and pylon contouring on the performance of a high-wing transport configuration using analytical calculations and some limited experimental data.

SYMBOLS

b	wing span
C_D	drag coefficient
ΔC_D	difference in drag coefficient at scheduled NPR and at flow-through NPR
$\Delta C_{D,nac}$	increment in drag coefficient due to nacelles and pylons
C_L	lift coefficient
ΔC_L	difference in lift coefficient at scheduled NPR and at flow-through NPR

C_p	pressure coefficient
c	local wing chord
c_ℓ	section lift coefficient
D	nozzle exit diameter
M	Mach number
NPR	ratio of nozzle exit total pressure to free-stream static pressure
V	velocity
WB	wing-body
WBN	wing-body-nacelle
Δx	longitudinal location of nozzle exit relative to wing leading edge, positive aft
y	spanwise location
z	vertical location, positive up
α	angle of attack
γ	ratio of specific heats
η	fraction of wing semispan, $\frac{2y}{b}$

Subscripts

AVE	average
CRITICAL	critical
i	induced
je	jet entrained
jet	jet
∞	free stream

THEORETICAL METHODS

In this paper and for the past several years at the Langley Research Center, the methods of References 4 and 5 have been used to predict the effects on airplane aerodynamic performance of jets exhausting from over-the-wing engine nacelles. The algorithm developed by Putnam (Ref. 4) is based on a vortex lattice representation of the wing lifting surface and line sink-source distribution to represent the effects of the exhaust jets. (See Fig. 2.) The method assumes that the flow external to the jet exhaust is steady, irrotational, inviscid, and incompressible. It was also assumed that the jet is not deflected by the free stream, the jet exhausts do not intersect or wash the wing, and the jet cross-sectional shape is not distorted by the flow field of the wing or by any cross flow components of the free-stream velocity. The line sink-source distribution consisting of a series of triangular elemental singularities was located on the longitudinal axis of the jet. The strength of the line sink-sources was adjusted to give the predicted inflow velocity caused by the entrainment effect of the jet.

Lan (Ref. 5) has developed a method of predicting the jet induced effects caused by blowing over a wing that accounts for the effects of the presence of the jet on the wing flow field, the presence of the wing on the jet flow, the effects of difference in Mach number of the jet and the free stream, and the effects of the jet entrainment. (See Fig. 3.) Lan used a quasi-vortex lattice

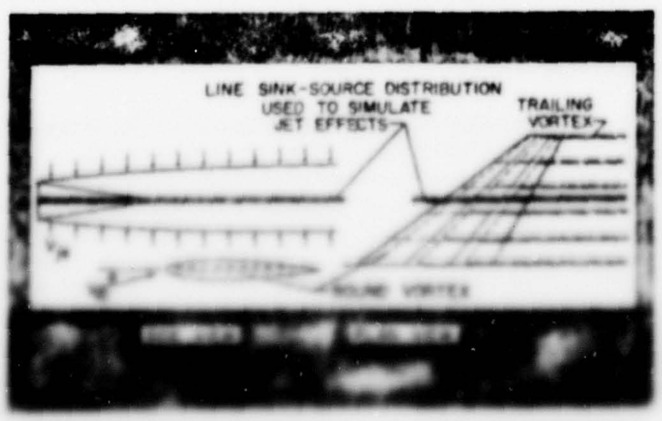


Figure 2. Analytical model of reference 4 for predicting jet exhaust interference effects.

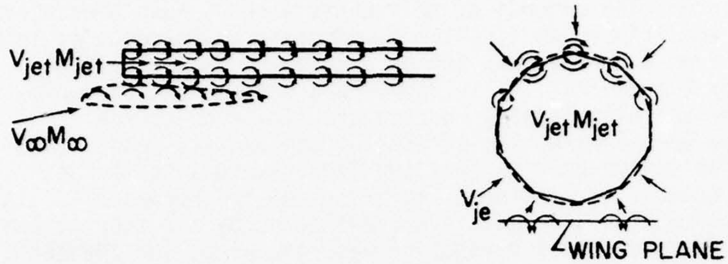


Figure 3. Analytical model of reference 5 for predicting jet exhaust interference effects

representation of the wing surface. To account for the Mach number nonuniformity between the jet and the external flow, two vortex sheets were used to represent the jet exhaust flow: one to account for the perturbation in the outer flow and the other for the jet flow. Jet entrainment effects are accounted for in this algorithm by specifying a normal velocity distribution on the outer vortex sheet representing the jet exhaust. The main assumptions used in developing this theory are that the flow perturbations, both inside and outside the jet, satisfy the Prandtl-Glauert equation, and all boundary conditions have been linearized, the jet is either of circular or rectangular shape with constant cross section and constant properties in the unperturbed flow for the purpose of the interaction calculations; and no fuselage, nacelles, or wing thickness are included in the solution.

In Figure 4, the predictions of the methods of References 4 and 5 are compared with the experimental data of Reference 6. The experimental data were obtained on the wing-body configuration shown on the figure and the nacelles were nonmetric. The data then are only for the jet interference on the wing and fuselage which are compatible with the theoretical methods. Both theories predict essentially the same lift increment due to jet blowing and correctly predict the variations with angle of attack and with the ratio of jet velocity to free-stream velocity. The methods, however, under-predict the magnitude of the experimental lift increment. Both theoretical methods predict a favorable increment in drag due to the

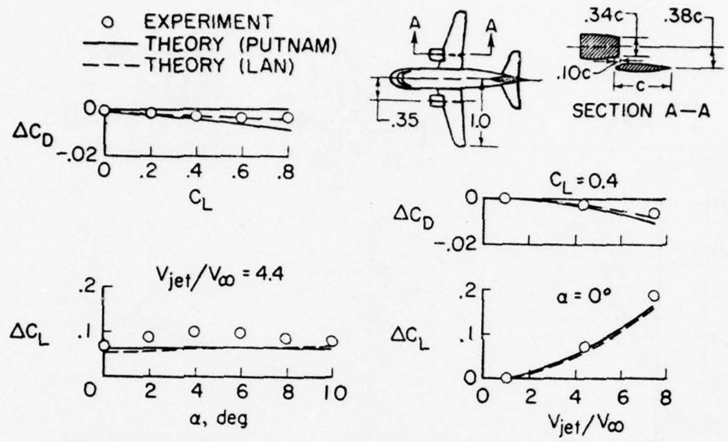


Figure 4. Comparison of experiment and theoretical predictions of Reference 4 (Putnam) and Reference 5 (Lan). $M_\infty \approx 0.2$.

the jet flow and the wing flow and the neglect of the Mach number nonuniformity between the jet and the external stream.

DISCUSSION

Jet Induced Effects on Airplane Performance

To illustrate the effects of the jet exhaust from over-the-wing nacelles on airplane performance, results from Reference 6 are presented as Figures 5 and 6, and a photograph of the experimental apparatus and results of Reference 7 are presented as Figures 7 through 9. Note that these experiments were conducted with the nacelles nonmetric so that the lift and drag shown are only for the wing and fuselage. These figures show the effects of jet blowing for nozzle pressure ratios representative of a turbofan engine with a fan total pressure ratio of 1.6. (See fig. 10.) The condition used as a basis for comparison is the nozzle pressure ratio for the nozzle total pressure equal to the free-stream total pressure (i.e., essentially the wind tunnel model flow-through condition for the nacelles). Note, also, that in some cases V_{jet}/V_∞ is used instead of the nozzle pressure ratio. In Reference 5, an effective velocity ratio is used to account for the effects of variable density between the jet stream and the external stream in the theoretical calculations. This effective jet velocity ratio is equivalent to the square root

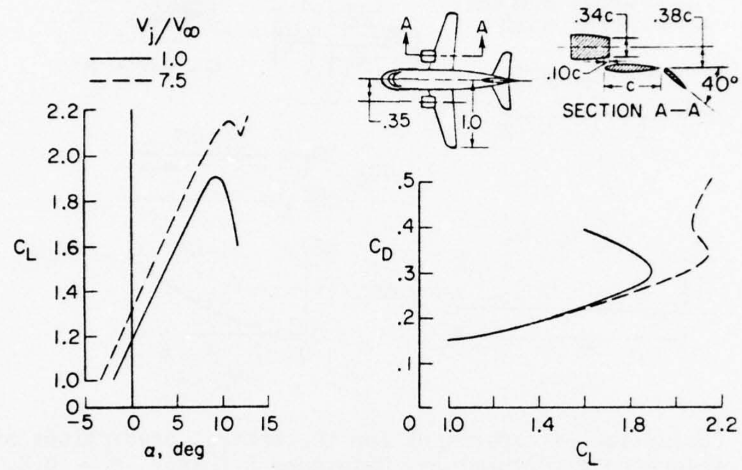


Figure 5. Effects of V_{jet}/V_∞ on lift and drag of configuration of Reference 6 with flaps deployed. $M_\infty \approx 0$

of the ratio of dynamic pressures of the two streams. Using this definition, a relationship between velocity ratio and nozzle total pressure ratio can be obtained and is given by the following equation:

$$\left(\frac{V_{jet}}{V_\infty}\right)_{effective} = \frac{M_{jet}}{M_\infty}\sqrt{\frac{\gamma_{jet}\,(NPR)}{\gamma_\infty}}\left[1 + \frac{(\gamma_{jet} - 1)}{2} M_{jet}^{\;2}\right]^{-\frac{\gamma_{jet}}{2(\gamma_{jet} - 1)}} \qquad (1)$$

At take-off and landing conditions (Fig. 5), the jet exhaust from over-the-wing nacelles causes an increase in lift at zero angle-of-attack and a large increase in maximum lift coefficient without any significant change in drag coefficient at low to moderate lift coefficients. This increase in maximum lift results from the increased circulation induced by the jet. It may be possible to obtain further increases in maximum lift by deflecting the jet down onto the wing upper surface and onto the deflected flap. In fact, Reference 8 showed just such an increase in circulation lift resulting from deflecting the jet down on the wing of a supersonic transport model at low speeds.

During climb, the jet exhaust can induce a significant reduction

in drag at lift, as shown in Figures 6 and 8. The data for a velocity ratio of 4.4 on Figure 6 is representative of engine operation at the start of climb. The data at a velocity ratio of 7.5 is shown only to illustrate effects on increasing this parameter on lift and induced drag. (See also Figure 4.) At a lift coefficient of 0.7, the jet exhaust induces a 6 percent increase in lift-drag ratio with a velocity ratio of 4.4 (a decrease in drag coefficient of .0030) and 20 percent increase with a velocity ratio of 7.5 (a reduction in C_D of 0.0110). Increasing Mach number causes a reduction in the favorable interference effects on drag. (See Fig. 8 and also note change in C_D scale from Fig. 6.) The data of Reference 7 shown in Figure 8 shows only a reduction of 0.0004 to 0.0008 in drag coefficient at a Mach number of 0.5 for the configuration shown in Figure 7. At cruise conditions (Fig. 9), the jet exhaust flow has essentially no effect on the drag of the configuration of Reference 7 with over-the-wing nacelles. This decrease in the favorable interference effect on drag with increasing Mach number results primarily from the decrease in the ratio of jet velocity to free-stream velocity. As shown in Figures 4 and 6, the magnitude of the beneficial interference effects increases rapidly with increasing velocity ratio. Examination of equation (1) shows that the effective velocity ratio is inversely proportional to the free-stream Mach number and directly proportional to the square root of the nozzle total pressure ratio. Since the nozzle pressure ratio only increases slowly with free-stream Mach number (Fig. 10), the

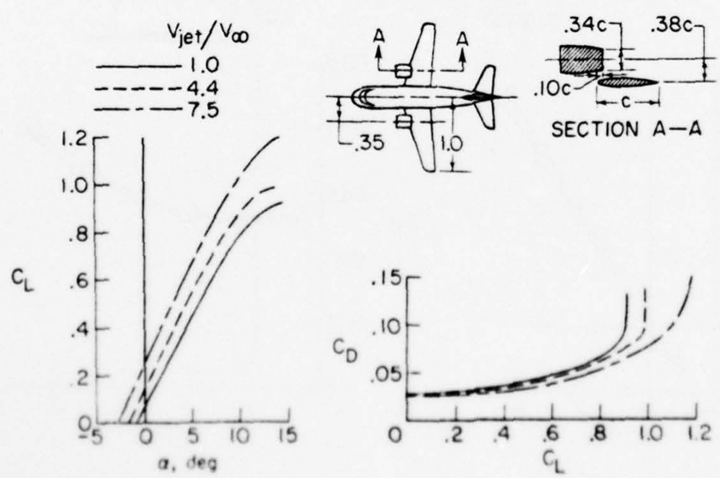


Figure 6. Effects of V_{jet}/V_∞ on lift and drag at climb conditions for configuration of Reference 6

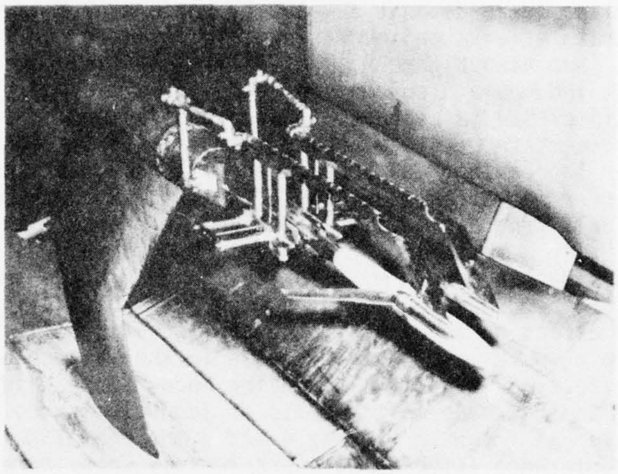


Figure 7. Experimental apparatus of Reference 7

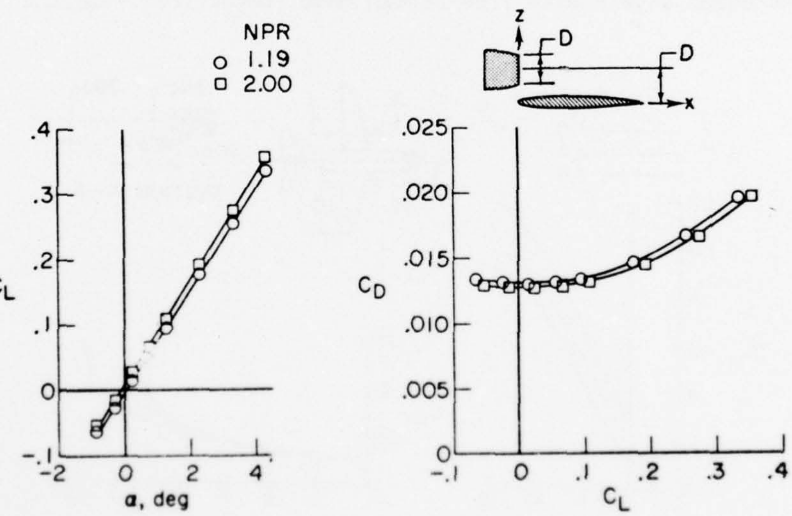


Figure 8. Effect of NPR on lift and drag for configuration of Reference 7 at M_∞ = 0.50

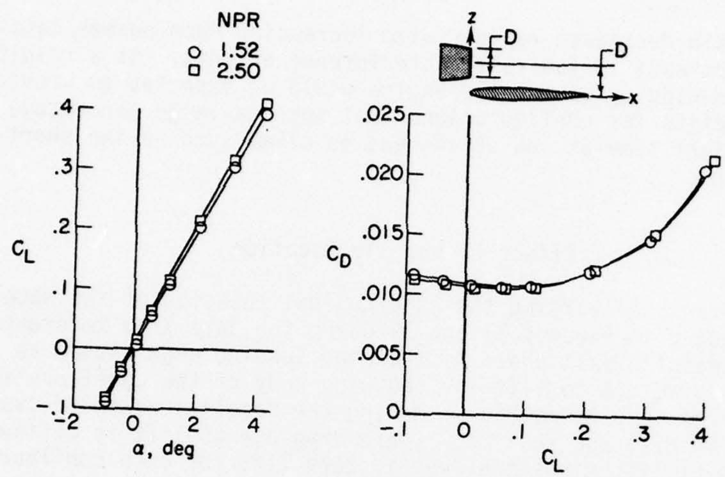


Figure 9. Effect of NPR on lift and drag for configuration of Reference 7 at M_∞ = 0.80

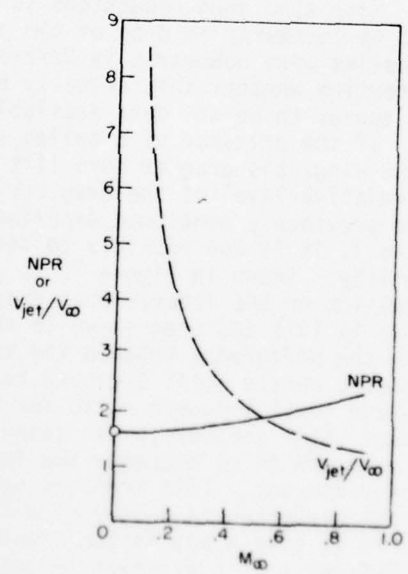


Figure 10. Nozzle pressure ratio and velocity ratio for a typical engine for a transport airplane

velocity ratio decreases rapidly with increasing Mach number, causing the rapid decrease in favorable interference effects. As a result, the over-the-wing pylon-mounted engine would be expected to provide greater benefits for configurations that spend a large percentage of their flight time at low speeds and in climb such as the short-haul airplane.

Effect of Nacelle Location

The effects of varying the longitudinal location of the nacelles are illustrated in Figures 11 and 12 using the data from Reference 7. Moving the nacelle exit ahead of the wing leading edge causes an increase in drag due to lift for the wing-body of the configuration (i.e. nacelles were nonmetric). Moving the nacelles rearward causes a decrease in drag due to lift. (Here drag-due-to-lift is defined as the drag at lift minus the drag at zero lift for each configuration. Because of model-support-system interferences in the data of Reference 7, it was not possible to determine the effect of nacelle location on drag at zero lift.) These changes in drag-due-to-lift may be caused by changes in wing circulation due to the jet exhaust flow or possibly by the presence of the nacelle effectively changing the camber of the wing. Note also that reductions in drag on the wing may be counteracted by increases in drag of the nacelles. However, because the nacelles were nonmetric in Reference 7, data are not available to determine whether this actually happens. (At present, there does not appear to be any data available which will clarify the situation.) If the presence of nacelles effectively changes the camber of the wing, the drag at zero lift would change and thereby change the relative level of the drag curves. Unfortunately, because of the previously mentioned experimental difficulties of Reference 7, it is not possible to determine the validity of this possibility. Shown in Figure 12 is the effect of nacelle longitudinal location on the interference drag due to jet blowing. (The increments in lift and drag shown in this figure and others to follow are the difference between the value of the coefficient at the scheduled nozzle total pressure ratio presented in Figure 10 and the nozzle total pressure ratio for nozzle total pressure equal free-stream total pressure.) In general, the effect of moving the nacelles rearward is to decrease the favorable effects of jet blowing on drag coefficient. This trend is opposite that shown in Figure 11 for the variation in drag-due-to-lift with nacelle longitudinal location. It is also opposite the trend predicted by the theory of Lan. In Reference 4, a decrease in jet-induced drag increment was predicted for a forward movement of the nacelle similar to these experimental results. However, both theoretical methods (Refs. 4 and 5) predict only small changes in drag with longitudinal location, and the absolute magnitude of the differences in interference drag predicted by the two methods would be small. In any case, since

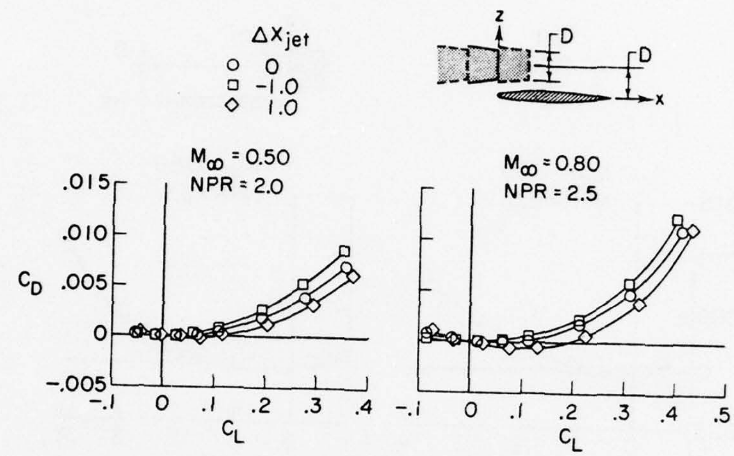


Figure 11. Variation of interference lift and drag increments with NPR for configuration of Reference 7

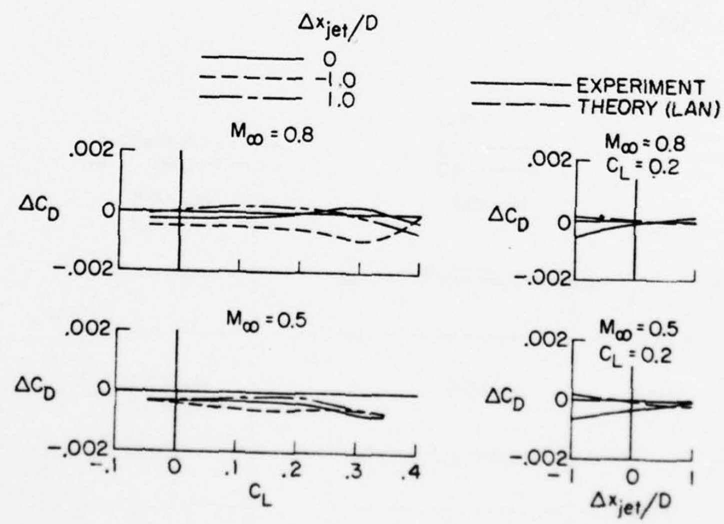


Figure 12. Effects of nacelle longitudinal location on interference drag due to jet exhaust flow for configuration of Reference 7. η_{jet} = 0.25, z_{jet}/D = 1.0

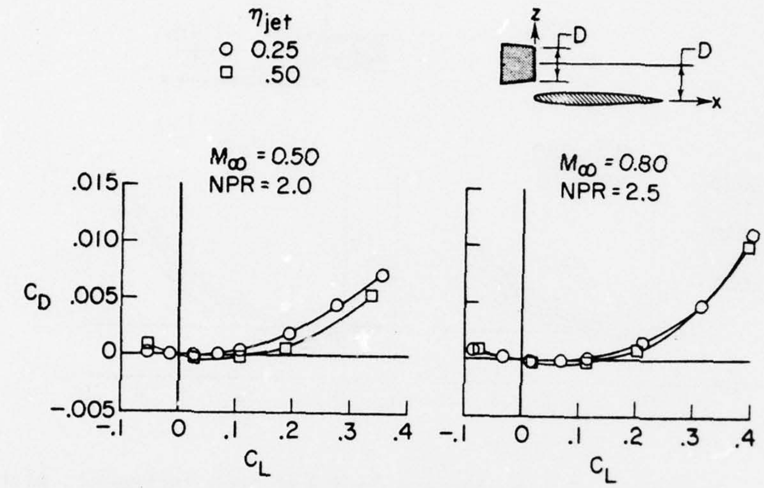


Figure 13. Effects of nacelle spanwise location on drag-due-to-lift for configuration of Reference 7. $\Delta x_{jet}/D = 0.0$, $z_{jet}/D = 1.0$.

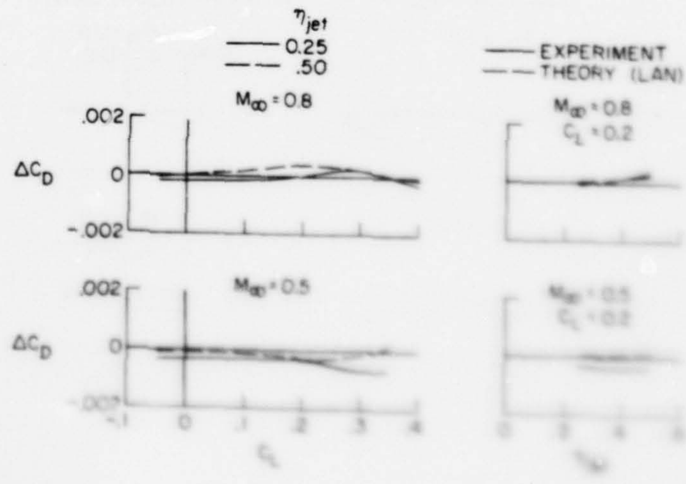


Figure 14. Effects of nacelle spanwise location on interference drag due to jet exhaust flow for configuration of Reference 7. $\Delta x_{jet}/D = 0.0$, $z_{jet}/D = 1.0$.

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the variation of jet induced effects shown in Figure 12 with nacelle longitudinal location are opposite to those shown on Figure 11, the changes in drag-due-to-lift with nacelle longitudinal location are not caused by the jet exhaust flow. Complete understanding of these phenomena awaits further experimental and analytical research.

The effects of nacelle spanwise location in the wing-body drag of the configuration of Reference 7 are shown on Figures 13 and 14. Moving the nacelle outboard caused a reduction in drag-due-to-lift. The magnitude of this reduction decreased with increasing Mach number. Generally the jet induced effects were more favorable with the nacelles located inboard. (See Fig. 14.) Here the theory of Lan is in good agreement with the experimental results in both magnitude and trend.

The effects of nacelle vertical location on drag are illustrated in Figures 15 and 16. Changing the vertical location of the nacelles had only small effects on the drag-due-to-lift of the wing body at Mach numbers of 0.50 and 0.80. The favorable effects of the jet exhaust flow increased slightly as the vertical location of the nacelle was reduced until the jet exhaust began to scrub the wing. Once the jet begins to scrub the wing, further reductions in nacelle height above the wing caused an increase in drag. Again the theory correctly predicts these trends with vertical location and the magnitude of the jet induced effects on drag.

The data presented in Figures 11 through 16 have shown that there are some large effects of nacelle location on the drag-due-to-lift of the fuselage and wing of transport airplane configurations at Mach numbers representative of climb and cruise. Unfortunately, because of experimental limitations and deficiencies of the investigations conducted to date, it is not possible to determine optimum nacelle location. The effects of the jet exhaust on airplane wing-body lift and drag are adequately predicted by available analytical methods.

Effects of Nacelle Contouring

To achieve satisfactory high-speed performance with an airplane having over-the-wing pylon-mounted engine nacelles, it is necessary that any disturbance of the wing flow field by the nacelle should be in a favorable direction. (See Ref. 9.) Specifically, the nacelle must not cause any isobar unsweeping but may be allowed to cause increased isobar sweep on the wing. If the nacelles were symmetric in the plan view, the inboard wing flow would feel the nacelle curvature, and the wing isobars would unsweep as they approached the nacelle. Any resulting wing shock would thus have less sweep and higher drag. If the nacelle inboard contour follows a wing-body

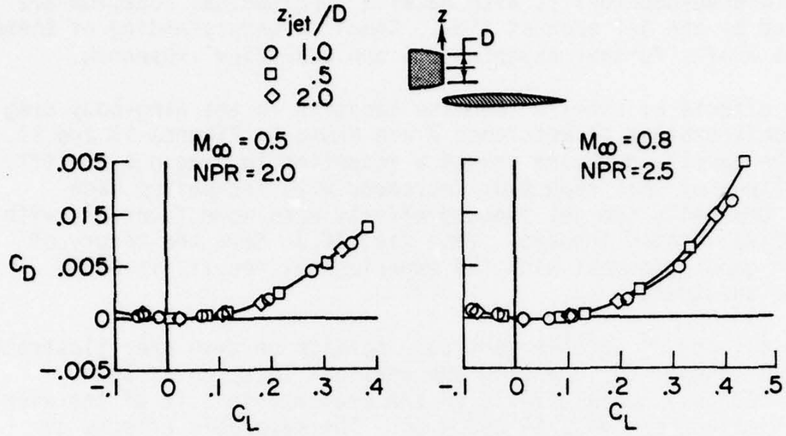


Figure 15. Effect of nacelle vertical location on drag-due-to-lift for configuration of Reference 7. $\Delta x_{jet}/D = 0.0$, $\eta_{jet} = 0.25$.

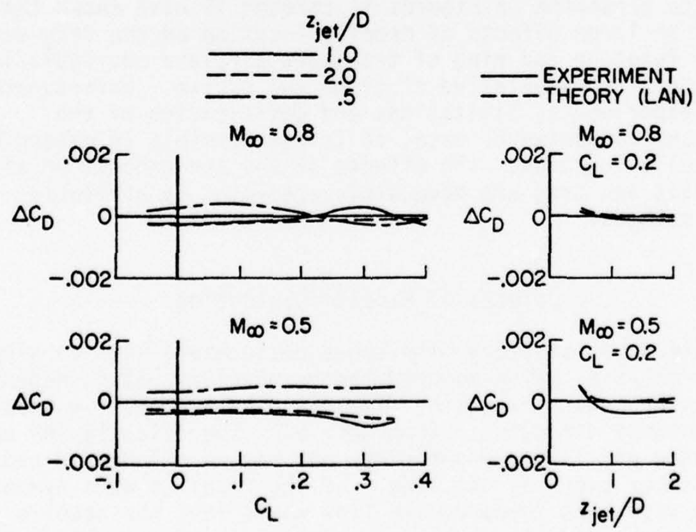


Figure 16. Effect of nacelle vertical location on interference drag due to jet exhaust flow for configuration of Reference 7. $\Delta x_{jet}/D = 0.0$, $\eta_{jet} = 0.25$.

stream sheet, the nacelle would be essentially invisible to the wing flow inboard of the nacelle and the isobar sweep and the resulting shock sweep at cruise will be similar to that of the wing-body. Since, because of nacelle thickness, the outboard contour of the nacelle cannot also follow a wing streamsheet, the outboard nacelle contour will present a boattail to the wing flow. The resulting stagnation condition at the nacelle exit on the outboard side will terminate the supersonic flow over the wing at that point, causing a shock wave to form in the wing flow. This wing shock will form with a sweep higher than the wing alone-shock as it moves outboard to join the wing-body shock pattern and result in a decrease in wave drag.

This design technique has been applied to the transport airplane concept with over-the-wing, pylon-mounted nacelles shown on Figure 17. The analytical methods of Reference 5 were used to complete the wing-body alone streamsheets and to thereby design the nacelles. Computed wing isobars with and without the contoured nacelles are shown in Figures 18 and 19 for the wing lower surface and upper surface, respectively. The wing lower surface isobars with contoured nacelle are very similar to the wing alone isobars. On the upper wing surface, the contoured nacelle causes only a small perturbation to the wing flow field inboard of the nacelle. The wing pressure isobars indicate that since the nacelle boattail is applied to the outboard side, a shock wave will form lying roughly along the line

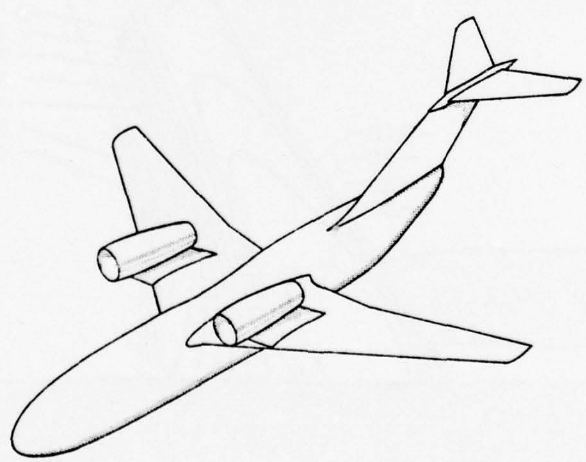
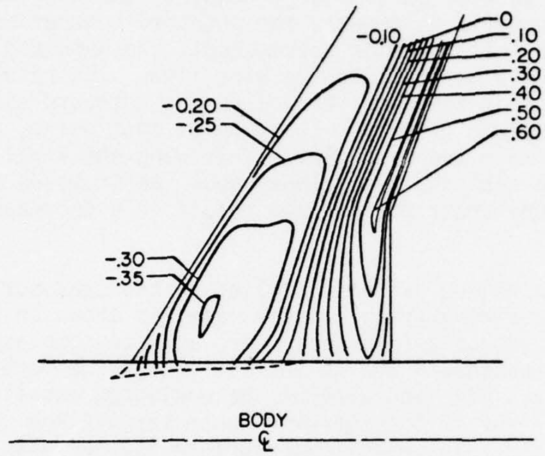
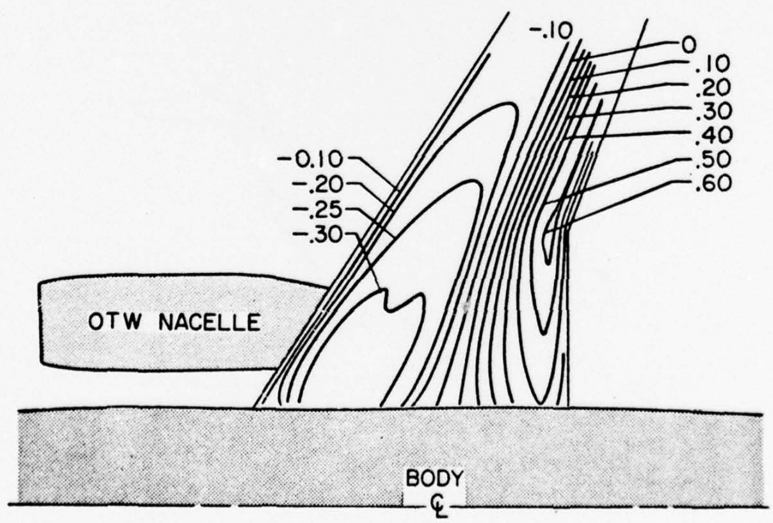


Figure 17. Conceptual transport configuration with over-the-wing pylon-mounted engine nacelles

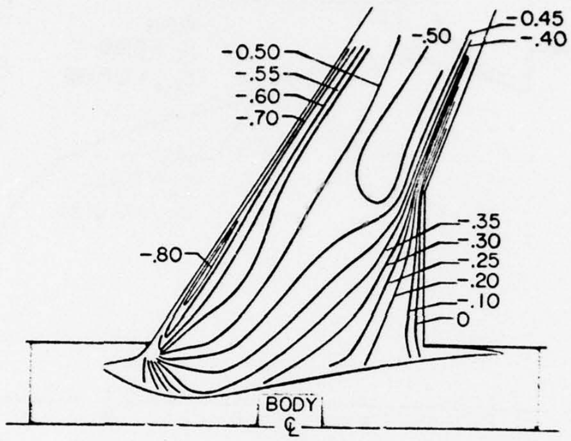


(a) Nacelles off

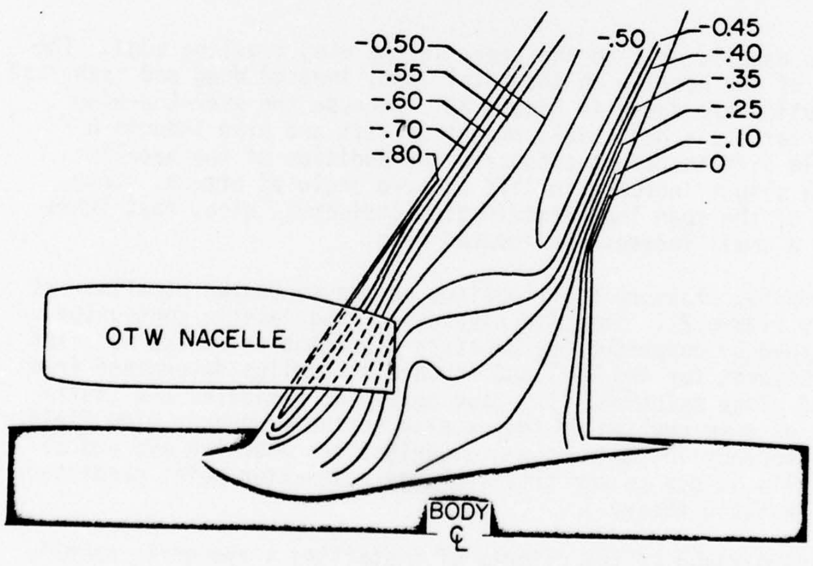


(b) Contoured nacelles on

Figure 18. Theoretical isobars on lower surface of wing of transport configuration. $M_\infty = 0.7$, $\alpha = 0°$.



(a) Nacelles off



(b) Contoured nacelles on

Figure 19. Theoretical isobars on upper surface of wing of transport configuration. $M_\infty = 0.7$, $\alpha = 0°$.

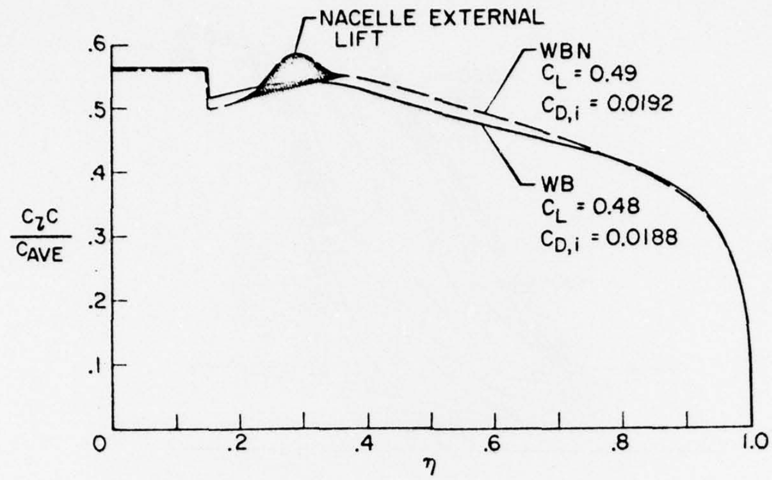


Figure 20. Effect of contoured nacelles on theoretical spanload distribution of transport configuration. $M_\infty = 0.70$, $\alpha = 0°$.

from the nacelle exit to the break in the wing trailing edge. The effects of the nacelle on the total lift, induced drag and span load distribution are shown in Figure 20. Because the over-the-wing nacelle carries a noticeable amount of lift and also induces a favorable lift increment outboard, the addition of the nacelle causes a slight increase in lift at zero angle of attack. Integration of the span load distribution indicates, also, that there is only a small increase in induced drag.

Computed pressure distributions on the contoured nacelles are shown in Figure 21. The effectiveness of the nacelle contouring can be seen by comparing the pressure distribution on the nacelles with pressures for the keel and inboard streamline determined from the wing alone solution. The good agreement indicates the invisibility of this portion of the nacelle to the wing-body flow field. The discrepancy in the pressure distributions near the aft end of the nacelle is due to the trailing edge stagnation point predicted by the inviscid theory.

A comparison of the effects of installing a symmetric nacelle and pylon and the effects of installing a contoured nacelle and pylon on the drag performance of a transport configuration is shown in Figure 22. These experimental results were taken from Reference 5. Installation of the symmetric nacelles and pylons caused a substantial reduction in the critical Mach number and an increase in

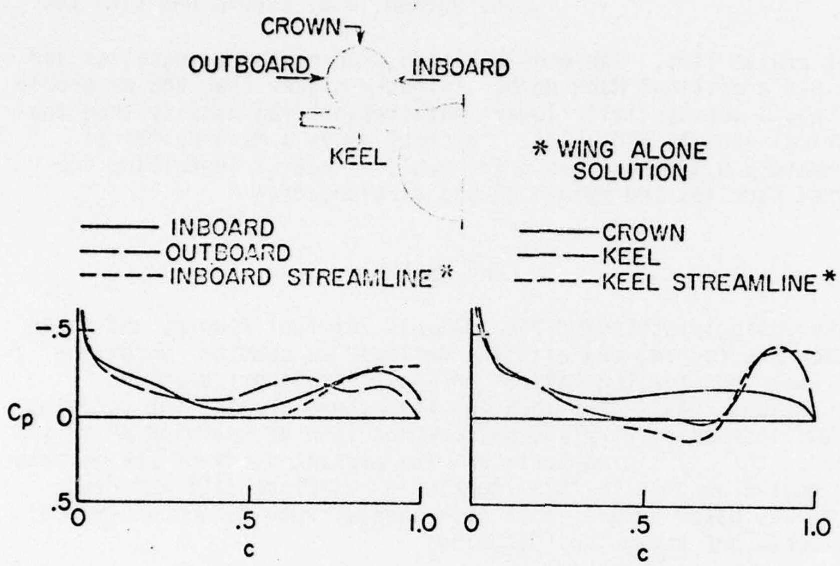


Figure 21. Theoretical pressure distributions on contoured nacelles. $M_\infty = 0.70$, $\alpha = 0°$.

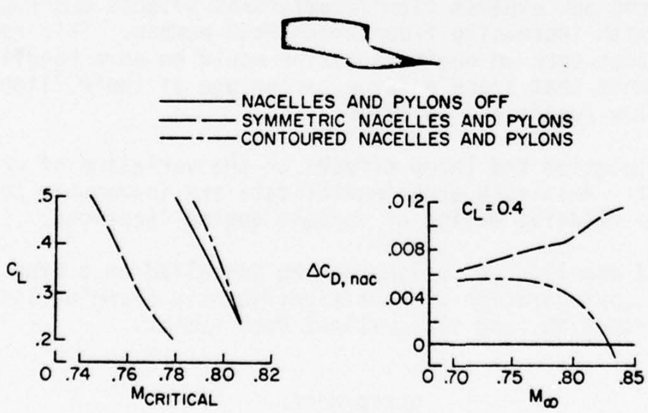


Figure 22. Effect of nacelle contouring on critical Mach number and nacelle installation drag. (Experimental data from Reference 5.)

drag at cruise lift. The configuration with contoured nacelles and pylons had a critical Mach number slightly higher than the wing-body alone and a substantially lower installation drag penalty than the symmetrical nacelle and pylon. In fact, above a Mach number of approximately 0.83, there was a favorable effect of installing the contoured nacelles and pylons on the airplane drag.

CONCLUSIONS

Increasingly stringent requirements for fuel economy and noise reduction are forcing the airplane designer to consider unconventional locations for the engines on future transport airplane configurations. An engine location that shows promise for reducing noise and increasing airplane performance is over-the-wing on pylons mounted on the upper-wing-surface. The present study of the effects of the engine nacelle in this location on airplane lift and drag performance, based on available experimental results and analytical predictions, has shown the following:

1. The jet exhaust flow from the over-the-wing nacelles causes an increase in wing-body lift and a decrease in wing-body drag-due-to-lift. These interference effects increase with increasing nozzle total pressure ratio.

2. At typical turbofan engine operating nozzle total pressure ratios, the jet exhaust flow interference effects decrease rapidly with increasing free-stream Mach number. This result implies that such an engine location would be more beneficial to airplanes that spend a large percentage of their flight time at low speeds and in climb.

3. Nacelle location had large effects on the variation of drag with lift. Available experimental data are inadequate to determine relative merits of various engine locations.

4. Contoured nacelles and pylons can be installed on a transport airplane configuration without significantly changing lift, drag-due-to-lift, and the critical Mach number.

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DISCUSSION

BRADLEY: (General Dynamics)

Were the data you presented those for an unpowered case, or was the jet flow simulated?

PUTNAM:

The calculations were done for a flow through case. The experimental data on the last point was a flow through case.

BRADLEY:

In one of your early figures, you varied the spanwise position of the nacelles. Was this in constant percent wing chord? In other words, did you move the nacelles in the chordwise position when you moved them spanwise?

PUTNAM:

Yes, when the spanwise position of the nacelle was varied, the exit was always at the wing leading edge.

BERNSTEIN: (Canadair Limited)

Was there any difference between the results with the high wing and low wing location?

PUTNAM:

The VFW worked with the low wing configuration. The research that we did was high wing, and there really has been no direct data obtained that you could compare directly. There have not been any consistent experiments conducted that I am aware of. There have been very few experimental investigations conducted and published on this kind of arrangement.

GOETHERT: (The University of Tennessee Space Institute)

You showed that in your nacelle experiments, you got low additional drag; as a matter of fact you even got a drag reduction. The question is, how do you explain that? You obviously measured this in a wind tunnel with jet flow through the nacelle.

PUTNAM:

No jet flow. That was flow through data. The inlet was just a duct, and the internal drag, I guess, was taken out.

GOETHERT:

Oh, I see, it is just a solid blockage.

MARSH: (Vought Corporation)

Did you do any measurements of stability effects as a result of the nacelle position?

PUTNAM:

In the data that we measured (we took pitching moment measurements), there were very few effects. There were some small changes in pitching moment coefficient at zero lift.

GOETHERT:

I would like to go back to my question. I am really amazed that yc have a solid body for necessarily low flow through. If you have a

real installation, you have the solid displacement greatly reduced by the flow through in this area. Therefore, the interference effects should be greatly different. So I just wonder, what is your opinion of the validity of your data comparison with full-scale flight?

PUTNAM:

Well, that was flow through. It had an inlet, a stream tube entering the nacelle, and a stream tube leaving the nacelle. And the Boeing experiment was flow through, had an inlet, and the flow was going through the nacelle. The only thing that was not simulated correctly was the mass flow. However, that is a flow through condition, so that there is inlet flow and exit flow. Does that clarify?

GOETHERT:

And you corrected for the drag of the through flow?

PUTNAM:

I do not know if they actually subtracted that internal drag term out or whether it is in that data or not.

WELLIVER: (Boeing Military Airplane Division)

The internal drag on the flow in the nacelle was taken out.

I might add another thing to that statement. There was a question here relative to over the wing, high wing, and low wing installations. There is quite an effect. We have done some testing on that, and in the spanwise direction on the low wings because of the fuselage, it has more to do with that than anything else. When the nacelle starts getting close to the fuselage, you get a channeling effect, and then things start to happen in a hurry. So there is quite an effect there, and you do not get the channeling obviously if you do a high wing version that is done carefully.

UPPER SURFACE BLOWING AERODYNAMIC AND ACOUSTIC CHARACTERISTICS

D. M. Ryle, Jr., J. A. Braden, and J. S. Gibson

Lockheed-Georgia Company

Marietta, Georgia

ABSTRACT

In the development of powered lift systems, the upper surface blown (USB) system appears relatively attractive in terms of simplicity and noise generation. However, the data base for USB is incomplete, and the problem of high speed cruise performance presents a serious technical challenge. The basic objectives of the USB Cruise Program were to identify key installation variables affecting USB cruise performance and to quantify aerodynamic effects through a systematic variation of nacelle geometric parameters. As an additional facet of the program, optimum cruise configurations were identified to evaluate needed compromises in the system when operating in the low-speed flight regime. Specialized testing designed to augment and clarify the origin and suppression of observed cruise drag penalties were also key elements of the program. Results from the transonic force tests demonstrate that jet-related aerodynamic phenomena can be effectively related to previously observed jet-flap effects. Additionally, a semi-circular (D-duct) nozzle is found to be a reasonably optimum shape from a cruise-drag standpoint. During transonic surface pressure testing, a basic phenomenon observed in the pressure measurements was a jet-induced pressure drag acting over the width of the nozzle and growing with the two-dimensionality of the jet. Additional vectoring of the jet, found at wind-on conditions relative to that found statically, is observed from the wake-flow measurements. An analytical program was used to provide support and guidance for the experimental effort and utilized a vortex-lattice technique for theoretical modeling. Good agreement is shown between the experimental and theoretical pressure distributions adjacent to and within the jet.

INTRODUCTION AND BACKGROUND

The challenge of shortened airport field lengths is an inherent part of the aircraft development cycle and is represented by several decades of related effort by the aerospace community. Generally speaking, conventional flight or cruise characteristics dictate the aircraft design requirements, with airport performance either a secondary requirement or simply a fall-out of the established design. This was acceptable in the early days of aviation until more efficient wing structures and engines brought about heavier aircraft with higher wing loadings requiring longer runways. At that point, the mechanical flap appeared to improve the maximum lift capability of the wing and thus improve takeoff and landing distance. Continuing increases in aircraft size led to multi-engine arrangements with wing-mounted engines and propellers. This accidental foray into powered lift, through the deflection of the propeller slip-stream by the trailing edge flaps, represents an evolutionary part of the high-lift development cycle. The cycle continued and was accentuated by the advent of the modern high-speed jet aircraft. High-wing loading and thin, swept-back wings are usually characteristic of these aircraft, which precipitates a dilemma for the designer since these same characteristics are detrimental to airport performance.

In response to this challenge, the aerospace industry developed sophisticated and complex high-lift systems currently utilized by such aircraft as the C-5A, B-747, DC-10, and other modern high-speed turbofan aircraft. These systems produce maximum-lift coefficients two to three times as high as those of unflapped wings; but in spite of this accomplishment, we see literally hundreds of vast airport complexes throughout the world incorporating runways of 8,000 to 12,000 feet in length. This tremendous investment in real estate and the ever-increasing pressure on these facilities generated the most significant motivation for STOL. There have been earlier STOL studies prompted by military tactical requirements, such as the Lockheed-Georgia Company investigation of STOL C-130 derivatives during the Air Force Rough Road ALPHA Program, Reference 1. While significant, these studies lack the economic impetus of commercial STOL operation which offered the promise of smaller, less expensive airfields, relief from serious air-traffic congestion, and better city center service. Additionally, STOL operation is presumed to more efficiently handle the short-haul traffic through the use of smaller aircraft operating from noncompetitive runways. These circumstances, augmented by the military AMST requirements, produced the flurry of STOL activity which peaked in the late 1960's and early 1970's and continues to this time in the NASA-Ames QSRA, NASA-Lewis QCSEE, and Air Force AMST programs.

This period produced a number of studies which were designed to

more accurately assess the economic impact of STOL, define STOL vehicle requirements, and develop the required technology. These results are typified in References 2 through 5. In general, these studies concluded that quiet, short-field aircraft can be economically viable, provide benefits to short-haul transportation, and also aid long-haul transportation through relief of airport congestion. Figure 1 is a schematic representation of the type of STOL vehicles used in these studies. This transport is designed to

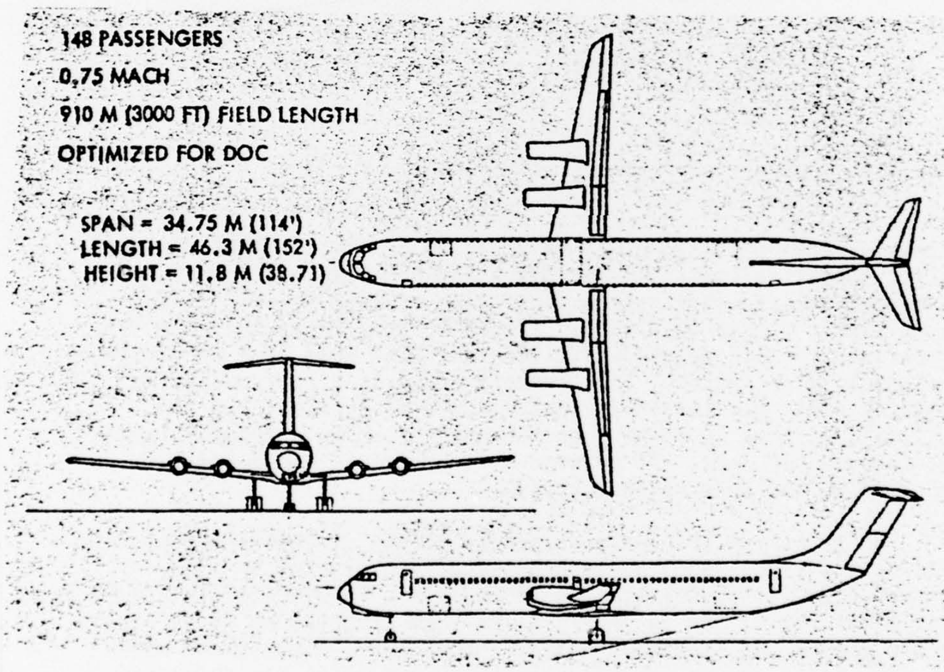

Figure 1. STOL - Systems Study Vehicle

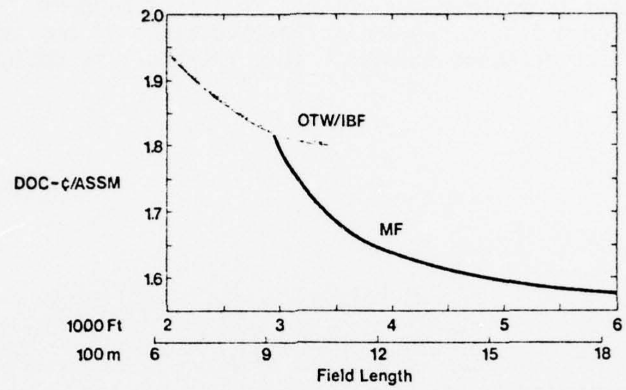

Figure 2. Comparison of High-Lift Concepts

accommodate 148 passengers and is selected on the basis of minimum direct operating cost. It employs a Lockheed-Georgia Company - developed powered-lift system consisting of over-the-wing/internally-blown-flap concepts (OTW/IBF). Figure 2 illustrates the economic comparison of this powered-lift aircraft with that of a conventional mechanical flap system. The ordinate scale is direct operating costs in cents per available seat statute mile calculated for realistic airline short-haul operations. For field lengths less then about 3000 feet, the powered-lift system is economically superior to the conventional system.

The aerospace community responded to the circumstances of this time period (1965-1975) by technologically exploring an array of powered high-lift systems, some of which are illustrated in Figure 3. This figure omits the deflected slipstream arrangement in deference to the prevailing sentiment against propellers, a sentiment which was especially strong several years back, but which has moderated as fuel costs have escalated.

Extensive wind-tunnel and analytical investigations were implemented with the primary emphasis on the low-speed, high-lift characteristics of these powered-lift systems. References 6-11 summarize results from some of this work. Although not the intent of this paper, it is appropriate to compare some of the characteristics of several systems. This is partially accomplished by reviewing Figure 4, which presents the landing configuration

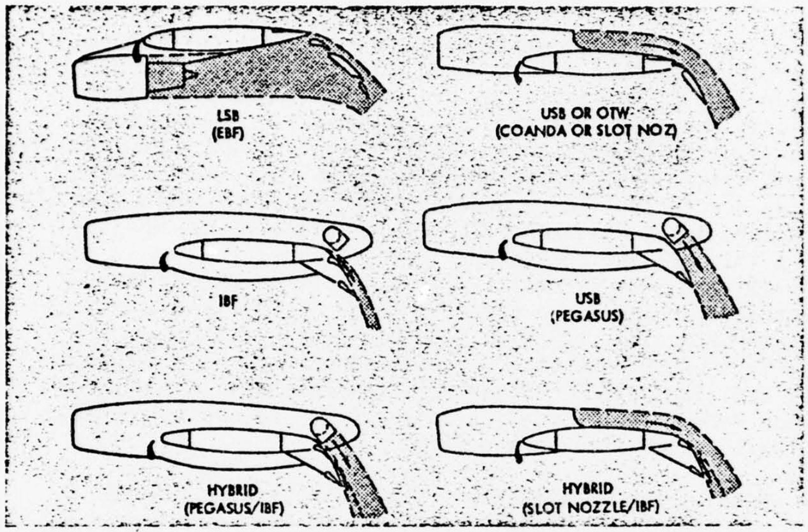

Figure 3. Powered Lift System

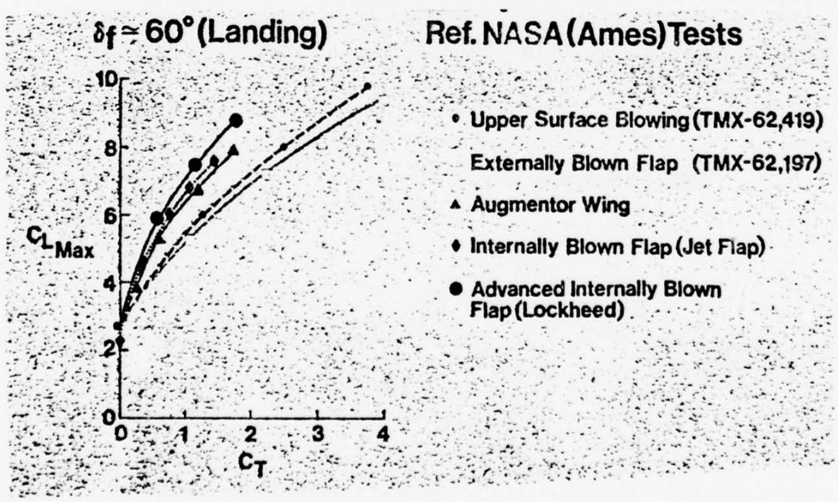

Figure 4. Maximum Lift Comparisons

maximum lift coefficient as a function of thrust coefficient for five concepts. At a typical STOL landing thrust coefficient, T_c, of

1.0, all of these systems demonstrate maximum lift coefficients of 5-7 or about twice that of unpowered high-lift systems. Furthermore, the performance of these systems are roughly comparable, which implies that the selection of a STOL powered-lift system could depend on other factors such as system weight, complexity, structural loads, and noise. The upper-surface-blowing (USB) or over-the-wing (OTW) concept utilizing Coanda turning appears attractive when these factors are considered. This is especially true when considering the difficulty and design penalty associated with noise abatement for high-powered STOL aircraft. It is intuitive that a low-fan pressure ratio engine and a nozzle shielded by the wing are conducive to airport noise reduction. Both of these conditions are implicit in the USB concept. Early testing verified these observations, as illustrated in figure 5 from Reference 12, which shows a ten PNdB reduction in sideline noise for a USB nozzle relative to an externally blown flap arrangement.

In summary, the advantages of the USB system, as listed in Figure 6, appeared sufficiently promising to warrant a technological attack on the disadvantages, also shown in Figure 6. The cruise problem is especially difficult, since the nacelle and wing upper surface flow fields are juxtaposed at freestream Mach numbers of 0.7 or greater. The engine exhaust introduces still another unfavorable flow-field element. Engine performance is, at best, questionably impacted by inlet immersion in the wing upwash and

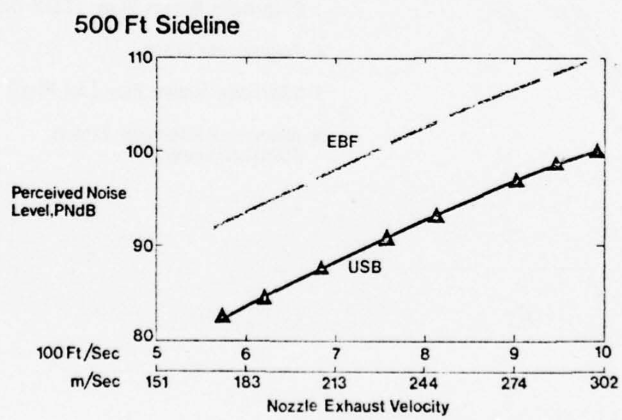

Figure 5. EBF and USB Noise Comparison

USB ADVANTAGES

Competitive High-Lift Performance

Adaptive to Hybrid Systems

Mechanical/Structural Simplicity

Competitive Acoustical Performance

USB DISADVANTAGES

Inherently Poor Cruise Configuration

Sensitivity of Wing Flow Field at Cruise

Lack of Comprehensive Aero/Acoustics Data Base

Conflicting High/Low-Speed Geometries

Conflicting High-Lift/Acoustical Performance

Figure 6. USB Advantages and USB Disadvantages

nozzle operation in the airframe flow field. Figure 7 illustrates the chaotic flow which can exist without proper recognition of these factors. This photograph was taken during Lockheed-Georgia Company wind-tunnel tests of an S-3A model incorporating a USB flow-through nacelle. The tunnel Mach number is 0.7 and visual flow is accomplished through the use of an emulsified solution of titanium dioxide, olaic acid, and oil. There are substantial areas of flow separation indicated, along with the development of a strong vortex system. These flow-field properties assure a large lift loss and drag increase which the force tests verified. Subsequent filleting and relocation of the nozzle vastly improved, but did not eliminate, these penalties. This illustration is not presented to imply that these problems are unsolvable. Rather, it is to indicate the difficulty of developing a satisfactory USB cruise configuration and the lack of a USB data base.

The Air Force, NASA, and the aerospace industry acknowledged the potential advantages of an upper-surface-blowing STOL configuration through a series of development programs designed to broaden the data base and resolve known problems. A good example is the excellent NASA-Lewis work in noise, propulsion effects, and interference as a part of the QCSEE engine program. Reference 13 gives

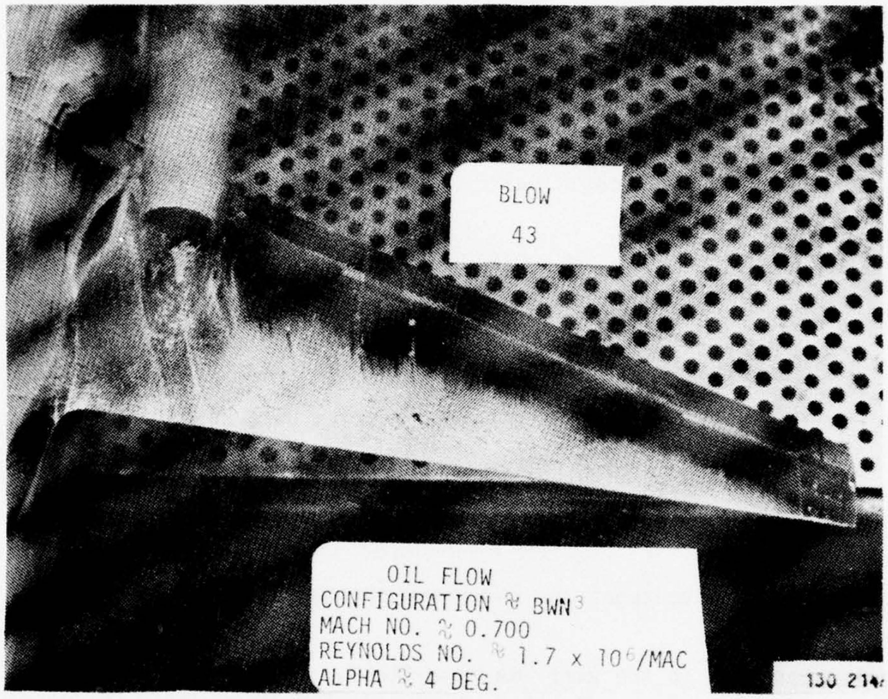

Figure 7. Potential USB Cruise Problems - S3A Oil Flow

an overview of this program. The Boeing YC-14 AMST transport represents a hardware manifestation of this concept. One of the key elements of the NASA-Langley USB research program was the development of a generalized and systematic USB data based in high-speed aerodynamics and acoustics. The program was implemented through two competitive contracts awarded to the Lockheed-Georgia Company. These research contracts were structured to develop effects and an understanding of aerodynamic and acoustic characteristics as influenced by design and operating conditions. This is accomplished through variations of appropriate parameters such as nozzle shape, nacelle location, engine pressure ratio, and others. A further objective was to isolate USB conceptual arrangements which enhance high-speed cruise performance and airport noise levels.

The remaining sections of this paper summarize completed portions of these contracted research programs and related activity.

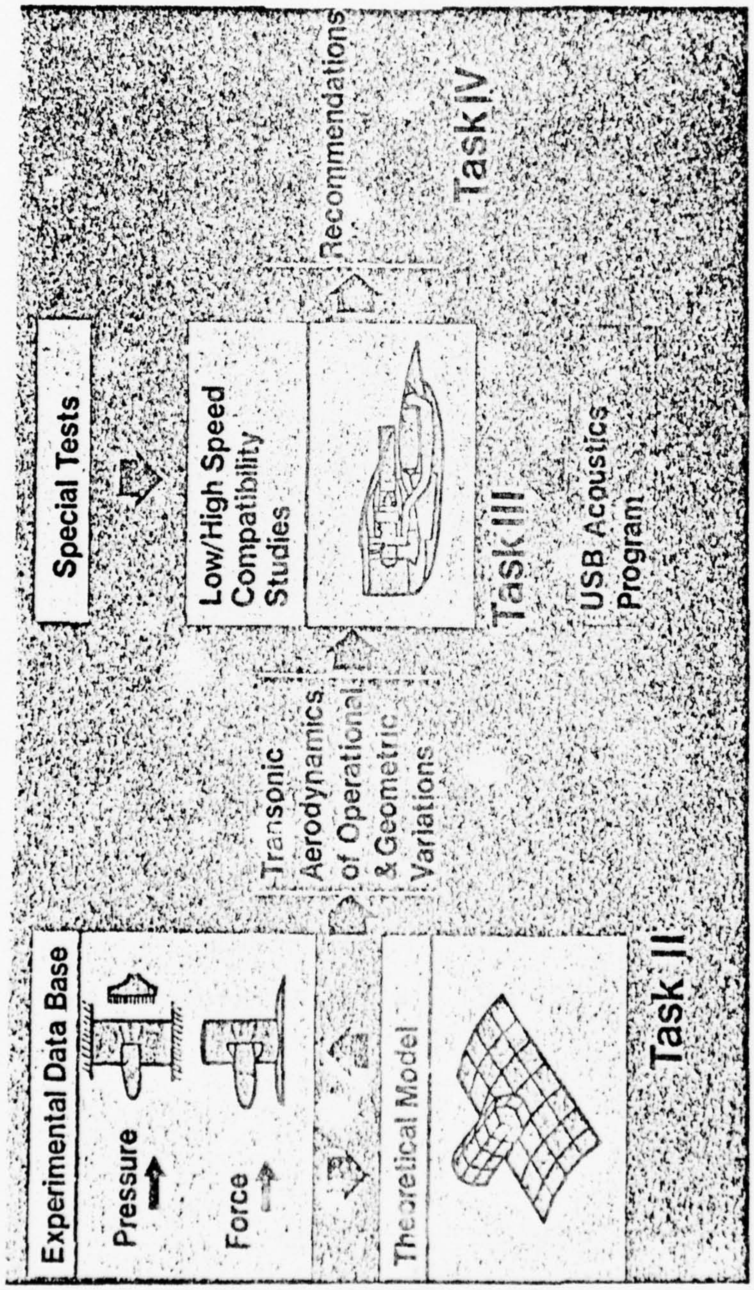

Figure 8. USB Cruise Program Objectives

USB CRUISE PROGRAM

The overall objective of the USB cruise program is to acquire a high-speed technology data base which identifies and evaluates those design variables affecting drag and engine performance. The selection of attractive USB nacelle and nozzle arrangements is an implicit element of the evaluation process. Secondary objectives include limited studies of the structural feasibility and mechanical practicality for selected candidate configurations.

Figure 8 illustrates these objectives in terms of the major tasks of the program. Task II is organized to establish effects and trends produced by parametric variations of operational and geometric characteristics of USB installations. This is accomplished experimentally with a high-speed wind tunnel program, augmented by a theoretical analysis. The results of this task indicate which USB configurations produce relatively good, high-speed performance. In Task III, these are subjected to "real world" analyses to assure satisfactory low-speed aerodynamic characteristics and design compatibility. The parallel acoustics program merges with the cruise activities at this point to complete the USB evaluation in terms of high-speed performance, airport performance, design realism, and airport noise. An additional element of Task III is the special tests which provide a limited evaluation of low-speed aerodynamics and high-speed wing scrubbing drag aft of the nozzle. The final part of the program, Task IV, is an identification of additional areas for research and development.

The objectives of this program are structured in accordance with the existing definition of suspected problem areas. The following list summarizes some of the variables impacting USB high-speed cruise.

1. Relative nozzle size
2. Nozzle aspect ratio
3. Nozzle discharge position
4. Afterbody boattail angle
5. Nacelle shape
6. Nozzle pressure ratio
7. Number of nacelles
8. Pylon length
9. Jet deflector angle
10. Wing sweep
11. Wing camber

EXPERIMENTAL INVESTIGATIONS

Test Facilities

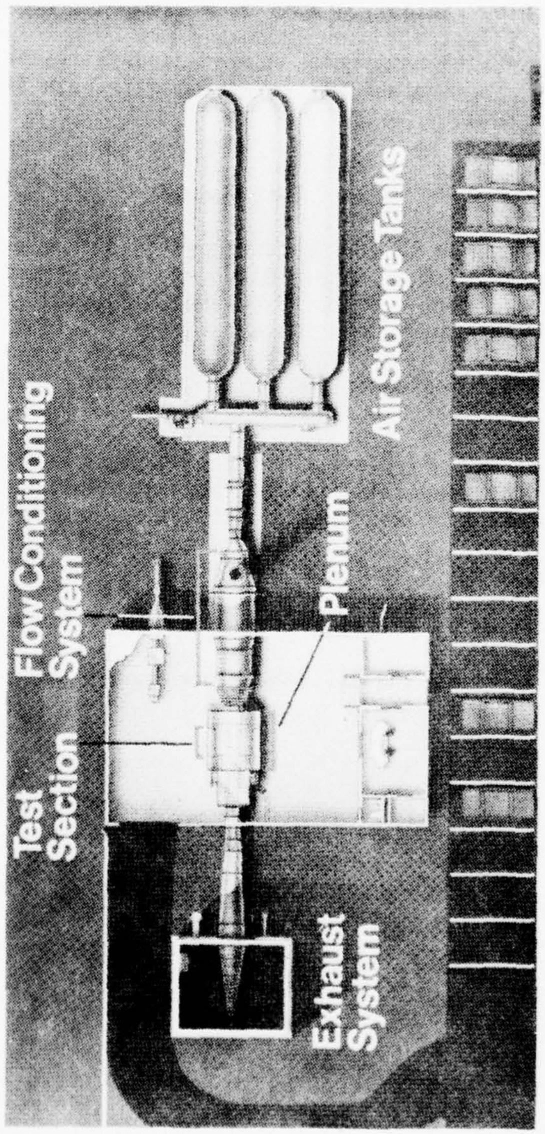

Figure 9. Lockheed Compressible Flow Facility (CFF)

The experimental investigations represent the major part of this program and are accomplished in the Lockheed-Georgia Company Compressible Flow Facility. This is a transonic blowdown wind tunnel and is illustrated in Figure 9. The 20 x 28-inch test facility features high Reynolds number capability, variable test wall porosity, and the ability to accommodate both two- and three-dimensional models.

Test Models

The wind tunnel models are designed to permit powered testing by ducting high-pressure nozzle discharge air in through a floor balance force system. Figure 10 schematically illustrates the "tinker-toy" concept which permits parametric variation of the effects noted thereon. In addition to the effects shown, it is possible to vary wing sweep and camber, forebody configuration, and nacelle number and position.

Nozzle aspect ratio is an especially important parameter, since low-speed studies indicate the desirability of spreading the engine efflux through a high aspect ratio nozzle. The model has the capability of varying this quantity from 1.25 (circular) to 6.0. Nozzle boattail angle is important in a USB application due to the interfering flow fields. The tested range varies from 6° to 36°, and exit position is varied from 10% to 50% of the wing chord.

Figure 11 shows the building block concept as appropriate to a

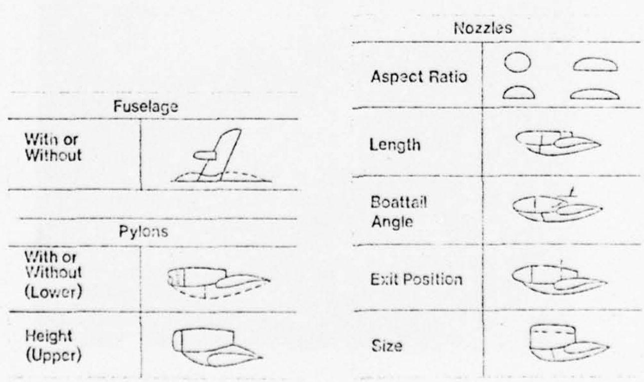

Figure 10. USB Cruise Program - Aerodynamic Effects Studies

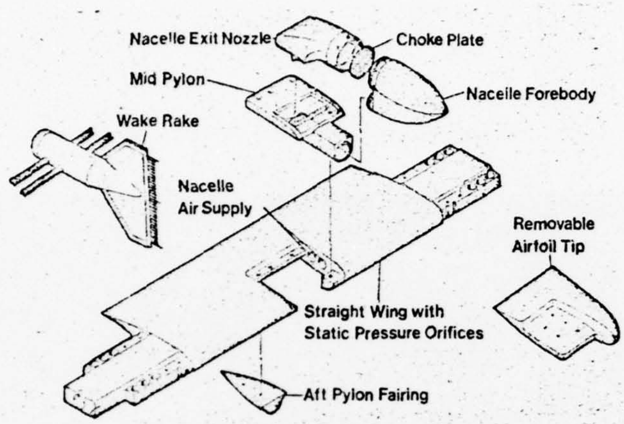

Figure 11. Typical Test Model

two-dimensional straight wing configuration. The nozzle discharge air is carried through the tunnel wall by way of a wing duct, which exhausts into the hollow, faired-over forebody. It then expands through a choke plate, which acts as a screen, and into the exit nozzle. These components can be tested in a three-dimensional mode by adding an airfoil tip and rotating the wing so that it is mounted on the wind tunnel floor.

Figure 12 shows the two-dimensional wing/nacelle pressure model installed in the tunnel with a traversing wake rake in place behind the nozzle exit. The pressure data identify the presence of shocks, separated flow, wing-lift distribution, and other phenomena. For this mode of testing, the floor and ceiling of the test section are porous, while the side walls in the vicinity of the model are solid.

Figure 13 shows a swept wing, three-dimensional, semi-span model with a simulated fuselage mounted on the tunnel floor. This installation is used to measure force data in the powered mode for a four-engined USB aircraft. It is appropriate to call attention to the large fillets between the nacelles and wing, and the wing and fuselage. These are empirically developed for minimizing nacelle interference utilizing visual flow techniques.

It should be emphasized here that the basic intent of the USB cruise experimental program was to parametrically develop data

WITH TRAVERSING WAKE RAKE

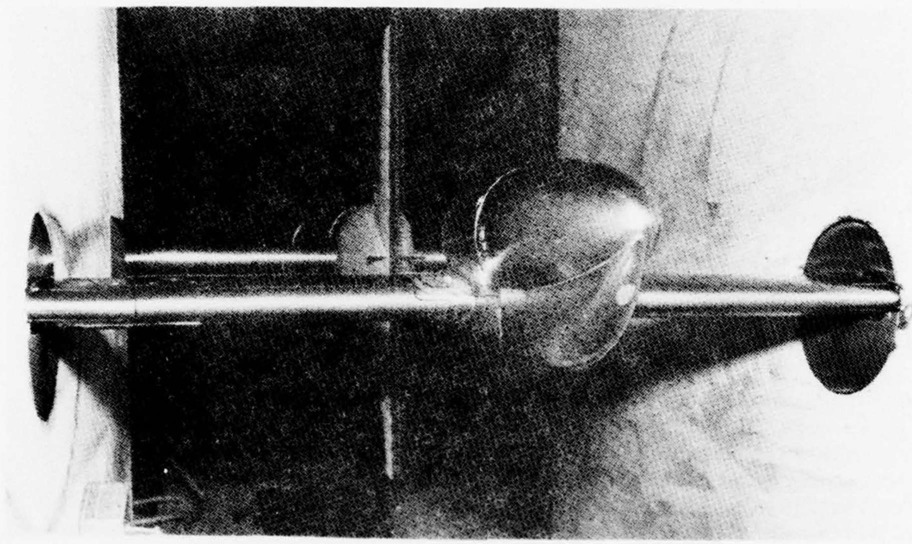

Figure 12. Wing-Nacelle Pressure Model

trends indicative of favorable cruise drag performance. The unrefined nature of the model components and the "tinker-toy" concept of configuration build-up would be expected to combine so as produce relatively high drag increments for the nacelle installations. The degree by which these drag increments could be reduced through design refinement was not necessarily a major consideration in these initial investigations.

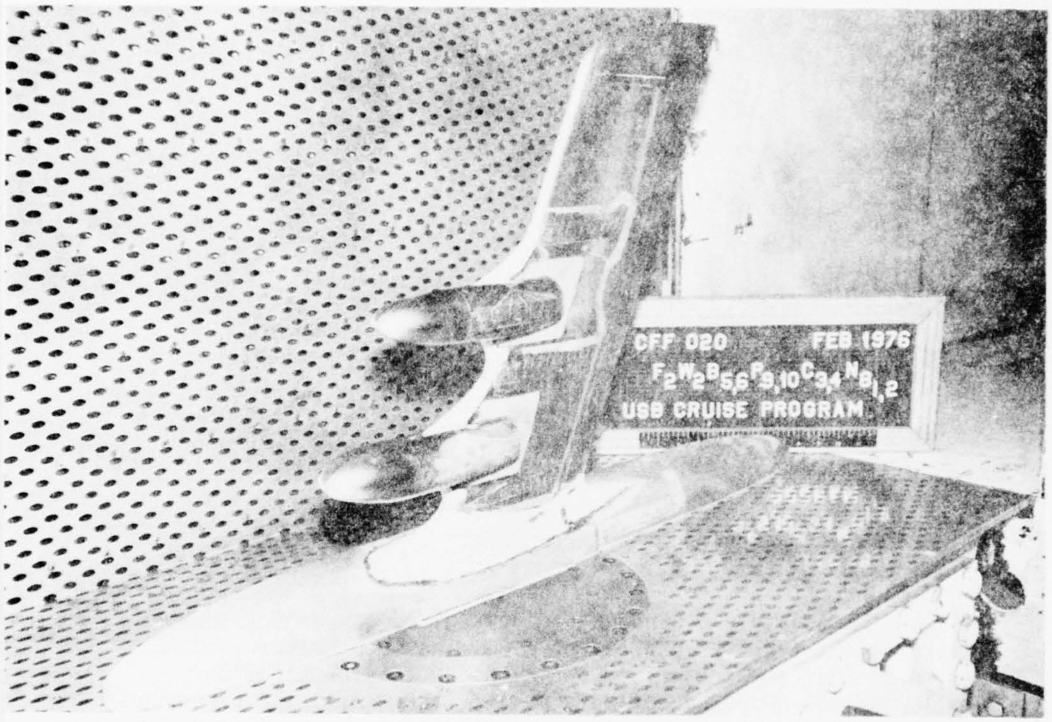

Figure 13. Swept-Wing Model in Test Section

Aero/Propulsive Interactions

Before proceeding to the details of results obtained in the experimental program, a qualitative discussion of the major aero-propulsive interactions occurring with USB installations can provide beneficial perspective to the data trends. Figure 14 shows a composite pressure distribution on an unswept wing in the vicinity

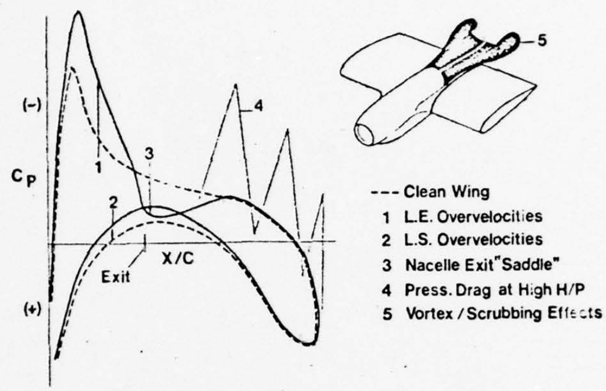

Figure 14. USB Cruise Design Problems

of, and immediately behind, a semi-circular, convergent nozzle operating at a pressure ratio of 2.6. The Mach number is 0.68, which is near the drag rise condition for this combination. Following the numbered callouts on the figure, (1) the presence of the nacelle generates "over-velocities" near the wing/nacelle intersection promoting strong leading edge shocks with increased chance for local, downstream separation. This effect can be minimized by the appropriate contouring and filleting of the wing/ nacelle junctures. Similarly, (2) the presence of underwing support structure creates interference flow fields on the wing undersurface. A supercritical airfoil section is particularly sensitive to such interference. Again, local contouring which recognizes the nacelle presence will moderate such effects. Near the nacelle exit, (3) the pressure distribution adjacent to the nacelle and jet takes on a "saddle shape" in responding to the choked exit condition of the nozzle. This "kink" in the distribution tends to force the leading edge shock to remain in a forward location with attendant high pressure losses and an increased potential for flow separation. The spanwise extent of this region can be suppressed to some degree by reducing the closure angle between the jet and wing upper surface. Along the jet centerline, (4) the pressure distributions reflect the compression and expansion shock patterns of the supersonic jet. When the jet follows the curvature of the airfoil closely, simple momentum considerations for the wing alone would suggest the onset of a pressure drag penalty which grows with increasing nozzle

pressure ratio. The magnitude of this effect will depend upon the design pressure ratio and the thickness of the wing, which tends to set aft-curvature of the wing upper surface. The vortex and scrubbing effects, (5) are associated with the entrainment and the roll-up of the three-dimensional jet along with surface friction losses.

Cruise Drag Accounting

The number and complexity of the phenomena just discussed impose a substantial burden on functional analyses of the expermental results and the isolation and possible treatment of specific problem areas. Both force and pressure data, static and wind-on tests, as well as analytical modeling have been used in combination to develop logical analysis techniques. The basic analytical approach has been to break down USB cruise drag penalty into its various components and then to analyze each of them individually. Brief discussions of the major drag/thrust-penalty components are contained in the following paragraphs.

Nacelle Friction Drag. The friction drag of the isolated nacelle is estimated by conventional techniques. No further assumptions are made here as to increased friction due to the high-velocity wing flow-field or to the use of a faired-over forebody in lieu of an operating inlet.

Nacelle Installation Penalty. The installation penalty is normally associated with jet momentum losses due to scrubbing of the wing surface by the jet. Such losses are quantified by a comparison of static test results from the isolated nacelle and similar tests made on the wing with nacelle installed. In terms of an effective cruise drag penalty and under the normally held assumption that such losses are essentially invariant with airspeed, a typical thrust degradation would be in the range of $\Delta C_D \approx .0010 - .0050$ at $C_\mu = 0.10$ (isolated nacelle thrust at $H_j/p_\infty \simeq 2.6$, $M_\infty = 0.68$).

Drag-Due-to-Lift. Drag penalties associated with lift generation fall into categories of either potential flow induced drag effects or those effects associated with changes in the viscous interactions. In the present context, these are lumped together to form a total drag-due-to-lift increment. It is identified by conventional means after reducing the measured lift and accelerating forces by the reactive thrust components.

Pressure Drag. A pressure drag component results from the high-velocity jet turning over the arc formed by the aft-wing surface. This effect is more clearly demonstrated in Figure 15, where the

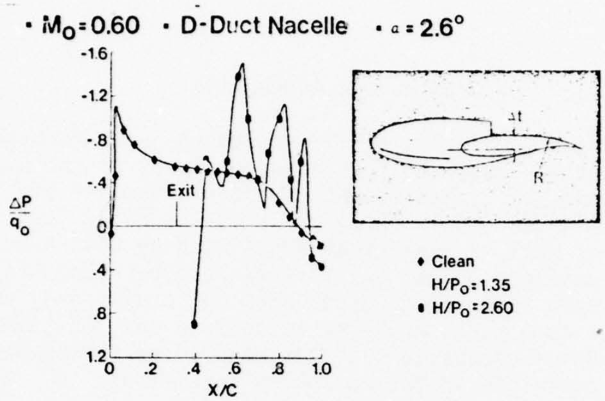

Figure 15. Aerodynamic Components - Pressure Drag

pressure distribution on the clean wing is compared with jet centerline pressures at several nozzle pressure ratios. Except at the nozzle exit, the centerline pressures approach the clean wing levels at a flow-through pressure ratio. As the nozzle pressure ratio advances to 2.60, the average (negative) pressure level increases and, in acting across the upper surface thickness, Δt, creates the pressure drag component.

As noted earlier, scrubbing losses evaluated from static tests are often assumed invariant with airspeed. Wake traverses behind a typical blowing nozzle, shown in Figure 16, indicate that substantially more jet vectoring can occur "wind-on" than is observed statically. Therefore, both pressure drag and scrubbing losses can vary with the test condition. The left side of Figure 16, derived from static tests, shows the jet cross-section as lines of constant, local total pressure-to-freestream total pressure ratio (H_ℓ/H). The right side of the figure provides the wind-on (M_0 = 0.6) cross-section at the same nozzle pressure ratio and angle of attack. Compared with the static case, the wind-on jet has a more concentrated form and closely follows the wing surface up to the trailing edge. In addition, a greater wake penetration by the high-speed core is indicated due to reduced mixing at the jet boundaries in the cruise condition.

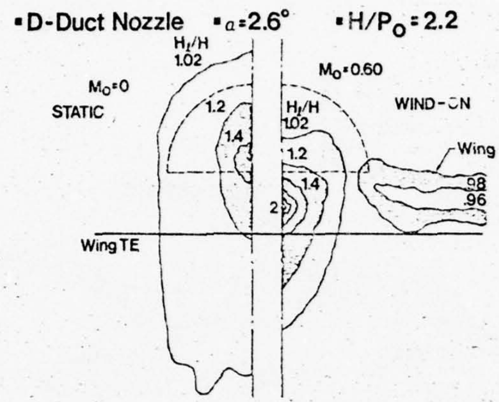

Figure 16. Jet Wake Isobar Comparison

Nozzle Geometric Effects

Boattail Angle/Aspect Ratio. Figures 17 and 18 portray typical variations in cruise drag penalty with the design geometry of the nozzle. The data are provided in ratio form, normalized to a selected value of the variable and as a function of nozzle pressure ratio. The effects of nozzle boattail (or roof) angle and nozzle exit aspect ratio are given in Figure 17. The limiting boattail angle (β), being somewhat pressure-ratio dependent, varies from about 20 to 25 degrees; at larger angles, separation effects become pronounced. At practical nozzle pressure ratios, the right-hand plot indicates that the D-duct (semicircular) exit shape appears to offer the best compromise between the high-pressure drag associated with the wider (high-aspect ratio) nozzles and a circular-shaped nozzle, which integrates poorly with the wing.

Size/Exit Position. Similar design variables of nozzle size and exit position are given in Figure 18. The parameter, $(\text{chord})^2$/nozzle area, defines the relative wing chord and nozzle size as tested; the drag coefficients are based on the frontal area of the nacelle in the form ΔC_{D_π}. These trends indicate that the drag per unit area of frontal area diminishes as the nacelle becomes larger. Thus, irrespective of mutual interference between nacelles, a two-engine (large-nacelle) configuration would be favored over a four-engine (small-nacelle) installation.

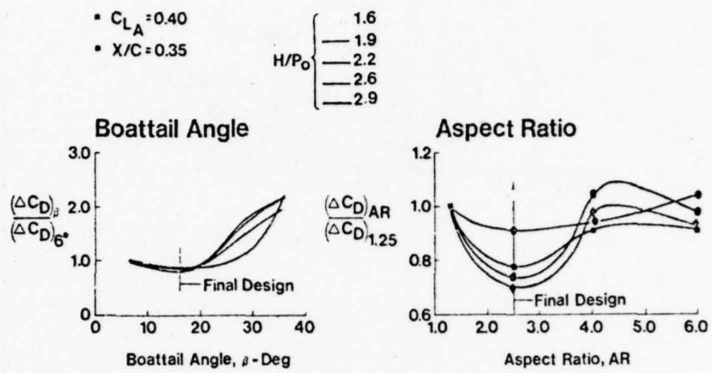

Figure 17. Nozzle Geometric Effects - Boattail Angle and Aspect Ratio

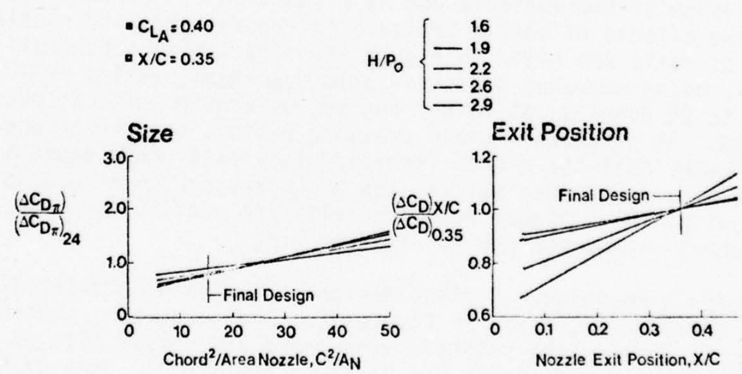

Figure 18. Nozzle Geometric Effects - Nozzle Size and Position

Also shown in Figure 18 is the effect of nozzle exit position. The trend of these data indicate, as is generally true with conventional, underslung installations, that the cruise-drag penalty diminishes as the nozzle exit is moved forward. Using data trends such as provided in Figures 17 and 18, the geometry of a candidate USB configuration, delineated on the figures, was selected for initial trade-off studies.

Theoretical Modeling

The analytical program performed a number of supportive functions relating to data analysis or data extrapolation while simultaneously providing essential clarification to otherwise obscure flow phenomena. The emphasis in this element of the program centered around development of effective mathematical modeling techniques reasonably representative of USB installations operating in the cruise regime.

Wing/Nacelle. A thick-wing program was selected for use as a base on which the nacelle and power package models could be superposed. The thick-wing model uses a variant on the vortex-lattice technique employing separate upper- and lower-surfaces with gaps at the leading and trailing edges. A single-surface, vortex-lattice representation was found to be adequate for describing the nacelle. Figure 19 illustrates the modeling technique as applied to a semicircular nozzle combined with an unswept wing; the surfaces of the power-effects model are also included.

Power Simulation. Power effects are simulated by combining vortex-cylinder and vortex-lattice methods. Figure 20 demonstrates how this combination can be effectively used to model the jet alone or the simulated powered nacelle. The "sink effect" produced by a suitable tapered vortex cylinder can also provide entrainment typical of flow into a real jet. As portrayed in Figure 19, the jet emerging at the exit is represented by an expanding, decreasing-strength ring vortex system. This system is tailored to conserve axial momentum while permitting entrainment of fluid from the local flow field at a rate consistent with standard results for axisymmetric jets. The figure shows the jet plume reshaped near the exit to reflect impingement effects caused by the relatively high boattail angle of the nozzle. The primary variable involved with this reshaping is the spreading angle in plan view as determined via flow visualization. In the case shown, a rectangular sheet at the wing trailing edge is assumed with the height of the jet determined by known area requirements. The remainder of the jet (i.e., between exit and trailing edge) is faired in accordance with mass-flow and momentum relationships. In the present work, no attempt is made to simulate free-jet turning, which has the effect

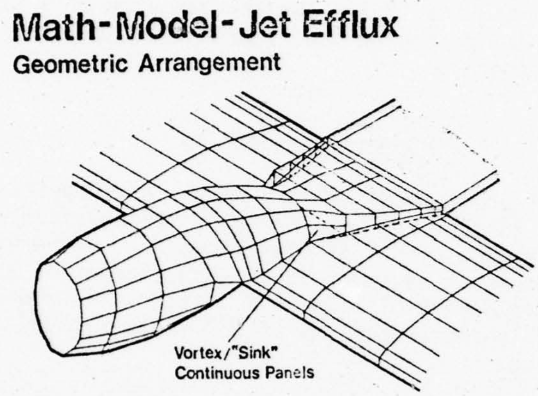

Figure 19. Theoretical Model of Wing-Nacelle-Jet

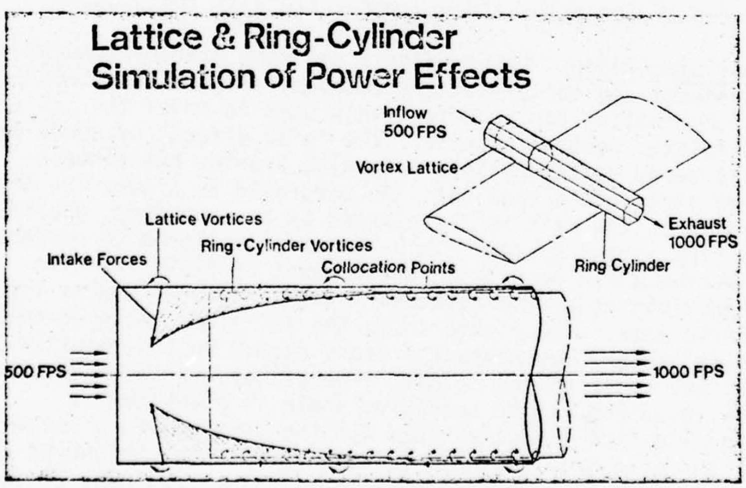

Figure 20. Power Package Simulation

of removing Coanda-turning forces from the basic flow model. These may be added retrospectively if desired. Use of these modeling techniques for the realistic simulation of areo/propulsive interactions has provided gratifying correlation with the experimental results.

Conclusions

1. Relatively large cruise-drag penalties were encountered as a result of jet scrubbing losses and a pressure drag produced by the vectoring jet.

2. USB nozzles designed for cruise with high boattail angles (i.e. $\beta > 25°$) or utilizing deflector plates can be severely penalized by scrubbing losses, boattail separation, and reduced lift augmentation.

3. A favorable cruise configuration would be represented by a contoured D-duct arrangement with the nozzle located as far forward as structural and weight considerations permit.

4. For a specified nozzle thrust, the more favorable nacelle configuration, from an interference drag standpoint, would be represented by a twin-engine configuration with larger nacelles rather than a four-engine version employing small nacelles.

5. Use of cruise flaps for additional jet vectoring provides no significant advantage to cruise performance of the tested configurations.

6. Pylon-mounted, upper-surface nacelles offer a potential cruise advantage at high nozzle pressure ratios by avoiding scrubbing and pressure drag penalties.

ACOUSTICS PROGRAM

The primary objective of the acoustics program is to provide a unified data base of USB noise trends and effects due to geometrical and operational parameter variations, which can be used in the design of practical USB aircraft configurations with low-noise characteristics. This program specifically excludes the usual internal engine noises, such as fan, turbine, and combustion noise, common to all turbine-powered aircraft. The emphasis is on the unique noise created by the jet exhaust flow and its interaction with wing and flap surfaces. Secondary objectives are (1) to assure that recommended low-noise design information is compatible with low- and high-speed aircraft operations and (2) to develop recommendations for further investigation of those technical areas where the state of the art is lacking.

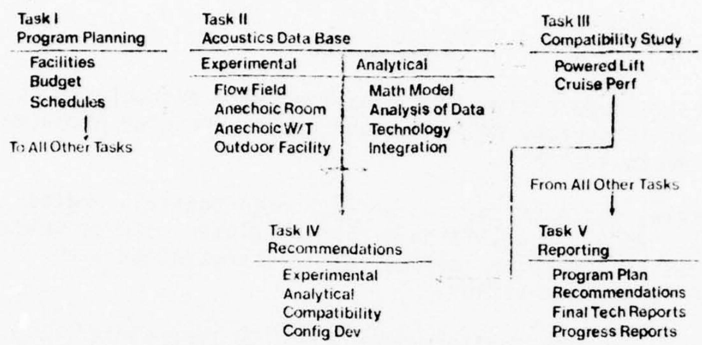

Figure 21. Acoustics Program Organization and Flow

The organization and flow of the acoustics program are shown in Figure 21. Task I was the initial planning and scheduling work that served all the subsequent technical and reporting tasks. Task II, which addressed the primary objective and represented about 70% of the entire contract effort, was itself divided into two major subdivisions - Experimental and Analytical. The larger experimental effort was further broken down into four distinct test programs, each in a different test facility. The concurrent analytical program utilized the results of all the test phases and provided continuity among all the elements of Task II. Task III consisted of studies to assure compatibility among low-noise characteristics and low-speed and high-speed aerodynamic performance, a joint effort with the USB aero contract. Task IV consisted of developing recommendations for needed further study for those technical areas which were beyond the scope of the current work effort. Task V includes all reporting activities.

As indicated in Figure 22, there are actually several sources of noise in a USB flow system. These include (1) the undisturbed jet flow, (2) wall-jet flow, (3) roll-up of the wall-jet flow, (4) the downstream wake of the entire jet flow, (5) impingement of jet turbulence on the wing for those configurations where the jet nozzle is evaluated above the wing, (6) trailing edge flow unsteadiness and shear gradient turbulence, and (7) aeroacoustic resonances that can exist in certain flow geometry situations. The first six source areas all produce broadband random-type noise, whereas the seventh source causes discrete frequency or tone-type noise. Source 6, the trailing edge flow, is believed to be the

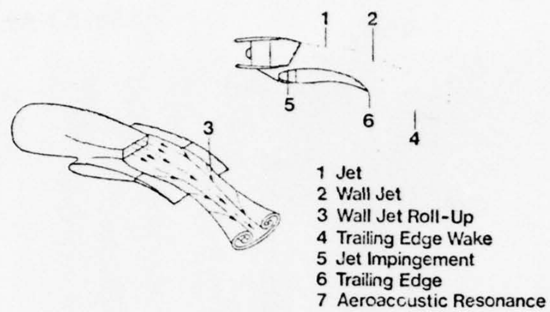

Figure 22. USB Noise Sources

predominant noise generating mechanism and, consequently, work in all phases of the overall program focused on that part of the USB flow field.

Experimental Investigations

The first test program was entirely a static investigation of USB flow fields to provide a better understanding of the basic steady and unsteady flow phenomena important to noise generation. The second test program was also static and was oriented entirely to acquiring parametric acoustic data. The third program was again static and utilized larger models for the acquisition of acoustics, flow-field, and propulsive-lift performance data. The fourth program was performed in an anechoic wind tunnel to obtain effects of simulated forward speed on USB noise generation and propagation patterns.

Flow-field Tests. This series of tests was performed on a static model with a two-dimensional wing of 6-inch (typical) chord and 20-inch length and is shown in Figure 23. Primary test parameters and parameter value ranges were:

Nozzle Pressure Ratio	1.1 to 1.5
Nozzle Shape	Round, Elliptical, "D" Shaped, Rectangular (Aspect Ratio 2, 4, 8)

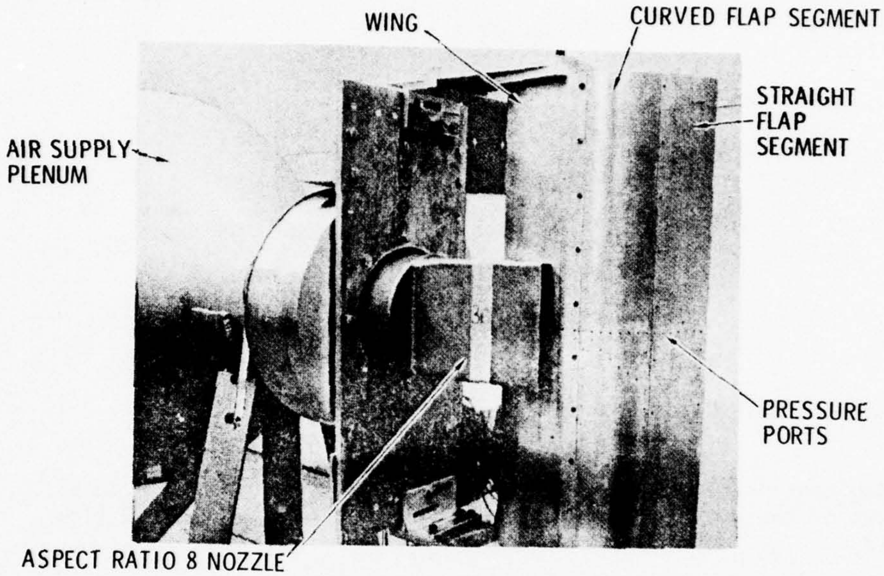

Figure 23. Flow Field Test Setup

Nozzle Impingement Angle	0° to 30°
Nozzle Chordwise Location	20% to 50%
Flap Deflection Angle	0° to 60°
Flap Radius of Curvature	∞ to 4"
Flap Length	1.5" to 4.64"

In addition, limited data were taken at several jet temperatures and nozzle vertical locations.

All possible combinations of these variables could not be evaluated due to time limitations. Therefore, the various parameters were usually evaluated relative to a baseline or representative configuration which utilized 2.55" flaps with a 3" radius of curvature and deflection angle of 60°, with an aspect ratio 4 rec-

tangular nozzle with the nozzle location at 20% chord and a nozzle impingement angle of 20°.

Several types of flow-field data were obtained, including flow visualization photographs. The first type of photographs were taken of wing-flap surface oil flows, as illustrated in Figure 24. This sequence shows how flow spreading and attachment vary as a function of nozzle impingement angle. The second type of photographs were obtained with a schlieren system. Typical examples of these photographs of the flow field are shown in Figure 25. The three cases shown correspond to the surface oil-flow photos just discussed. Flow separation prior to the trailing edge is rather obvious at 0°; at 10°, it is just barely attached at the edge, and is well attached at 20°.

The major parts of the flow-field data base were obtained with a hot-wire system. This included measurements of velocity profiles, turbulence profiles, and a limited amount of turbulence space-time correlations. Typical turbulence profiles are given in Figure 26. These profiles are at several spanwise locations in the area just above the flap trailing edge. For the profile exactly on the nozzle centerline, the relatively quiescent jet potential core extends aft of the trailing edge, resulting in the peculiar shape of that profile.

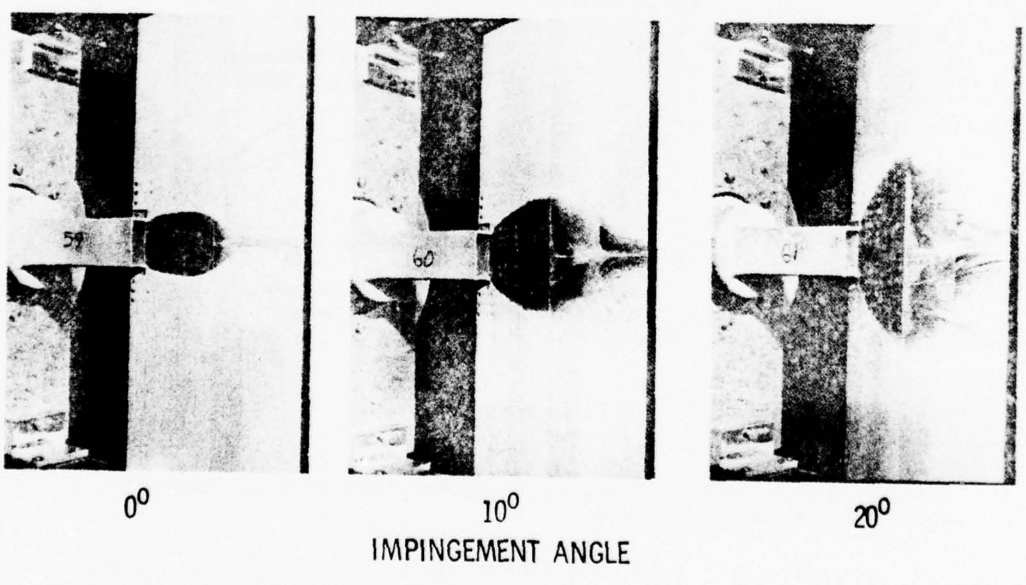

Figure 24. Oil Flow Visualization

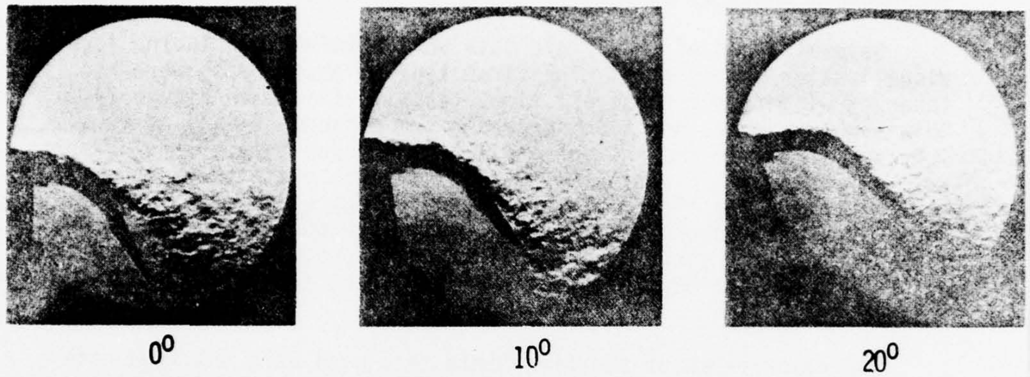

Figure 25. Schlieren Flow Visualization

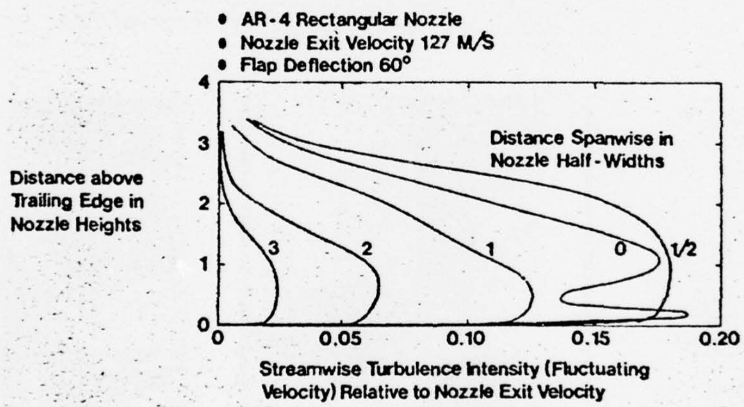

Figure 26. Trailing Edge Turbulence Profiles

Limited static surface pressure measurements were also taken to supplement the hot-wire data.

In addition to supplying flow-field data to assist in the basic understanding of USB noise generation, a major task was to define configurations with attached flow at the trailing edge for subsequent acoustic testing.

Parametric Acoustic Tests. This series of tests, which is the largest single element of the entire USB noise program, was performed in the anechoic room shown in Figure 27. The model is shown inverted and attached to a polyurethane-foam-covered and muffled air supply at the lower left center of the figure. Support arches with attached data acquisition microphones can also be seen, as well as the sound absorbing wedges lining the surfaces of the room. The model could be rotated in order to obtain noise data at other than the planes in space shown in the figure. However, the data illustrated in the following discussions were taken at one point only for simplification. That point is directly opposite the bottom of the wing, which simulates noise trends and effects for a ground observer directly under a USB aircraft. Trends at this location, in most cases, are similar to trends at other points below the wing as well. In addition, the acoustic results described are for attached flow conditions. The models, test parameters, and parameter ranges discussed in the previous section were the same for this test program.

Nozzle Exit Velocity. Nozzle exit velocity has a major effect on USB noise. Both noise level and peak frequency increase as jet velocity (V_j) increases, as indicated in Figure 28. The spectral data shown are for a series of jet velocities with all other parameters constant. The overall level of noise is typically proportional to $V_j^{5.5}$ directly under the model, proportional to $V_j^{5.0}$ in the forward quadrant, and varies to $V_j^{7.5}$ in the extreme aft quadrant.

Nozzle Impingement Angle. As the nozzle angle relative to the top of the wing is increased, the flow is spread out more over the wing and flaps. The noise spectrum, as can be seen in Figure 29, is affected mainly in the mid-frequency range, where the lower noise levels correspond to higher impingement angles.

Nozzle Shape. The only nozzle shape parameter that appears to be significant is aspect ratio. Higher aspect ratios mean more flow spreading causing mid-frequency noise reductions similar to the nozzle impingement angle results.

Flow Path Length. The subject of flow path length is concerned with two geometric parameters: (1) nozzle horizontal location on the wing and (2) flap trailing edge length. Either parameter changes the total flow path length between the nozzle exit plane and the flap trailing edge. As flow length increases, higher frequency noise decreases regardless of which of the two parameters' length was varied, as illustrated in Figure 30. Next to jet velocity, flow length is the second most important basic parameter which affects USB noise.

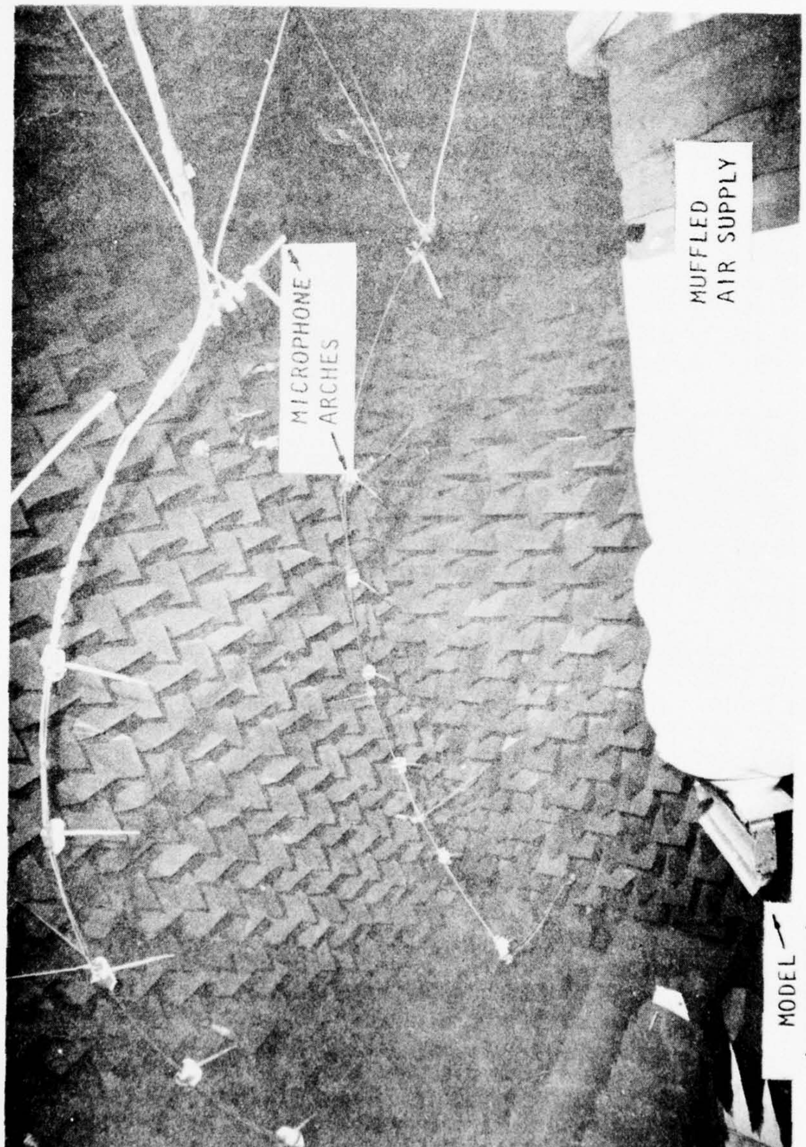

Figure 27. Anechoic Room Test Setup

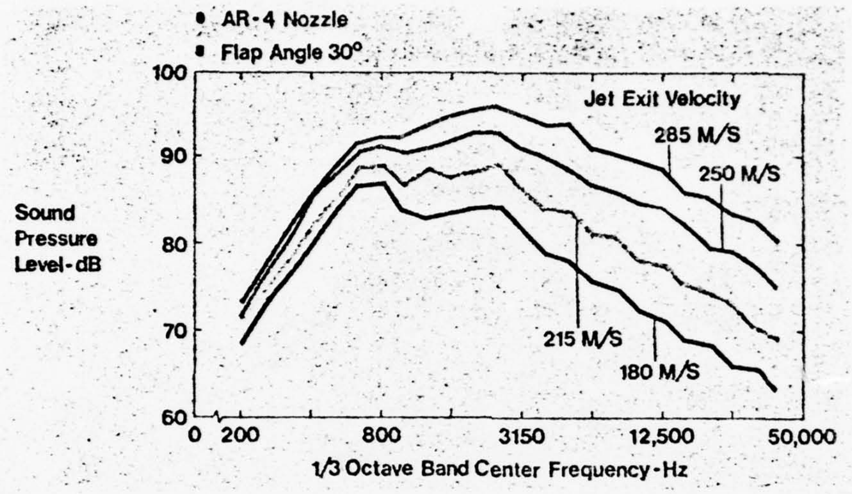

Figure 28. Effect of Jet Exit Velocity

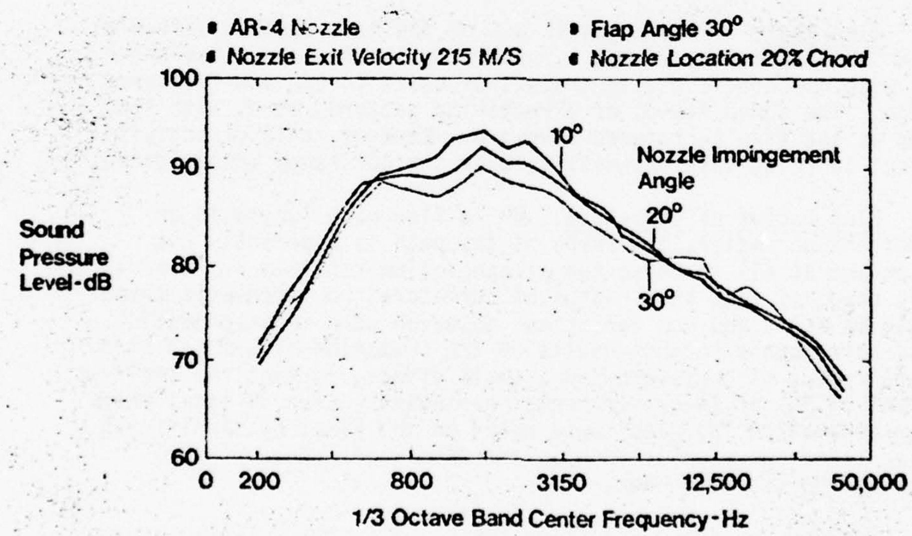

Figure 29. Effect of Nozzle Impingement Angle

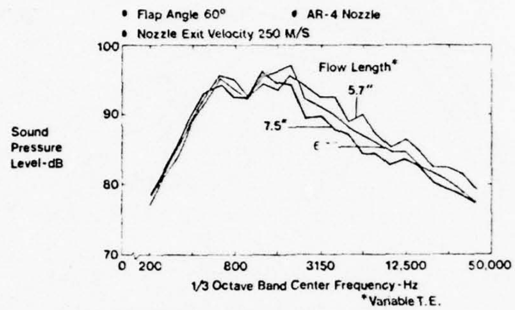

Figure 30. Effect of Surface Flow Length

Flap Angle. Flap angle is one of the more obvious variables in a USB system, but it has a rather small effect on noise under the wing. There is a slight spectrum shift to the low-frequency range. The sound field, or directivity pattern, moves with the flap as the flap is rotated downward. However, this directivity effect is relatively insensitive over the 60° range investigated.

Flap Radius of Curvature. While flow path length is an important parameter, the shape of the path is apparently not important at all to noise for attached flow conditions. Over a wide range of flap knee radius of curvature, no systematic trend could be found, and the variations observed were inconsequential. This corresponds to the results of the companion flow-field study where radius of curvature had a small effect, in fact the smallest effect of any of the experimental variables. Even in cases where flow separation "bubbles" were noted on the flap, no significant noise trend was seen as long as the flow reattached prior to leaving the trailing edge.

Scaling and Noise Reduction Tests. This third test program was conducted on the static outdoor model test stand shown in Figure 31. The model wing and flap system was two and a half times larger than the previously discussed model. Flow-field measurements, in the form of hot-wire velocity and turbulence profiles and oil flow patterns, were made which compared well with the smaller scale model data. Acoustic data also scaled well where noise intensity was directly proportional to nozzle flow area and frequency was inversely proportional to the linear dimensions. Both these trends

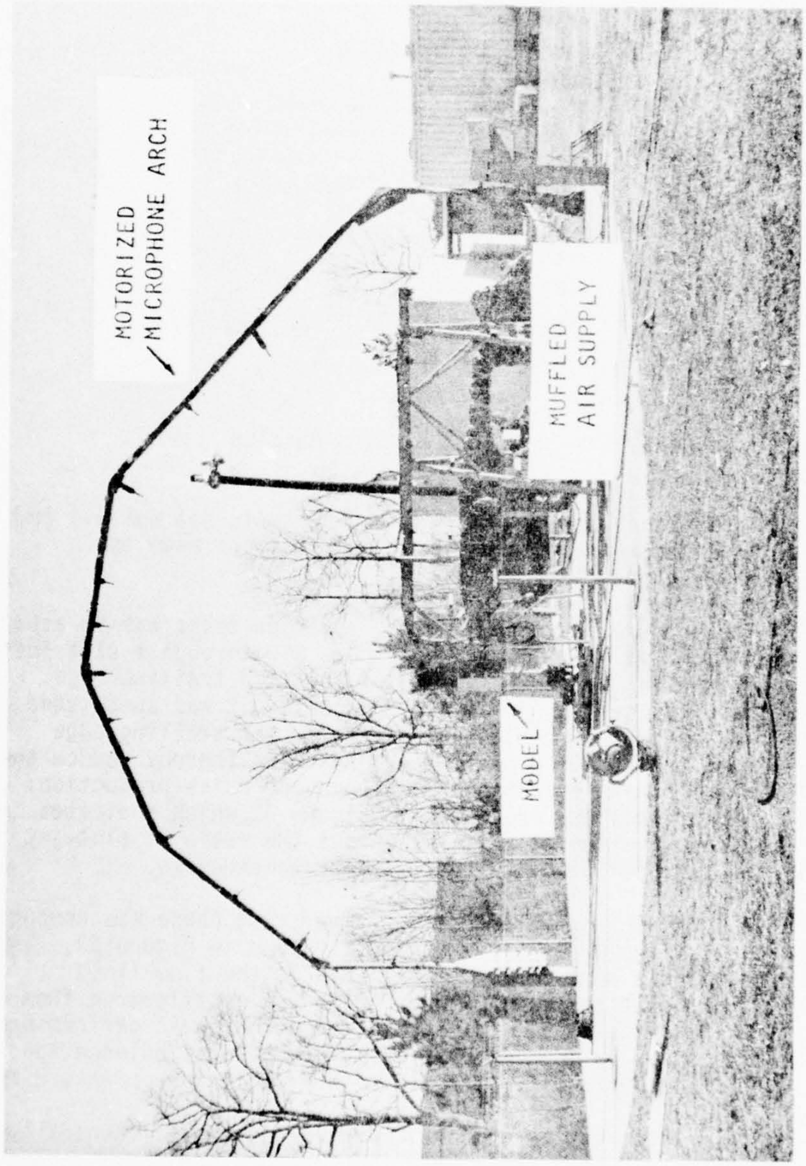

Figure 31. Outdoor Test Facility

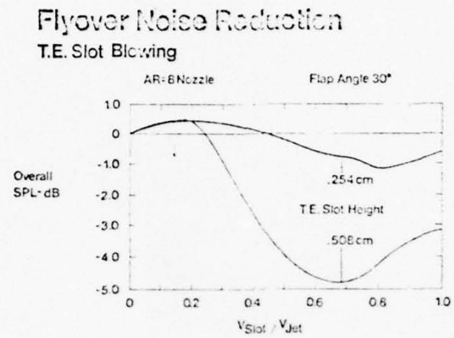

Figure 32. Typical Trailing Edge Blowing Results

are essentially the same as for ordinary subsonic jet noise. In addition, lift, drag, and thrust measurements were made to correlate with noise characteristics.

One of the more successful noise reduction tests was an active air blowing concept. The idea was to blow air through a slot just upstream of and on the lower surface of the flap trailing edge. Based on previous testing of similar concepts, it was speculated that a thin layer of air injected just below the trailing edge would thicken the trailing edge shear layer and thereby reduce the velocity gradient and resultant turbulence and noise production. Typical results obtained are shown in Figure 32, which indicates overall noise reduction in decibels versus the ratio of slot-jet velocity to main-jet velocity for two slot thicknesses.

Anechoic Wind Tunnel Tests. The final test phase was conducted in the free-jet type anechoic wind tunnel shown in Figure 33. The model used was the same scale as that used in the flow-field parametric acoustic tests. The model wing was cantilevered from a fairing for noise testing. Later, limited aerodynamic performance data were obtained by mounting the wing from a force balance, and velocity profiles were also obtained with a multi-tube pressure rake.

Typical noise results are indicated in Figure 34. Generally, there is a low-frequency noise reduction, whereas the high-frequency range is essentially unaffected.

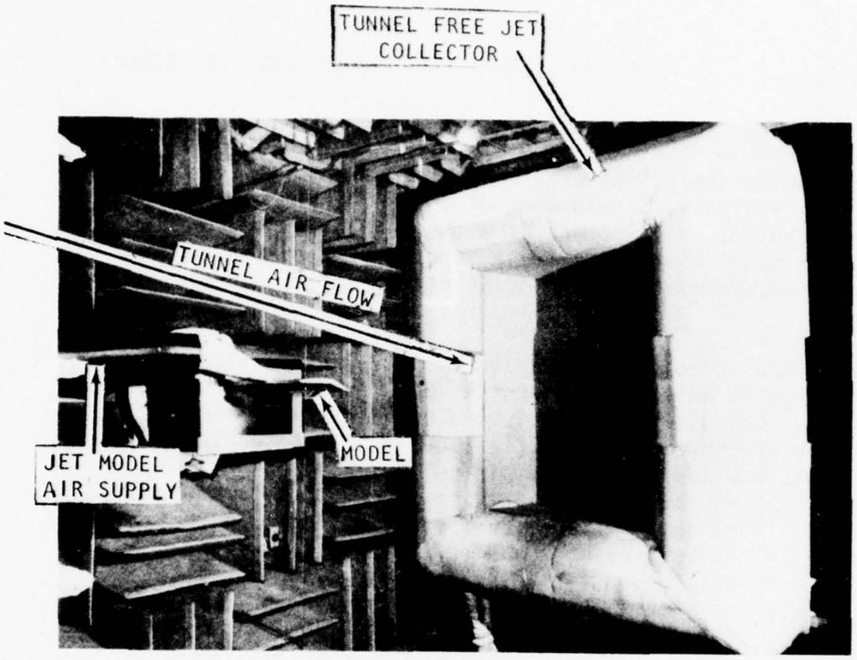

Figure 33. Free Jet Anechoic Wind Tunnel

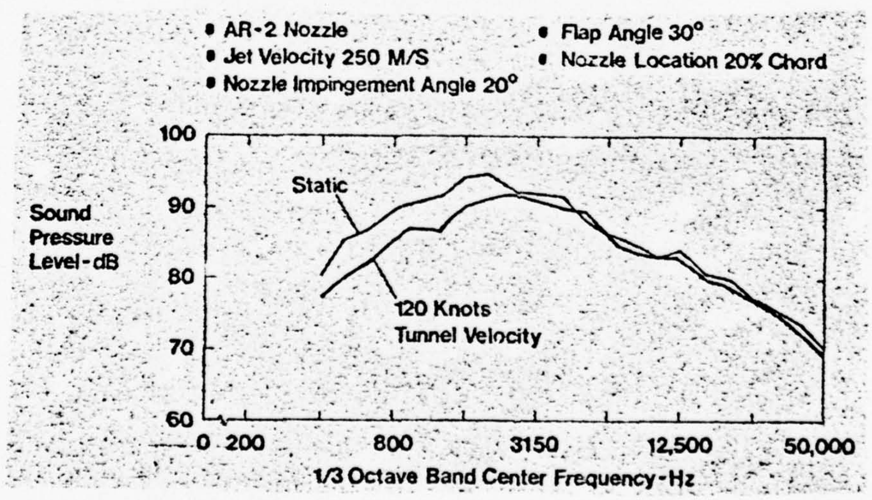

Figure 34. Typical Wind Tunnel Results

Analytical Investigations

The companion analytical program performed several functions including data analysis, development of an empirical noise prediction technique, development of a theoretical approach to mathematically model blown-flap noise, and the use of the empirical noise prediction procedure in the noise-aerodynamics compatibility study.

The empirical prediction technique is based primarily on the parametric acoustic data. Consequently, input data for a specific prediction job consists of geometric and operational parameters, such as nozzle pressure ratio, nozzle location, flap length, etc. This method was derived from the empirical collapse of data of the type shown in Figure 35. This figure shows how spectrum shape and sound pressure level data collapse at a specific angle (115° from leading edge) in the vertical fore and aft plane. The empirical technique is especially good for aircraft design studies, since it is essentially based on design-type parameters. It can predict noise at a given point or at a matrix of points from which ground noise contours or noise "footprints" can be machine plotted.

The theoretical approach is based entirely on relating radiated noise to steady and unsteady trailing edge flow parameters. The parameters needed are flow Mach number, flow thicknesses, flow velocity gradient, turbulence convection velocity, scale of

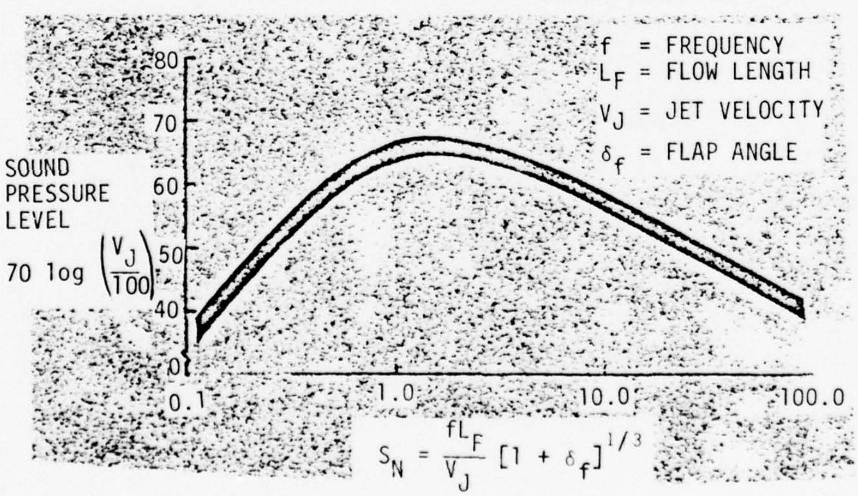

Figure 35. Normalized Spectra

anisotrophy, decay rate, and turbulence spectrum shape. Appropriate data of this type were provided from the flow-field tests, and the resulting noise calculations compared well with the experimental noise data. The theoretical model is most useful for the investigation of basic generation processes and noise reduction concepts.

Low Noise Aircraft Design Conclusions

The primary parameters which control USB noise and the appropriate design variables are:

Trailing edge flow velocity (nozzle pressure ratio)

Flow path length (nozzle chordwise location and/or flap length)

Flow spreading (nozzle aspect ratio, angle with wing, vertical location)

Trailing edge flow velocity at the flow-field centerline is the single, most important low-noise parameter. Flow path length and flow spreading have small effects on trailing edge velocity, but exhaust nozzle pressure ratio (determined by basic engine cycle) has the predominant effect (see Figure 28). Therefore, in the design of a blown-flap aircraft, consideration should be given to utilizing as low a nozzle pressure ratio (as high a bypass ratio) as possible.

Flow path length can be extended by moving the engines forward on the wing or by utilizing a flap system of longer chord. Lengthening the flow path over the wing and flap structures primarily reduces high frequency noise (see Figure 30), which is highly beneficial from the community noise standpoint, since high frequency noise is more annoying than low frequency noise.

Flow spreading is a complex phenomenon primarily controlled by nozzle aspect ratio, nozzle angle relative to the wing, and vertical location relative to the wing. These three design variables are not necessarily independent variables. For instance, if the nozzle is located above the wing surface, it must also be angled down toward the wing or flaps in order to achieve flow attachment and the desired lift augmentation. Flow impingement angle, controlled by basic nozzle angle relative to the wing (or nozzle roof angle or an external deflector) is important to the peak noise area, as indicated in Figure 29. Nozzle aspect ratio is the overriding variable concerning nozzle shape. As an example, elliptical "D" shaped, or rectangular nozzles, all with the same aspect ratio, have essentially the same noise characteristics. However, high aspect ratio nozzles

are lower in peak noise level than low aspect ratio nozzles, since high aspect ratio nozzles spread the flow much like an increase in nozzle impingement angle. Vertical location relative to the wing is also important because, as the nozzle goes higher, flow mixing turbulence increases prior to jet impingement on the wing and the shielding effect is less. Therefore, even if elevating the nozzle location and increasing the jet impingement angle causes increased flow spreading, the noise below the wing generally goes up.

As in all aircraft design programs, trades must be made between the various desired features in the overall design. Unfortunately, low noise design features generally cost aircraft performance in some way. Consequently, there is no truly optimum low noise design when it comes to an integrated powered-lift system. A best compromise design can be evolved, but only after a complete aircraft design study including the evaluation of factors such as noise, weight, system complexity, cost, aerodynamic performance, mission requirements, etc.

RECOMMENDATIONS

As the overall USB aerodynamics and acoustics programs progressed, deficiencies in the state of the art of several technical areas became obvious. As a result, a number of recommendations were evolved regarding future research needs. The types of recommendations addressed relate to:

(1) An improved understanding of characteristic phenomena associated with noise generation and aero/propulsive interactions

(2) The development of improved aerodynamic concepts and noise reduction methods

(3) Better understanding of speed effects on noise and the compromises necessary to effective utilization of USB at both low and high speeds.

(4) A more complete definition of USB as a total system component with special emphasis on sonic fatigue and soundproofing, along with structural, mechanical, and subsystem interfacing

(5) Improved methods for integrating low-noise, high-performance characteristics into effective propulsive-lift configurations

Specific research needs in the aerodynamics and acoustics areas are discussed in the paragraphs which follow.

Recommendations - Aerodynamics

Pylon-Mounted (Over-the-Wing) Nacelles, High Speed. Additional experimental/analytical studies are in order to provide guidelines for the optimum integration of the powered OTW pylon/nacelle with the wing at cruise conditions. Limits of potential flow theory in providing realistic trends in pylon shape, nacelle position, and contour in the presence of the jet need additional study and experimental verification.

Pylon-Mounted (OTW), Low Speed. In concert with item (1) above, low-speed investigations are needed to explore the effectiveness of various mechanical means of deflecting the jet down onto the wing surface for powered-lift benefits. A more extensive survey of known devices such as "eyebrow" deflector plates with the simultaneous recognition of high-speed design compromises, are mandatory to further refinements of this concept.

Integration Configuration Design Studies. The USB Cruise Program was designed to provide an exploratory data base from which more refined cruise designs could evolve. A recommended program would utilize the experimental findings and the analytical tools developed under the present study to provide second-level refinements to the fully integrated wing/nacelle/jet combination with minimum cruise drag as a basic objective.

USB OTW Aircraft Design System Study. An element of the USB Cruise Program involved a preliminary system study of the USB concept from the standpoints of aerodynamics and acoustics operating under both cruise and airport environments. A more comprehensive study of this type should be made with the subject concepts broadened to include OTW designs.

Analytical Model Correlation. The basic elements of the computational technique were developed under the USB Cruise Program. More extensive correlation of the math model with experimental data would delineate areas of needed refinement.

Jet Efflux Analytical Modeling. The theoretical modeling of a jet fully responsive to varying pressure fields and boundary conditions was beyond the scope of the data base program. To realize the full potential of the analytical techniques developed representing typical reactive systems, present constraints on jet behavior downstream of the nozzle need to be relaxed.

Jet Plume Boundary Optimization. As a consequence of the isolation of the pressure drag penalty as a major contributor to cruise drag, investigations for suppressing this effect are needed. Figure 36 postulates several approaches to airfoil modification

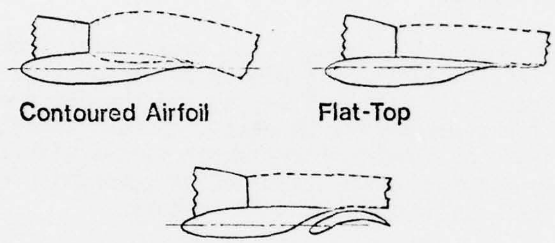

Figure 36. Jet Drag Reduction - Airfoil Modifications

behind the nacelle with the objective of minimizing pressure drag. Since pressure drag and scrubbing drag penalties are both associated with the attached jet, the recommended investigations should include more intensive studies of the latter penalties.

Recommendations - Acoustics

Basic Understanding of USB Noise. There are four areas under this general heading that need additional work. First, over-the-wing or OTW (i.e., nacelle and jet exit above the wing, but jet vectored down onto the wing for high-lift performance) noise and flow data are needed to bring that configuration's data base up to the level of that of the blended or faired nacelle. The second item is the need of more sophisticated flow data (more space-time turbulence correlations, for example) for both the blended nacelle and the OTW configurations. Thirdly, there is a need for additional acoustics tests of the effects of upstream temperature and turbulence. Last, there is a need to further develop the basic acoustic theory to assist in basic understanding, prediction accuracy, and in noise reduction efforts.

Noise Reduction Techniques. Once a better understanding of USB acoustics is in hand, efforts should be made to pursue noise reduction concepts including (1) geometry and operating parameter

optimization, (2) passive devices (e.g., irregular trailing edges), and (3) active devices (e.g., trailing edge blowing).

Flight Effects. The flight effects observed were generally typical of other similar tests for locations under the wing. However, the magnitude of the effect at different angles (fore and aft, as well as laterally) relative to the model is quite variable and the reasons for this variability are unknown. Therefore, more diagnostic testing of the true nature of flight effects on noise is needed.

Fuselage and Wing Environments. Near-field noise and fluctuating surface pressures are indicated to be rather severe for USB installations. The experimental data to date are primarily of the spectral-distribution-at-a-point variety. Additional data of the time- and spatial-correlation type are needed for better definition of structural vibration and sonic fatigue problems and in fuselage soundproofing analyses and design.

Low-Noise Configuration Design Study. The results of all the previous technology areas (basic noise source understanding and characteristics, flight effects, noise reduction concepts, near-field effects, etc.) are ultimately used in the search for optimum low-noise design in a USB system. An optimum system design or best compromise must also consider low-speed aerodynamic performance, as well as possible cruise performance effects. Such a system optimization can be most effectively done as an integral part of a complete aircraft system study, and a study of that type is recommended.

Concluding Remarks. It is hoped that NASA will be able to make comparisons of the extensive data base from this and several other recent USB noise programs and the data that will emerge from the NASA/AMST cooperative program and the QSRA program. To date, all published research work has either been done analytically or done with models on small- and large-scale test rigs. The missing link between the existing bank of data and real aircraft data (statically and in flight) should be investigated at the earliest possible date in order to confirm model and analytical techniques to advance the state of the art.

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DISCUSSION

CHASZEYKA: (Office of Naval Research)

You made several recommendations there with respect to research and development on acoustics. I am wondering, at what level are you casting these recommendations? Is it more basic information or a better understanding of what is already known and published?

RYLE:

I think it has to be both because I do not think you develop a better understanding of what exists until you have the basic research which provides that. And I do not think you pick up the one or two PnDb noise reduction until you understand what it is exactly you are working on. Does that answer your question?

MAUS: (The University of Tennessee Space Institute)

Can you tell me to what extent the wing acted as an effective noise shield?

RYLE:

There is pretty conclusive evidence to indicate several PnDb reduction in noise due to wing shielding. Very early NASA-Ames results (which were presented at NASA-Ames a number of years ago) showed a significant reduction in noise, all other factors being equal, between a USB arrangement and an EBF arrangement.

MAUS:

I guess I was thinking of a comparison of sound pressure data, above the wing and below the wing in your experiments.

RYLE:

No, I am sorry I cannot answer that question.

BRADLEY: (General Dynamics)

You showed force tests with parametric movement of the nacelles. Were all of these data taken with faired-over inlets?

RYLE:

Most of the data were taken with faired-over inlets. This is the way these drag data were measured. We took the isolated nacelle, put it on a force balance, measured the data, and correlated it with nozzle pressure ratio which was recorded at the nozzle exit. Then

we put the nacelle in combination with the model itself and repeated it again in the static case in order to pick up the scrubbing drag, and, I guess, you could call it the propulsion effect for the static case alone. Then, of course, we tested it with the wind on, and the difference between the two is what you are actually seeing here. We did measure some flow-through cases, some hard-bodied representation of the flow-through nacelles, simply for correlation. We also used an upstream pipe flowing into the nacelle in order to see if this made any particular difference. We get a great big difference between flow-through data and faired-over nacelle data, but we are pretty well convinced that the exit conditions are by far the most important, and therefore if you have those reasonably well simulated, then you get the major effects.

BRADLEY:

I have the general feeling that the parametric results are questionable if they are based on faired-over inlets, insofar as local interference between the wing and the nacelle is concerned. I believe that attempts to contour the nacelle and wing to achieve favorable interference would be incorrect if based on experimental data with faired-over inlets. But you say you did correlate your data with flow-through inlet data?

RYLE:

I guess I feel that the power effects are so much more significant than any other parameter that once those are simulated, you probably have reasonable trends.

BRADLEY:

Maybe so, but I am not sure I agree with that.

RYLE:

Well, you work on fighters anyway, and we work on transports!

BRADLEY:

That is right. We do look at a little higher speed regimes sometimes. However, the fundamental flows are the same in transonic flow.

WEINRAUB: (Naval Air Systems Command)

Have you developed any methodology to determine what the implications of the model scale acoustic data would be on full scale?

RYLE:

Well, the only scaling data that we had was just the scaling

data between the anechoic room and the data which we got from the outdoor rig, and that is two and a half times. This correlation was as expected with the noise level being proportional to the flow area, with the spectrum being altered, from a frequency point of view, inversely proportional to the dimensions. That is fairly standard with respect to scaling.

DENNING: (Rolls-Royce (1971) Limited)

I wonder if you could explain why you have the flap angle in the Struhal number correlation. Is it a flap angle measured in radians, by the way?

RYLE:

Yes, I think so. It must be flap angle in radians.

DENNING:

I do not understand the physical significance.

RYLE:

I am not sure that I do either. It is an empirical factor which collapses the data. It just works!

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THRUST AUGMENTATION AND NOISE ATTENUATION OF EJECTOR SHROUDS

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ABSTRACT

Results of a study on the static aerodynamic performance and acoustic characteristics of a shrouded slot nozzle are presented (no external flow). Experiments were carried out on a slot nozzle of aspect ratio w/h = 27 with an ejector shroud having a total cross sectional area of about four times the area of the primary nozzle. The aerodynamic performance of the nozzle-shroud system is characterized by the ratio of entrained secondary air to primary air and the thrust of the system compared to that of an unshrouded reference nozzle. The acoustic data taken during the tests consisted of overall sound pressure level (OASPL) contours in a plane perpendicular to the nozzle span (flyover plane) and contours in the nozzle plane. The noise output of the device is generally characterized by the maximum sound pressure level in the flyover plane ($OASPL_{max}$). Some of the principal findings of this study are: (1) Both the aerodynamic performance and the noise attenuation characteristics of the ejector shroud were found to improve with increasing shroud length over the range tested. (2) Thrust performance of the longest ejector shroud configuration improved with shroud divergence angle up to an area ratio of 1.2 and thereafter decayed. Noise attenuation generally decreased as the shroud walls diverged. (3) For high-temperature primary flow at high subsonic Mach numbers, the noise-attenuating liners in the shroud were significantly quieter in comparison to the hard-wall shroud, possibly due to the increased noise generation by the higher velocity of the hot primary jet, rather than a direct effect of temperature.

INTRODUCTION

This paper is concerned with an investigation of the noise characteristics and thrust performance of a shrouded slot nozzle. A schematic diagram of a shrouded nozzle consisting of a primary nozzle that exhausts into a constant area duct is shown in Figure 1. External air is ingested by the ejector action and mixes with the primary air producing, ideally, a well-mixed flow stream at the end of the duct. The entrained secondary air flow provides an increase in the momentum of the exhaust jet, causing an increase in the thrust of the system as compared with the primary jet alone. The duct may

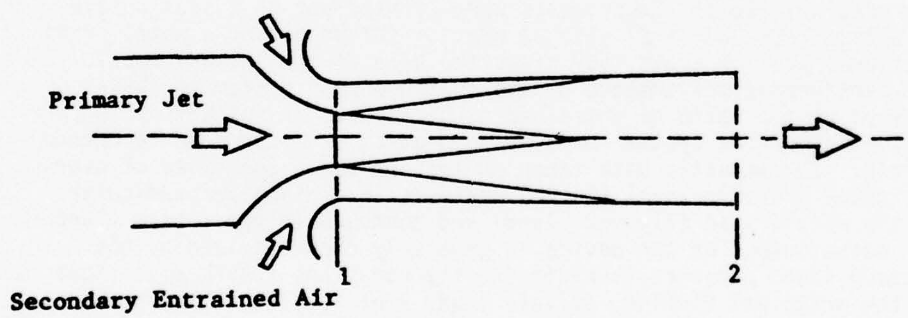


Figure 1. Schematic of Ejector Nozzle

be diverged or a diffuser added to improve the pressure recovery and cause a greater entrainment of secondary air.

Acoustically, the ejector represents an effective noise suppression device for three reasons: (1) the large secondary flow causes a reduction in the mean shear near the nozzle exit, (2) the jet stream exiting from the shroud has a velocity that is substantially reduced compared to the primary jet and (3) the shroud walls act as a noise shield. Further noise reduction can be obtained by fitting acoustic lining to the inner wall of the shroud.

A shrouded nozzle such as was investigated in the present study is more likely to be employed as a powered lift device for STOL aircraft. Such a device would be deployed during take-off and landing and folded back into the wing for cruise. Figure 2 shows a sketch of an augmentor wing in a take-off configuration and folded away for cruise. In powered lift operation air from the turbofan is ducted through the wing section and exhaust from a nozzle through the augmentor wing shroud. The enhanced thrust generated by the ejector action more than compensates for the duct losses and makes the augmentor wing one of the more attractive powered lift devices being considered for STOL aircraft. The aerodynamic advantages of this device have been discussed in some detail by Whittley (Ref. 1).

The primary impetus for the present work was the investigation by Goethert and Borchers (Ref. 2) in which experiments were carried out on a pair of circular nozzles in a rectangular shroud. Sound power measurements were made by Goethert and Borchers using both hard wall and acoustically lined ejectors of various shroud lengths. Results for the hard wall ejector showed an increase in sound power reduction, with increasing shroud length confirming the earlier results of Middleton (Ref. 3). The sound power reduction appeared to be approaching an asymptotic value of about 7 dB for hard-wall internal surfaces (no liners). A comparison with a theoretical calculation suggested that a substantial portion of the noise was being generated in the mixing region beneath the shroud and thus would be reduced by acoustically lining the ejector channel. Subsequent tests with a lined ejector produced a substantial further reduction in noise output, but caused unexpectedly large internal flow losses impairing the thrust gain by the ejector.

The results obtained by Goethert and Borchers were considered to be quite encouraging and worthy of further investigation. Based on the experience gained in these tests, a new experimental program guided by theoretical calculations was undertaken to attempt to optimize the acoustic and aerodynamic performance of a shrouded nozzle.

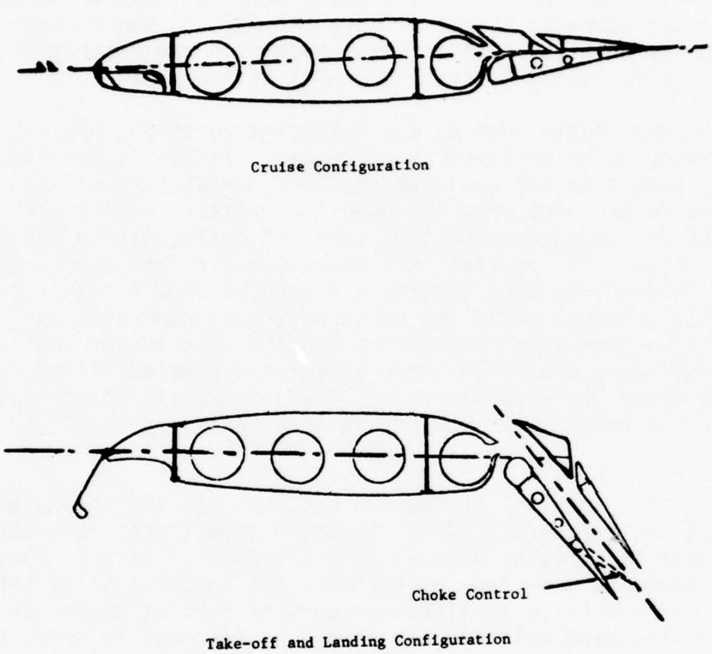


Figure 2. Augmentor-Wing Section

THEORETICAL ANALYSIS

Fluid dynamically the augmentor wing can be modelled satisfactorily by a simple ejector as shown in Figure 3. The ejector consists of a primary nozzle exhausting into a constant or variable area shroud. It is possible to apply the equations of one-dimensional fluid mechanics to the ejector to calculate the rate of air entrainment and the thrust augmentation. One of the first analyses of this type was performed by von Karman (Ref. 4) for incompressible flow. This work has been extended by Goethert and Borchers (Ref. 2) to take into account compressibility effects when the stagnation temperatures of the primary and secondary streams are equal. This analysis was further extended in the present study to include the

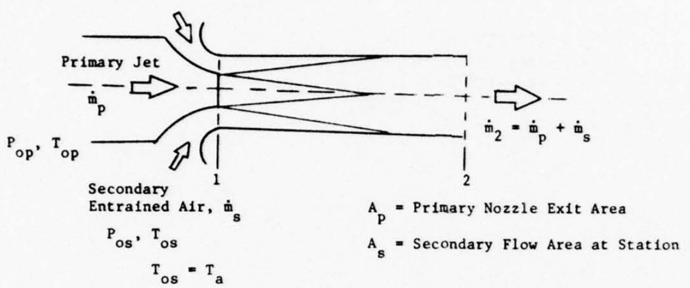


Figure 3. Shrouded Nozzle

effects of different stagnation temperatures.

In order to estimate the noise reduction and thrust augmentation obtainable with the shrouded nozzle, the results of the theoretical calculations for that system must be compared with a suitable reference nozzle. In the present analysis, as in Ref. 2, the reference nozzle was taken to be an isentropic nozzle operating at the same stagnation pressure and temperature as the shrouded primary nozzle and passing the same mass flow. The physical variables for the reference nozzle are typically indicated by a prime and thus the above conditions may be expressed by

$$P_{op} = P'_{op}$$

$$T_{op} = T'_{op}$$

$$\dot{m}_p = \dot{m}'_p$$

To satisfy these conditions, the area of the reference nozzle generally must be slightly different from the area of the primary

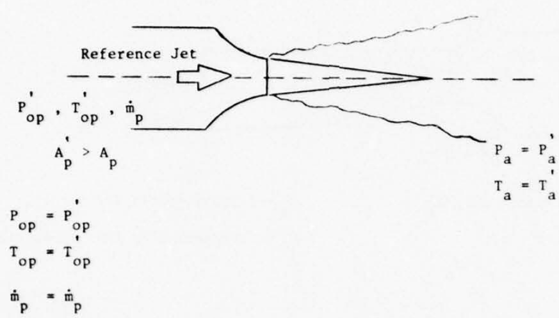


Figure 4. Reference Nozzle

nozzle. This is illustrated in Figure 4. The equality of the stagnation pressures, assuming that both systems exhaust into the same atmosphere can be expressed in terms of a reference Mach number, M_p', such that

$$\frac{P_{op}}{P_a} = \frac{P'_{op}}{P_a} = \left(1 + \frac{\gamma - 1}{2} M_p'^2\right)^{\frac{\gamma}{\gamma - 1}}$$

This basis of comparison is carried over to the experimental part of the research in that the results for the ejector are compared with a reference nozzle operating at the same reference Mach number, M_p'.

In order to calculate the noise reduction and thrust augmentation for the shrouded nozzle, flow variables must be determined to satisfy the following equations (see Figure 3):

a) Continuity Equation

$$\rho_p A_p V_p + \rho_s A_s V_s = \rho_2 A_2 V_2$$

b) Momentum Equation (parallel shroud walls)

$$\rho_p A_p V_p^2 + \rho_s A_s V_s^2 + P_1 A_1 = \rho_2 A_2 V_2^2 + P_2 A_2$$

For non-parallel walls, an additional term $\int_1^2 PdA$ must be added to the left hand side of the equation.

c) Energy Equation

$$\dot{m}_p h_{op} + \dot{m}_s h_{os} = \dot{m}_2 h_{o2}$$

d) Pressure Condition

$$\frac{P_2}{P_{os}} = \frac{P_a}{P_{os}} = 1$$

In addition, isentropic flow is assumed to prevail between the stagnation chambers and the nozzle exit at station 1.

The geometry of the shrouded nozzle system is specified by the ratio of the secondary to primary flow areas, $AR = A_s/A_p$, and by the shroud divergence ratio, $DR = A_2/A_1$. It is assumed that the primary nozzle lip has no area so that

$$A_1 = A_s + A_p$$

This is correct for the station immediately downstream of the nozzle lip.

The algebraic manipulations used to prepare the system of equations for an iterative solution are similar to those carried out in Reference 1 and will not be presented. The results of the theoretical calculations are expressed in terms of three parameters:

the mass flow ratio

$$K_m = \frac{\dot{m}_s}{\dot{m}_p}$$

the sound power reduction based on Lighthill's V^8-Law

$$\Delta PWL = 10 \log_{10} \frac{A'_p}{A_2} \left(\frac{V'_p}{V_2} \right)^8$$

and the static thrust ratio of the ejector system to the reference nozzle

$$TR = (1 + K_m) \frac{V_2}{V'_p}$$

It should be noted that the theoretical expression for the sound power reduction assumes that all of the noise generated by the turbulent mixing of the primary and secondary flow streams is absorbed beneath the shroud. Thus the ΔPWL values obtained from the analysis represent maximum values that would only be obtained for a perfectly absorbing shroud. Similarly, the thrust calculation assumes that the shroud is sufficiently long so that complete mixing occurs and does not take into account losses due to viscosity or flow separation. Thus, the calculations are for an ideal ejector and provide optimal values against which experimental results can be compared.

EXPERIMENTAL INVESTIGATION

Description of Model

A sketch of the shrouded nozzle used in the experimental investigation is shown in Figure 5. A rectangular geometry was chosen for both the primary nozzle and the shroud to take advantage of the better directional characteristics of the sound radiated from a high aspect ratio slot nozzle. Also, the noise radiated from a slot nozzle has a greater high frequency content than the noise from an equivalent area circular nozzle. This higher frequency noise is more easily attenuated.

The primary slot nozzle has an exit area of 2.4 in^2, an aspect ratio of 26.67, and a slot height of 0.3 inches. The shroud duct has a basic height of 1.2 and the same width, 8 inches, as the nozzle. This gives a nominal area ratio for the ejector $A_s/A_p = 3.0$ which is not unreasonable for STOL applications.

The shroud is made in three sections, each 5 inches long, so that shroud lengths of 5, 10 and 15 inches could be tested. The sketch in Figure 5 shows two sections of the shroud in place. The

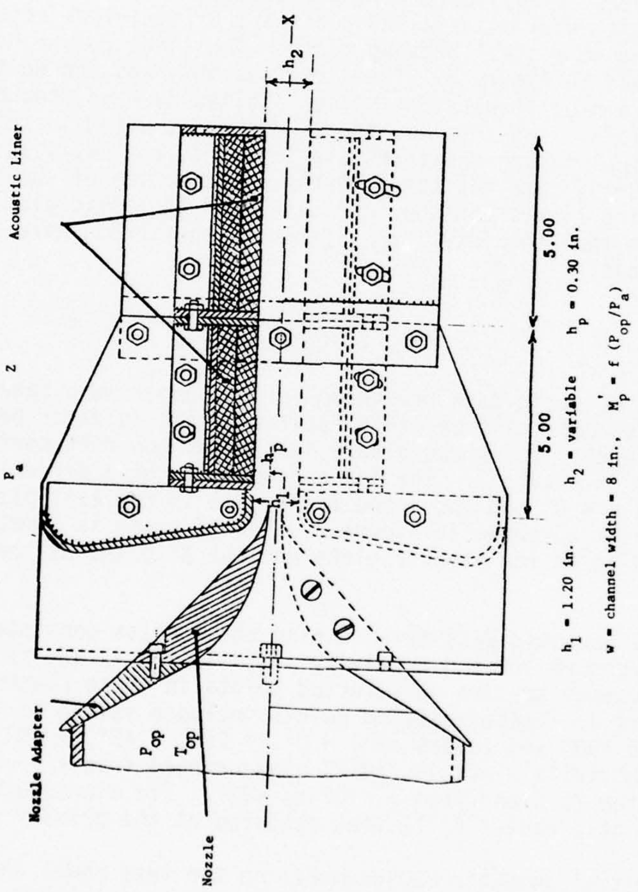


Figure 5. Sketch of Ejector-Shroud Model

shroud walls can be diverged by rotating them about pins in the side walls at the primary nozzle exit station. In this way, the shroud divergence ratio can be varied continuously from $A_2/A_1 = 0.9$ to $A_2/A_1 = 1.4$ for all three shroud lengths.

The inner surfaces of the shroud were made of solid aluminum blocks which could be removed and replaced by acoustically absorbing liner blocks. The liners were constructed from two sheets of a porous felt metal material spaced apart by stainless steel honeycomb and bonded to a steel backing plate. The liner construction is illustrated in Figure 6. The liner assembly was bonded together by using a high temperature solder applied by hand around the edges of the block. Two sets of liners were constructed in this manner with nominal facing sheet resistances of 10 cgs Rayls and 30 cgs Rayls. Great care was taken in the construction of the noise attenuating liners and the fabrication of the solid aluminum blocks to assure that the interior surfaces of the shroud were as smooth as possible.

Test Procedures

The acoustic data presented in this paper were taken in the UTSI aeroacoustics free field facility which is described in detail in Reference (5). A coordinate system for the test configuration is shown in Figure 7. The jet exhausts in the X direction along the axis $\theta = 0°$ and the Y and Z axes lie in the exit plane of the primary jet as shown in Figure 7. The XZ plane is sometimes referred to as the flyover plane and the XY plane as the sideline plane.

The acoustic data taken during these tests consisted of overall sound pressure level directivity plots in the XZ and YZ planes and the frequency spectra at selected points in these planes as shown in Figure 7. These selected points included values of $\theta = \pm 30°$, $\pm 60°$, and $+90°$ and values of $\phi = 0°$, $+ 25°$, $+ 45°$, $+ 65°$, and $+90°$. The directivity plots in the XZ plane ranged from $\theta = -60°$ to $+90°$ and in the YZ plane from $\phi = 0°$ to $180°$. The microphones were located at a radius of 13 feet centered at the primary nozzle exit.

Initial acoustic measurements on the test model showed that the sound radiated to the $\theta = + 90°$, $\phi = +90°$ position (See Figure 7) contained noise which was radiated from the ejector inlet. Since it was the purpose of this study to investigate the noise reduction potential of the ejector geometry exclusive of any sound radiated from the inlet, a sound shield was mounted as shown in Figure 7. The shield was made of 8 ft. x 4 ft. x 1/2 in. thick plywood with 4 inch thick foam glued on both sides. The shield was mounted on the model so as to make an angle of $\theta \simeq + 120°$ to the jet axis. This

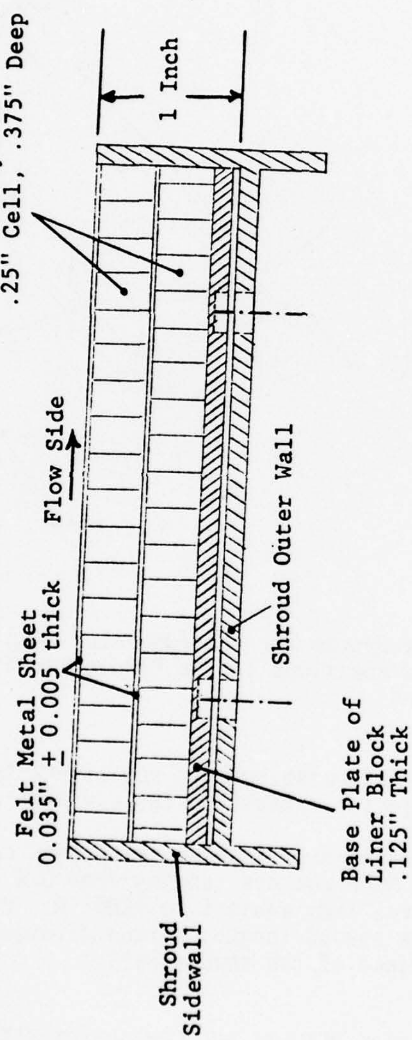


Figure 6. Sketch Showing Shroud Wall Lining

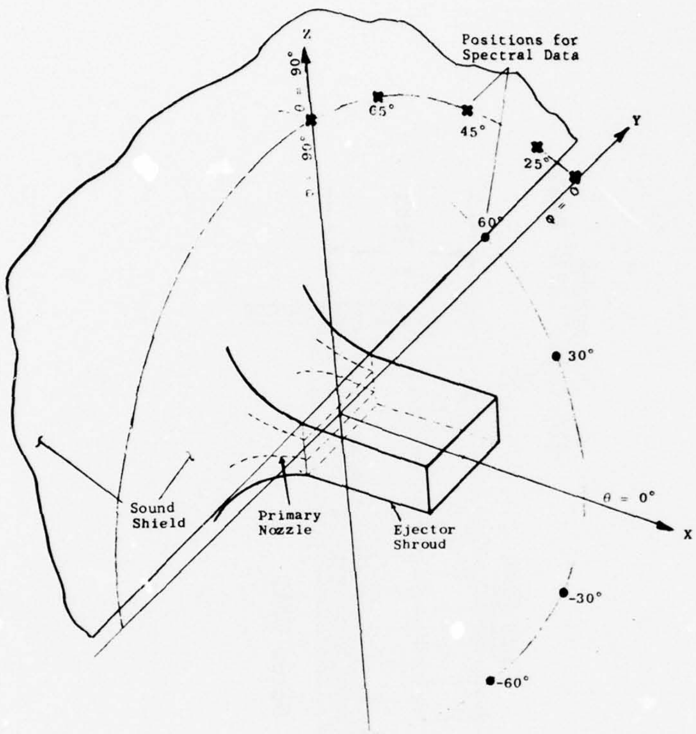


Figure 7. Coordinate System for the Augmentor Wing Indicating Planes and Positions in the Planes For Acoustic Measurement

set-up blocked the inlet noise, but did not affect the sound radiating from the ejector exhaust to the θ = +90° position.

In the experimental investigation, tests were conducted for various primary flow Mach numbers ranging from 0.5 to choking and stagnation temperatures from ambient to 1200° R. Parameters varied during the tests were shroud length, shroud divergence area ratio, and acoustical impedance of the shroud wall.

EXPERIMENTAL RESULTS AND COMPARISON WITH THEORY

Fluid Dynamic Results

In order to determine how well the model ejector shroud was

functioning fluid dynamically, measured values of the ratio of secondary mass flow to primary mass flow (K_m) and thrust ratio (TR) were compared to the values predicted by one dimensional theory. Experimental values for the primary and secondary mass flow rates were determined from the static pressure measured at the nozzle exit plane and the primary and secondary stagnation pressures.

Figure 8 compares the mass flow ratio (K_m) obtained from the model ejector system for both hot and cold shrouded jets with the theoretically predicted values. For low divergent ratios, $A_2/A_1 \leq 1.2$, the 15 inch shroud data shows excellent agreement with the prediction. This indicates that these shrouds are sufficiently long so that the primary and secondary streams mix thoroughly beneath the shroud. Although the mass flow ratio for the 15 inch shroud increases with increasing area ratio over the entire range tested, for the larger values of A_2/A_1 the data is noticeably below the prediction. The trends for the 10 inch shroud are similar to those for the 15 inch shroud, except that the divergence between experimental and predicted values of K_m begins to occur at an area ratio slightly greater than one. Obviously, the 5 inch shroud is not long enough to produce good mass induction, and this configuration was therefore not tested extensively.

Figure 9 shows the measured values of the thrust augmentation compared with theoretically calculated values. The experimental values peak at $A_2/A_1 = 1.2$ for the 15 inch shroud and at $A_2/A_1 = 1$ for the 10 inch shroud. However, these values are considerably below the predicted values even for the 15 inch shroud. This is

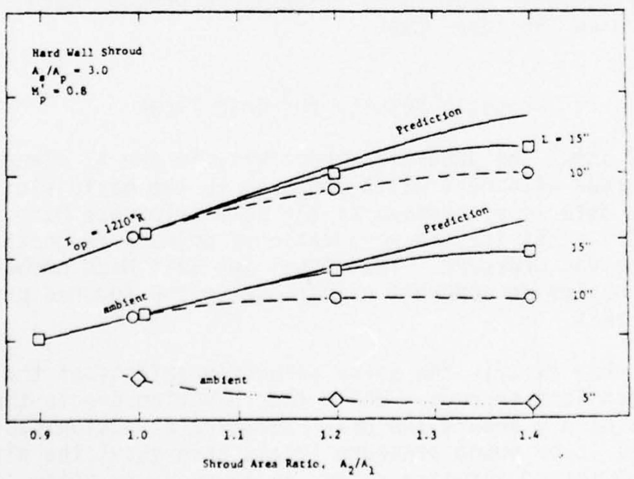


Figure 8. Comparison of One-Dimensional Theoretical and Experimental Results of Mass Flow Ratio for Shroud Nozzle (Hard Walls)

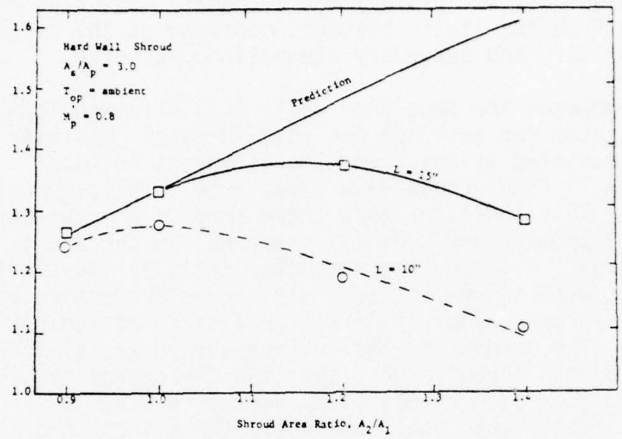


Figure 9. Comparison of One-Dimensional Theoretical and Experimental Results of Thrust Ratio for Shroud Nozzle (Hard Walls)

because the thrust ratio depends on the product K_mV_2 and pumping at a lower rate than ideal K_m will also cause the shroud exit velocity V_2 to be lower than ideal. Hence the product K_mV_2 is considerably below the ideal case.

Acoustic Results for Cold Flow

Figure 10 shows the acoustic directivity in the XZ plane of the ejector shroud with hard walls compared to the basic slot nozzle. These data were obtained at the same reference Mach number, M'_p = 0.9, that is, the same ratio of primary stagnation pressure to ambient pressure. The actual jet exit Mach number for the shrouded nozzles is somewhat higher due to the reduced pressure at the nozzle exit.

The data show clearly the noise reduction ability of the 10 inch and 15 inch long shrouds. While the reduction due to the 10 inch shroud is mainly around the peak radiation direction, the 15 inch shroud has lower sound pressure levels throughout the plane. The 5 inch long shroud actually causes an increase in noise throughout the XZ plane. Figure 11 shows that the noise reductions obtained with the longer shrouds are primarily due to decreases in the components above 1000 Hz.

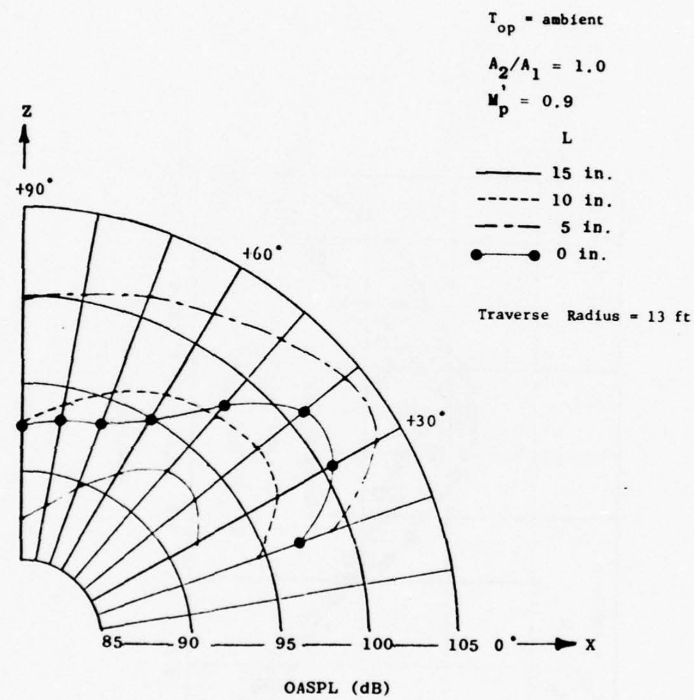


Figure 10. OASPL Directivities in XZ Plane for Three Ejector Lengths (Hard Walls)

Figure 12 shows that as the divergence ratio (A_2/A_1) increases, the overall sound pressure level increases throughout the XZ plane. Figure 13 shows that this increase is due primarily to the increase of noise components near the peak frequency. This trend is contrary

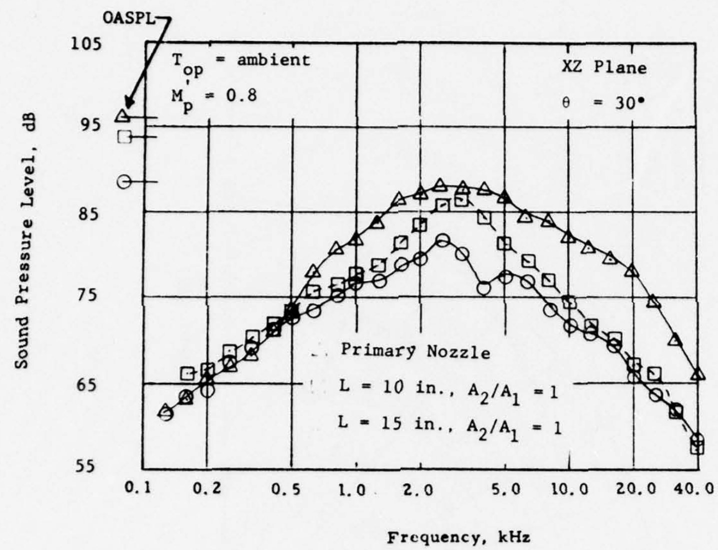


Figure 11. Sound Pressure Spectra for the Primary Nozzle and the 10 and 15 Inch Shrouded Nozzle (Hard Walls)

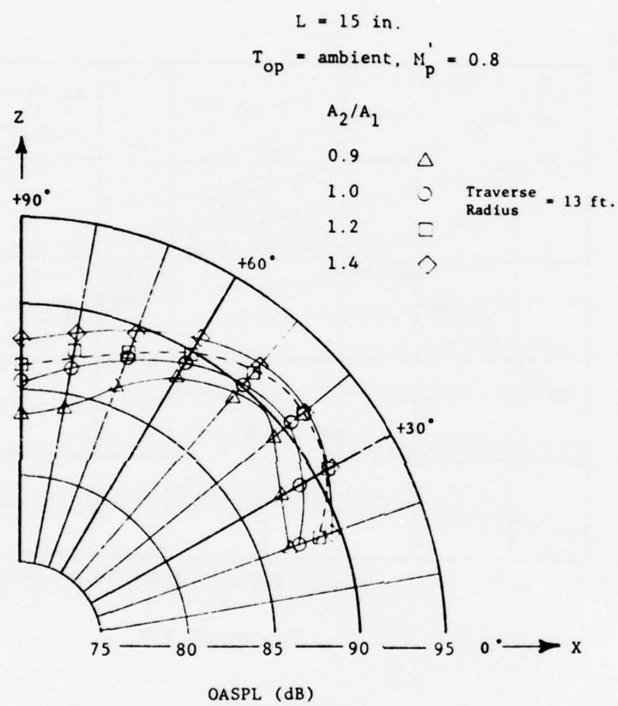


Figure 12. OASPL Directivities in XZ Plane for 15 Inch Shrouded Nozzles with Different Area Ratios (Hard Walls)

to what is predicted by the theory, since increasing the divergence ratio does cause a reduction in mean shroud exit velocity and hence should produce a reduction in noise generated by the shroud jet.

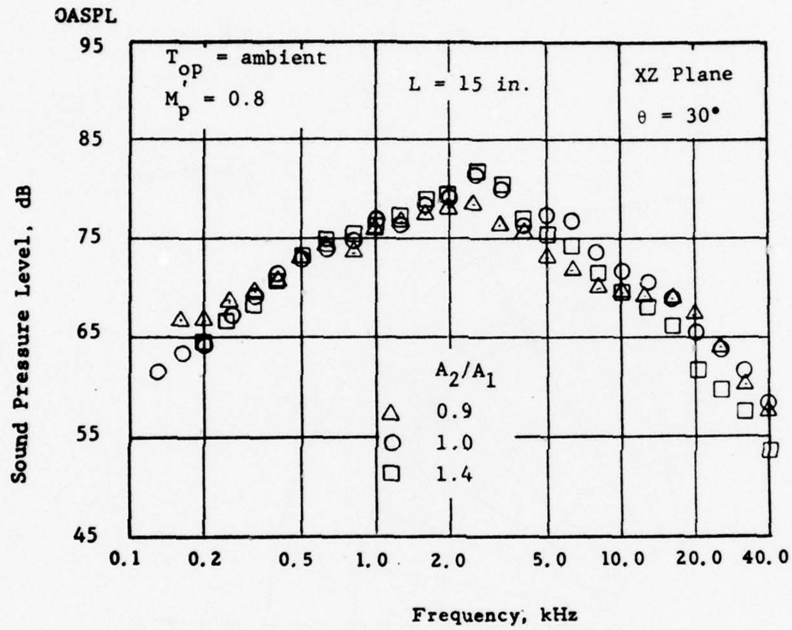


Figure 13. Sound Pressure Spectra for 15 Inch Shrouded Nozzles with Different Shroud Area Ratios (Hard Walls)

The result suggests that the increasing divergence ratio allows more of the noise generated beneath the shroud to escape to the far field. Indeed, the diverging channel may be acting as a horn to amplify sound generated by the primary jet.

Figure 14 gives a summary of the acoustic data for hard wall shrouds and cold primary flow. The ordinate in this figure is

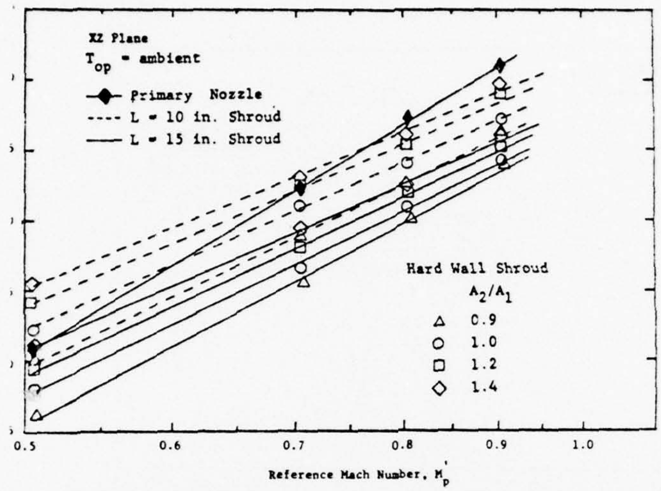


Figure 14. Maximum Sound Level Variation with Reference Mach Number for 10 and 15 Inch Shrouded Nozzle

maximum overall sound pressure level in the XZ plane - this normally occurs around 45° to the jet axis for cold flow. The abscissa in Figure 14 is the reference Mach number, which allows the most meaningful comparison between the shrouded nozzle data and data for the primary nozzle alone. In addition to re-emphasizing the trends cited previously with respect to shroud length and divergence ratio, this figure shows that the noise reduction achieved by the shroud relative to the primary nozzle increases with increasing reference Mach number.

Since noise reduction and thrust augmentation are the most important quantities characterizing the performance of an ejector system, it would be interesting to examine the variation of one with respect to the other. This is done in Figure 15 for both the 10 and 15 inch long shrouds operating at different pressure ratios and shroud divergence ratios. The 15 inch shrouds are clearly superior in thrust augmentation as well as noise reduction capability. The 15 inch shroud with shroud area divergence ratio equal to 1.2 produces maximum thrust augmentation at all operating pressures. The noise reduction, however, was measured to be a maximum when the shroud area ratio (A_2/A_1) was equal to 0.9 and might further increase as the area ratio decreases. For any given shroud geometry, the noise reduction increases with increasing reference Mach number. The thrust augmentation remains constant or decreases only slightly (5%) with increasing nozzle pressure ratio.

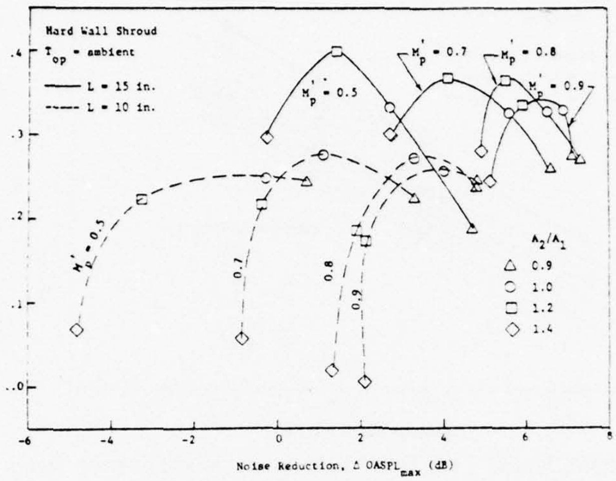


Figure 15. Variation of Thrust Augmentation with Noise Reduction for 10 and 15 Inch Shrouded Nozzles

The ejector system thus seems to be more efficient at higher operating pressure ratios. This should be particularly noted for the 10 inch shroud which is successful in attenuating the noise only at higher reference Mach number.

Effect of Noise Attenuating Liners

Figure 16 shows the OASPL directivities in the XZ plane for the 15 inch shroud with noise attenuating liners compared to corresponding data for the hard wall shroud ($R = \infty$). The use of acoustic lining is seen to reduce the noise level throughout the plane by about 2 dB below hard wall shroud for the test conditions shown. There is no significant difference, under these conditions, between the noise attenuation obtained from the lines made from 30 Rayl facing sheet and those made with the 10 Rayl material. Figure 17 shows that the further attenuation achieved with the lined shroud is due to reduction of components above 2000 Hz. Similar comparative for the 10 inch shroud showed no further reduction due to the liner.

Figure 18 shows a comparison of the lined and unlined ejectors as a function of shroud exit velocity. Also shown in this figure are data taken for the 15 inch shroud alone; that is with the primary nozzle removed and a contoured transition provided between

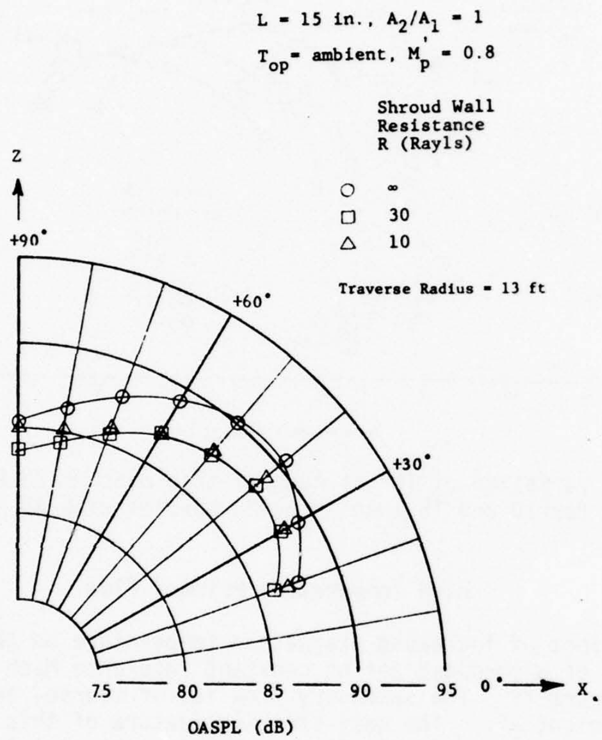


Figure 16. OASPL Directivities in the Flyover Plane of 15 Inch Shrouded Nozzles with Hardwall and Lined Shrouds

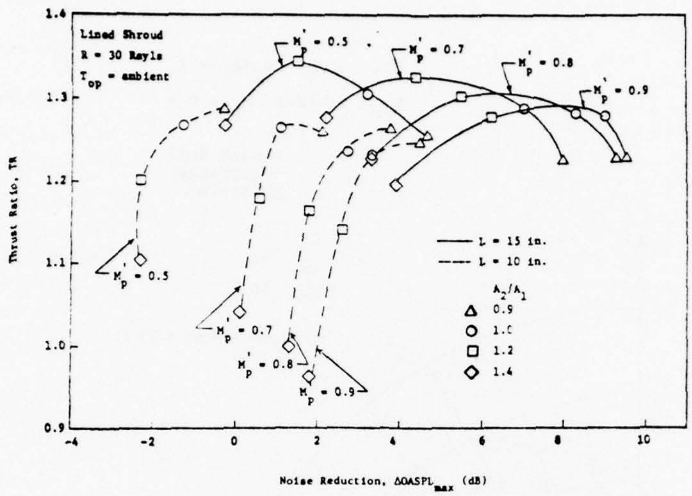


Figure 20. Variation of Thrust Augmentation with Noise Reduction for 10 and 15 Inch Shrouded Nozzles with 30 Rayl Liners

High Temperature Primary Flow

The effect of increased stagnation temperature on the sound directivity of a shrouded jet at constant reference Mach number is shown in Figure 21. The secondary flow is, of course, entrained from the ambient air. The most striking feature of this figure is the increased sound levels that occur at higher jet temperatures. This is mostly due to the increased sound generation of the higher velocity primary jet. The difference in temperature between the exhaust jet and the surrounding air also results in an increased refraction of sound generated by the jet. This causes the direction of maximum noise radiation to be rotated farther from the jet axis for high temperature primary flow.

Figure 22 shows the directivity patterns of the 10 inch and 15 inch hard wall shrouds compared with unshrouded primary jet. The peak noise for the two shrouds is very nearly the same, but the directivity of the 15 inch shroud has a much sharper peak. Away from the direction of peak radiation the longer shroud is quieter by about 3 dB. For $\theta < 60°$, both shrouds produce a substantial reduction in sound pressure level compared to the primary jet. Figure 23 shows that this is primarily due to reductions in the mid and high frequency components.

Figure 24 summarizes the acoustic results for hard wall shrouds

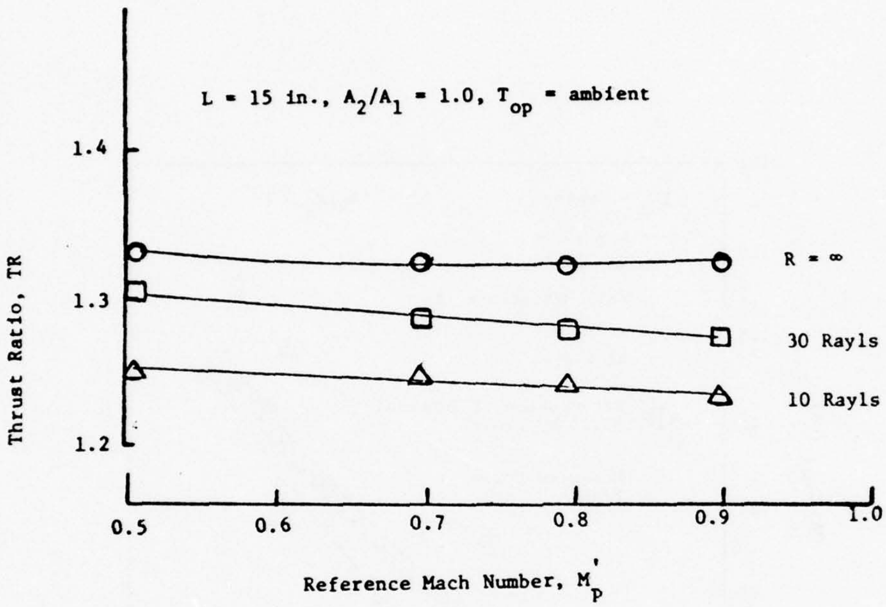


Figure 19. Effect of Shroud Wall Resistance on the Thrust Augmentation of 15 Inch Shrouded Nozzle

both the 10 and 30 Rayl lined shrouds, the thrust augmentation is clearly decreased with decreasing shroud wall resistance. This is thought to be due to increased friction and to reversed flow occurring in liner blocks. Although care was taken in the construction of the liner to minimize these effects, they could not be eliminated entirely.

Figure 20 summarizes the acoustic and thrust data for the lined shroud using the 30 Rayl facing sheet. The noise reduction figures on the abscissa are with respect to the unshrouded nozzle at the same reference Mach number. The general features of these figures are the same as for the hard wall shroud, although the noise reduction values are somewhat larger and the thrust ratio values smaller.

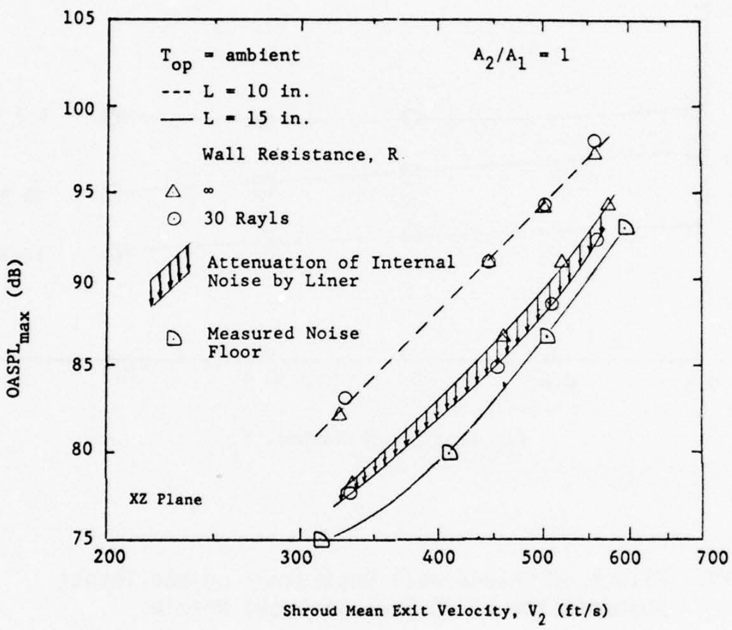


Figure 18. Variation of Maximum Sound Level with Shroud Exit Velocity for Hard Wall and Lined Shrouded Nozzles

produce a modest improvement for this configuration. The maximum sound pressure level for the 15 inch lined shroud at high velocities is only about 1 dB above the noise floor.

The effect of shroud liner on thrust augmentation is shown in Figure 19. Although the same noise reduction was obtained with

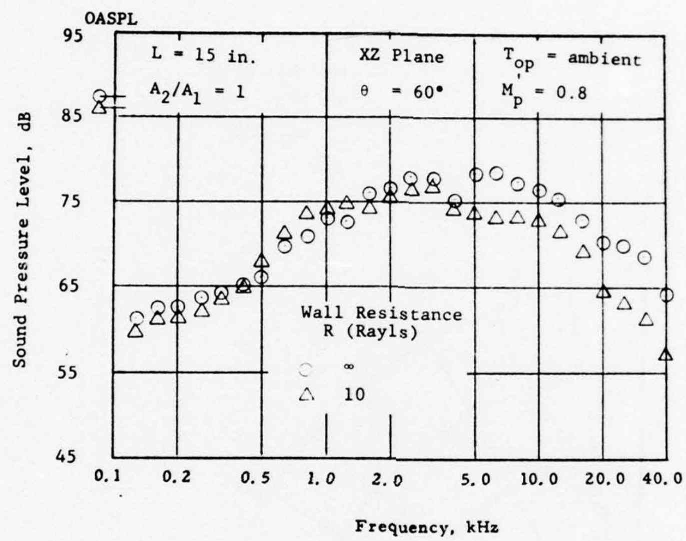


Figure 17. Sound Pressure Spectra of 15 Inch Shrouded Nozzles with Hard Wall and Lined Shrouds

the stilling chamber and the ejector shroud. The noise from this configuration should be due entirely to the shroud exhaust jet and should represent the noise floor for the ejector system, attainable only for complete absorption of all noise generated beneath the shroud. The data show that the 15 inch ejector with hard walls is within 5 dB of the noise floor over the entire operating range. Consequently the addition of noise attentuating liners can only

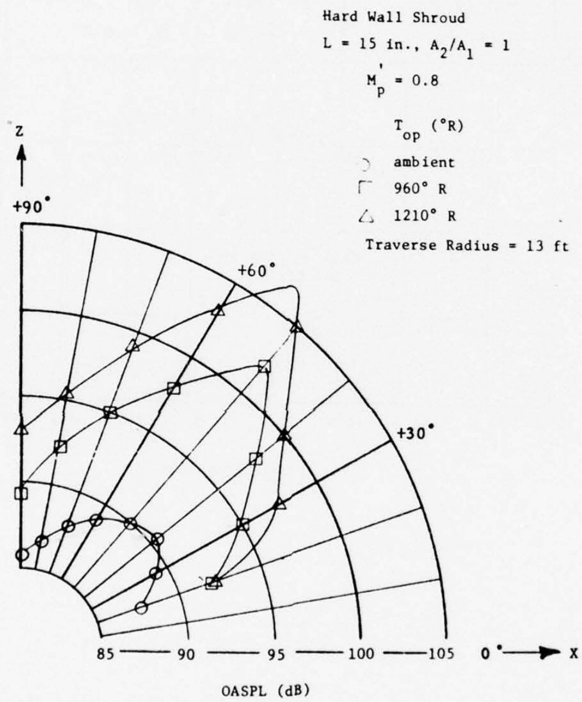
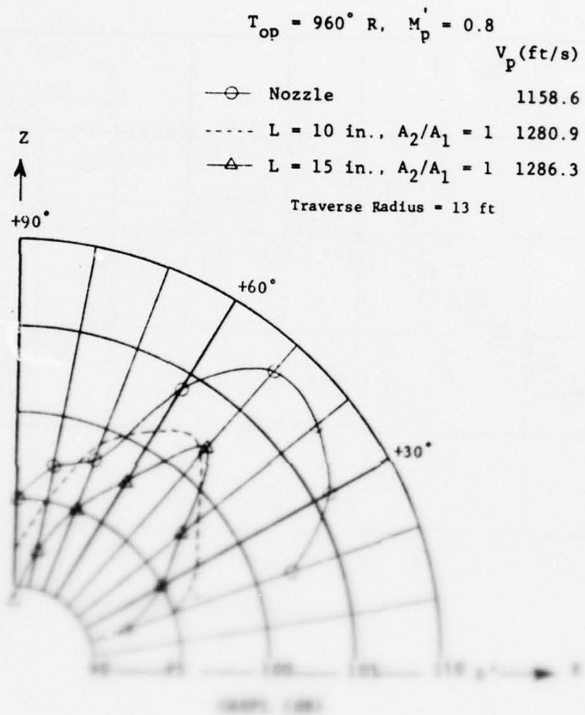


Figure 21. OASPL Directivities in Flyover Plane of 15 Inch Shrouded Nozzles with Hot and Cold Primary Jets

with high temperature primary flow. This figures shows that a 5 dB reduction in the maximum noise level can be achieved with the 10



T_op = 960° R, M'_p = 0.8
V_p (ft/s)
Nozzle 1158.6
L = 10 in., A_2/A_1 = 1 1280.9
L = 15 in., A_2/A_1 = 1 1286.3
Traverse Radius = 13 ft
Z
+90°
+60°
+30°

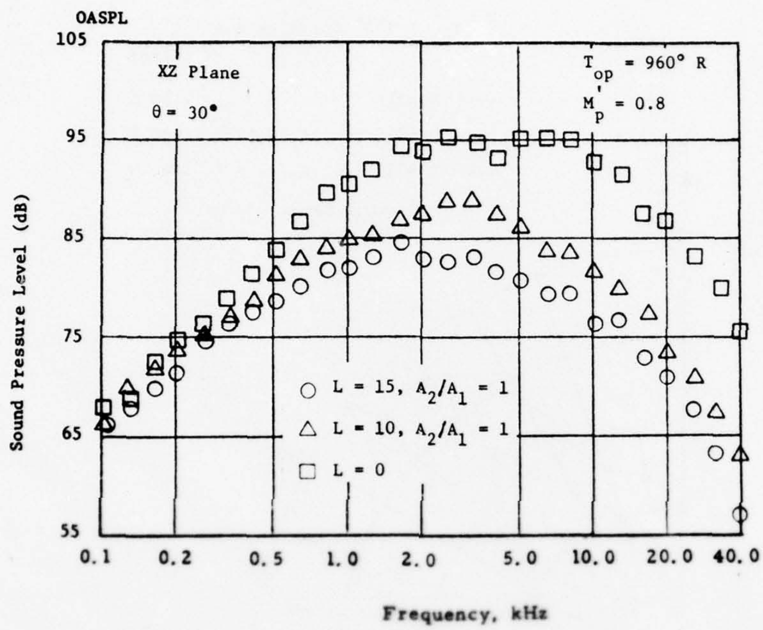


Figure 23. Sound Pressure Spectra for Unshrouded Hot Jet and 10 and 15 Inch Shrouded Hot Jets (Hard Walls)

inch shroud at high reference Mach numbers. Although little further reduction in $OASPL_{max}$ is obtained by increasing the shroud length to 15 inches, the sound levels in other directions have been seen to decrease by as much as 2 dB.

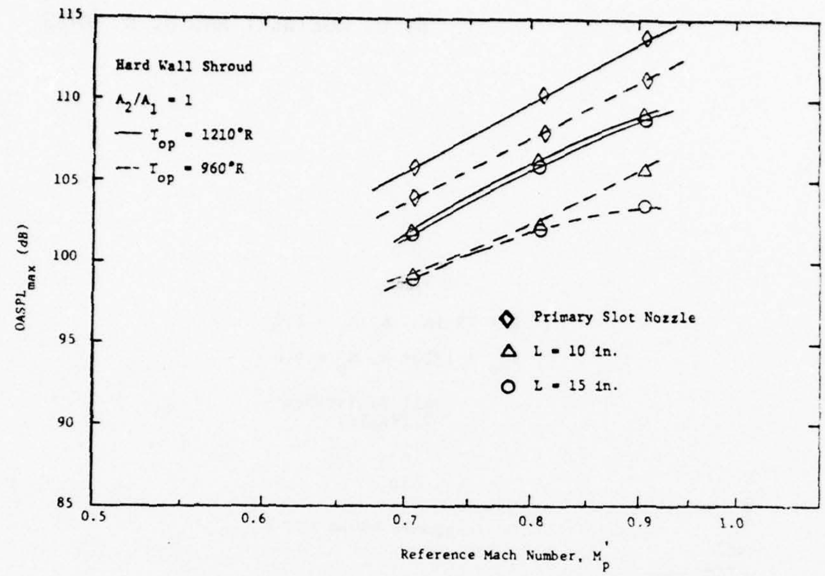


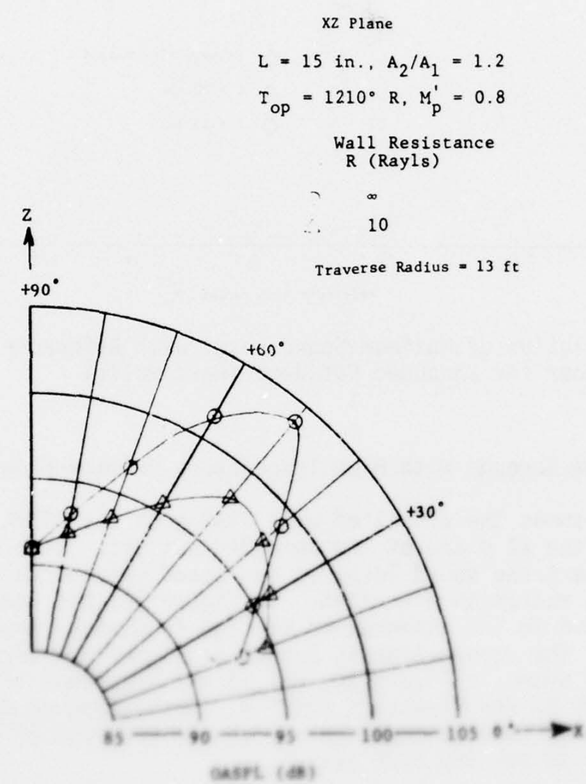
Figure 24. Variation of Maximum Sound Level with Reference Mach Number for Shrouded Hot Jets (Hard Walls)

Lined Shrouds with High Temperature Primary Flow

Figure 25 shows the effect of wall lining on the OASPL directivity in the XZ plane of the shrouded hot jet. With the use of acoustic liners, the sound level is decreased considerably around the peak radiation direction. In Figure 26, this reduction is seen to be due to the attenuation of high frequency components ($f > 2000$ Hz). The attenuation is larger at higher jet temperatures. For example, as shown in Figure 26, the 10 kHz component is attenuated 14 dB by the liners at 1210° R. Corresponding data at lower temperatures show attenuations in the 10 kHz band of 11 dB at 960° R and 3 dB for the cold jet.

By assuming that jets emerging from the unlined and lined shrouds at the same exit velocity (V_J) make equal contributions to the far field sound, the capability of the liners to absorb the sound generated beneath the shroud can be evaluated. [illegible]

[illegible]


XZ Plane
L = 15 in., A_2/A_1 = 1.2
T_{op} = 1210° R, M_p' = 0.8
Wall Resistance
R (Rayls)
∞
10
Traverse Radius = 13 ft
Z
+90°
+60°
+30°
85
90
95
100
105
0°
X
OASPL (dB)

[illegible]

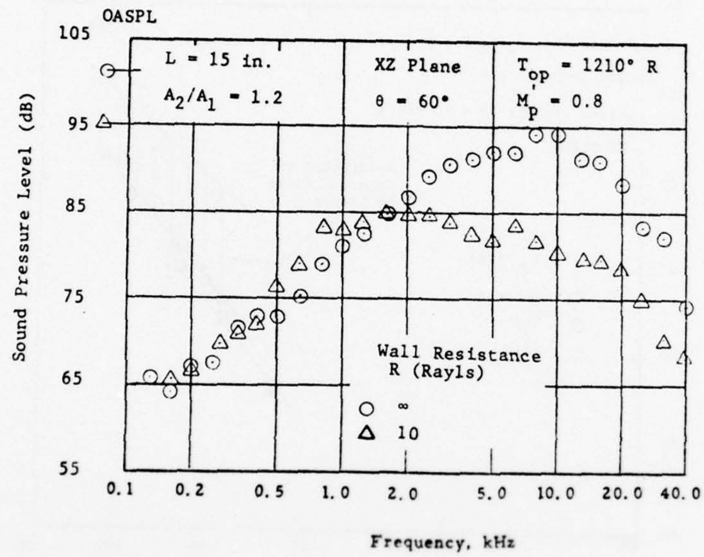


Figure 26. Sound Pressure Spectra for 15 Inch Shrouded Hot Jets with Hard Wall and Lined Shrouds

for high temperature flow, is to a large extent, due to the fact that the higher velocity hot jets are generating more internal noise beneath the shroud than the cold jets. It has been shown (Figure [illegible]) that in some cases for cold flow, the far field noise for the hard wall shroud is only a few dB above the noise floor. Thus only small further reduction can be achieved by the addition of liners. This

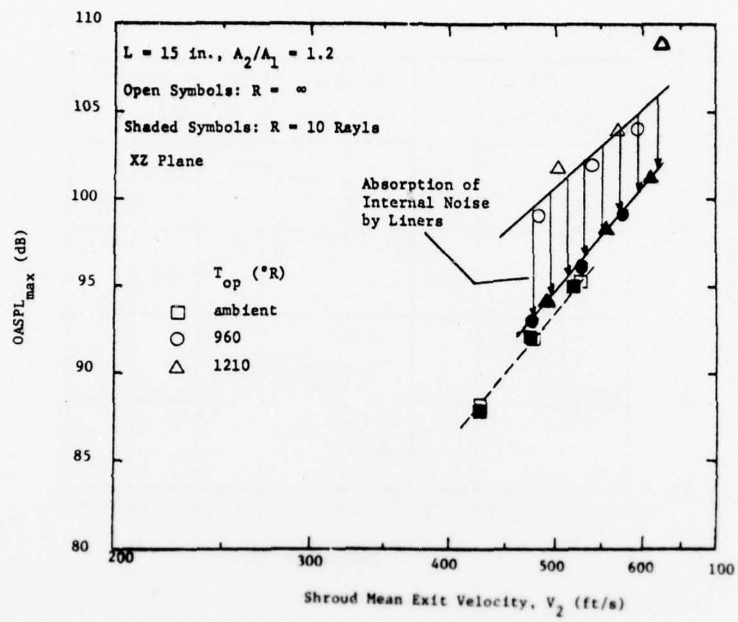


Figure 27. Variation of Maximum Sound Level with Shroud Exit Velocity for 15 Inch Shrouded Jets with Hard Wall and Lined Shrouds

is not the case for the high temperature primary jet - the noise from the hard wall shroud is still well above the noise generated by the shroud exhaust jet. The internal noise for the hot jets is still the major noise source and this can be reduced by the use of

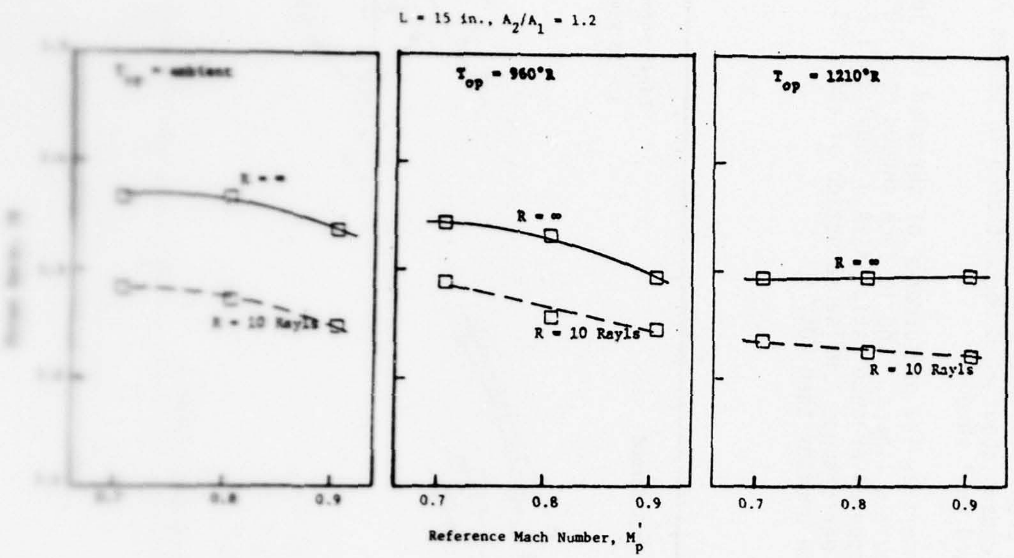
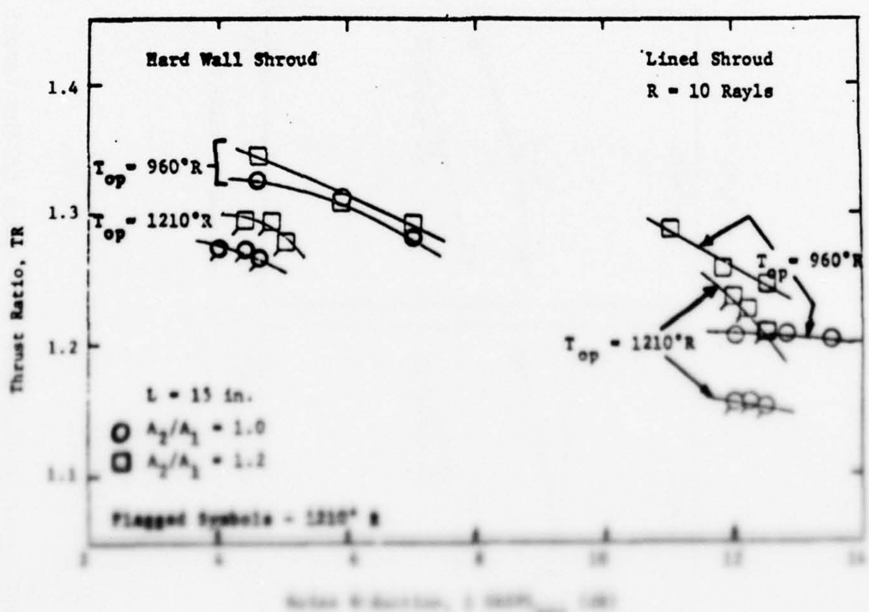


Figure [illegible]. Effect of Shroud Wall Resistance on the Thrust Augmentation of 15 Inch Shrouded Jets

attenuating liners.

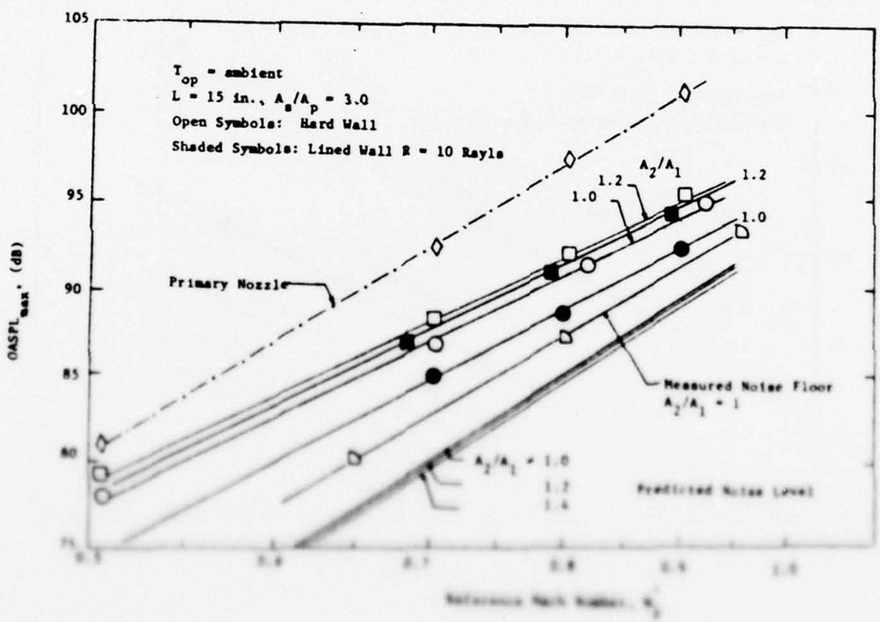
Figure 28 shows the effect of shroud wall impedance and primary stagnation temperature on the thrust augmentation of the shrouded nozzle. Increasing the primary jet temperature causes a slight degradation of the propulsive performance of the shrouded nozzle. The 10 Rayl liner causes about an 8 percent drop in thrust ratio compared to the hard wall shroud.

Figure 29 summarizes the performance of shrouded hot jets with hard walls and acoustic liners. The 15 inch shroud with 10 Rayl liner wall and shroud divergence ratio (A_2/A_1) equal to 1.2 is seen to have excellent performance - noise reduction of about 13 dB and thrust augmentation greater than 20%.

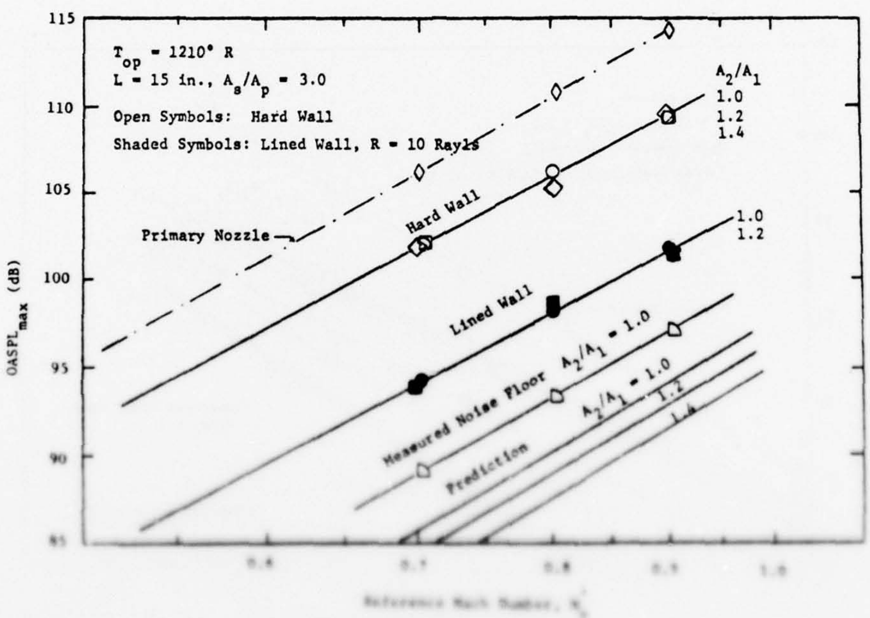


Comparison with Theory

Comparison of measured acoustic data for the 15 inch shroud with the theoretical predictions based on one dimensional inviscid flow calculations are presented in Figure 30 for cold primary flow and in Figure 31 for T_{op} = 1210° R. The predicted noise levels



given in these figures were obtained by scaling the primary nozzle data according to the V^8 law.



$$\Delta OASPL_{max} = 10 \log_{10} \frac{A'_p}{A_2} \left(\frac{V'_p}{V_2}\right)^8$$

This relation assumes that the maximum intensity scales just like the sound power and thus does not account for the forward focussing effect due to source convection. Also ignored in this calculation is the effect of nozzle aspect ratio on the directional characteristics of the radiated sound. Considering these shortcomings, it was felt that the theoretical prediction could not be used to reliably determine the noise floor of the ejector system. For this reason the shroud inlet was modified and acoustic data was taken for flow only through the shroud. These data were taken on the 15 inch, parallel wall shroud ($A_2/A_1 = 1$) for ambient stagnation temperature and a stagnation temperature of 810° R corresponding to $T_{op} = 1210°$ R. The cold flow data were presented in Figure 18 as a function of shroud exit velocity. The change in stagnation temperature had no measurable effect on the data shown in that figure. The data have been replotted in terms of M'_p in Figures 30 and 31. These data represent the actual noise floor for the 15 inch shroud with $A_2/A_1 = 1$. In all cases the measured noise floor is above that predicted on the basis of scaled primary nozzle data. The calculated values are, however, always within 3 dB of the experimentally determined values.

Of particular interest in Figures 30 and 31 is how closely the noise level for the parallel wall 15 inch lined shroud approaches the noise floor. For the cold flow case the lined shroud noise level is within 1.5 dB of the noise floor at $M'_p = 0.9$. This corresponds to a noise reduction of 9 dB compared to the primary nozzle. At $T_{op} = 1210°$ the noise from the shrouded nozzle is within 5 dB of the noise floor corresponding to a noise reduction of 13 dB compared to the primary nozzle. These comparisons show that the noise attenuating liners for the 15 inch shroud with $A_2/A_1 = 1$ are effective in attenuating a large proportion of the sound generated beneath the shroud. A potential for further noise reduction does exist, but such reduction will become progressively more difficult to achieve as the theoretical limit is approached.

[illegible]

[illegible]

total temperature.

The overall best results were obtained with a shroud length of about 12 times the shroud height. A solid, parallel wall shroud of this length produced a noise reduction of about 7 dB and a thrust augmentation of about 35% for a near choked, cold primary jet. A shroud length of about 8 times the shroud height gave substantially inferior values for noise attenuation and particularly for thrust augmentation. Obviously, the primary jet is not capable of mixing as well under the shroud of reduced length.

A few tests were conducted for a very short shroud with a length of only 4 times the shroud height. The results imply that the nozzle flow does neither substantially entrain secondary air nor mix appropriately with it. Thus, the short shroud is not only ineffective in noise attenuation, but was also generally found to be accompanied with a thrust reduction instead of a thrust increase.

Theoretical calculations on ejector shrouds were carried out assuming inviscid, one-dimensional flow with complete mixing underneath the shroud. A comparison of experimentally determined values of secondary air entrainment and thrust augmentation with these calculations shows good correlation for the longest shroud with parallel or slightly diverged walls.

Use of noise attenuating liners in the shroud produced little further noise reduction for cold flow. However, for high temperature primary flow at high subsonic Mach numbers the lined shroud was significantly quieter than the hard wall shroud. This is believed to be primarily due to the increased noise generation by the higher velocity primary jet rather than a direct effect of temperature. A noise reduction of 13 dB was obtained from the lined shroud with near choked, high temperature primary jet. The corresponding thrust augmentation was approximately 20%.

The noise floor for the ejector system was determined by making sound measurements for flow through the shroud only with the primary nozzle removed. The far field noise from the longest acoustically lined shroud with parallel walls was found to be within 5 dB of the noise floor for high temperature primary flow and within 2 dB of the noise floor for cold flow.

The remarkable potential of properly designed ejector shrouds capable of significantly reducing the exhaust noise (13 dB), [illegible]

of ejector shrouds even further beyond the potential identified in the current experiments.

The effectiveness of super-mixing devices should be examined to shorten the mixing length between primary and secondary flow and thus also the required length of the shroud. Based on experiments of other investigators (Ref. 6), the shroud length could be reduced to at least 1/2 of its current length without penalty of thrust augmentation. However, the effect of super-mixing on noise is not known and needs to be determined.

The design of noise attenuating liners for ejector shrouds should be studied in more detail to define optimum liner configurations. Such improved configurations are expected to increase further the noise attenuation of shrouds and reduce the sizeable penalty caused by the current liner design.

In the current experiments, the shrouds are designed either with constant area throughout their length or with constant divergence or convergence. It is believed that a more effective design for rapid mixing would be to have a constant area upstream section followed by a suitably selected divergent part.

ACKNOWLEDGMENT

Appreciation is expressed to the Federal Aviation Administration for support of this research under Contract DOT-FA72WA-3053. The authors are especially indebted to Messrs. R. S. Zuckerman and R. Padgett for their review of the research progress and continuous valuable encouragement. Mr. R. Padgett served as the Technical Officer for this project in the FAA Aircraft Noise Research Group headed by Mr. R. S. Zuckerman.

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DISCUSSION

CRAGIN: (General Dynamics)

In the first part of your presentation you were showing the nozzle with the plate underneath and different lengths of plate. If you said it I missed it, but over what pressure ratio ranges were you operating those nozzles?

GOETHERT:

Under this contract we are restricted to subsonic flows, so the maximum point on our curves was just below choking of the exhaust nozzle.

CRAGIN:

Next, you said that you can conclude you had no thrust loss with this type of situation. Now, even with a 132 plate, is there no appreciable scrubbing drag there?

GOETHERT:

Well, I was a little inaccurate on this. You get an additional skin friction drag behind the nozzle. We have some data on this. But this drag is relatively small. But there is some additional drag, you are correct.

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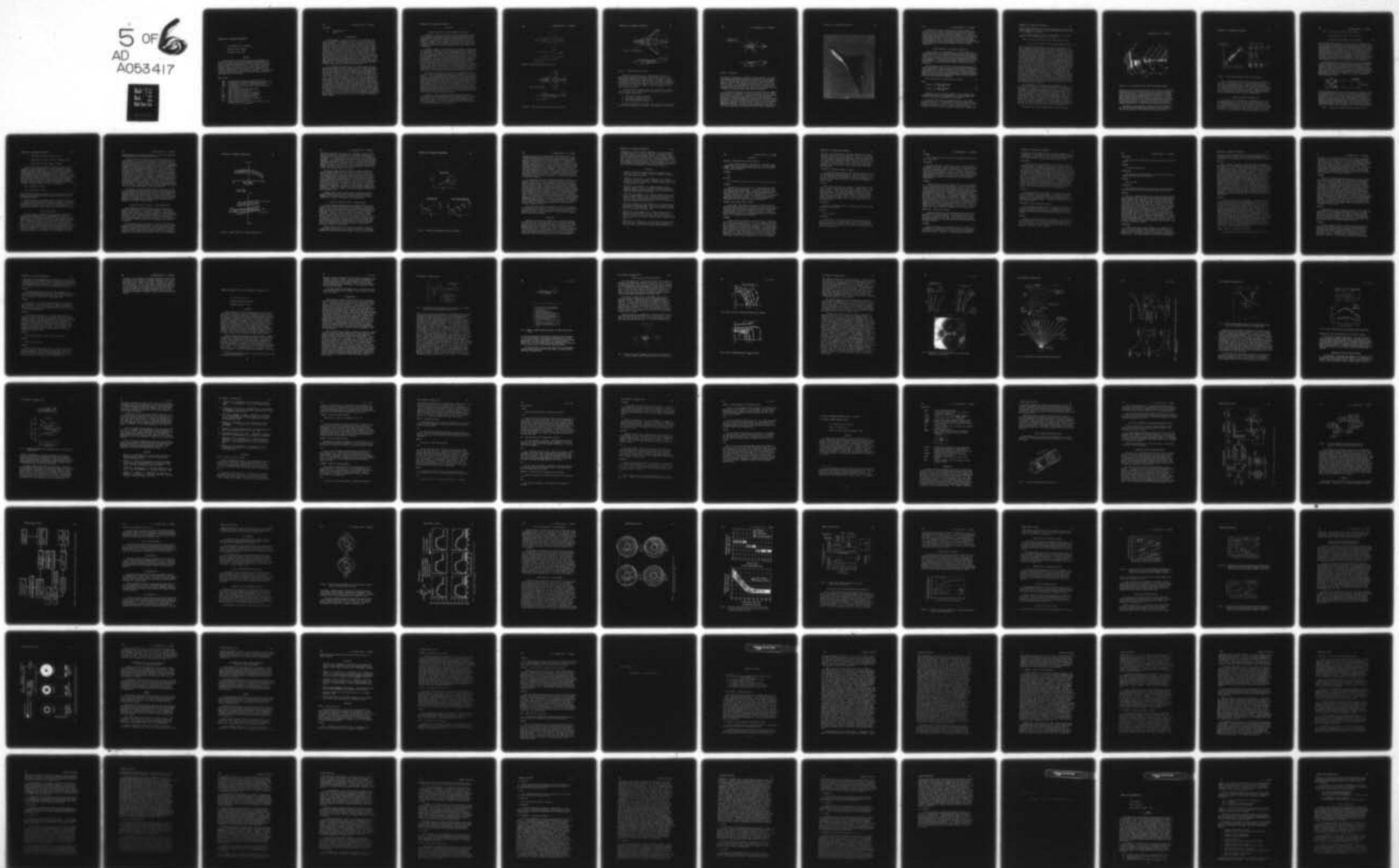

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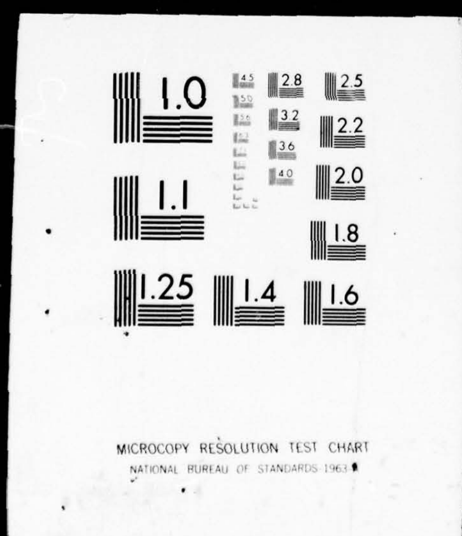

COMPATIBILITY TECHNOLOGY REQUIREMENTS

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General Electric Company

Cincinnati, Ohio 45215

ABSTRACT

The achievement of successful engine-airframe integration on short-haul aircraft imposes a broad spectrum of compatibility problems introduced by the propulsive system concepts presently being explored or envisioned for future use. In this paper, propulsion system concepts are reviewed for their potential engine stability problems. Based upon this review, some of the compatibility technology requirements for future systems are discussed. The implications for testing and analytical studies are addressed, and the apparent technology voids are identified.

Nomenclature

CDGC	Circumferential Distortion Generation Coefficient
CDGθ	Circumferential Distortion Extent Generation Coefficient
CDTC	Circumferential Distortion Transfer Coefficient
CDTθ	Circumferential Distortion Extent Transfer Coefficient
EX	Extent Function
KC	Circumferential Distortion Sensitivity
KR	Radial Distortion Sensitivity
PS	Static Pressure, Also Used as a Subscript
PT	Total Pressure, Also Used as a Subscript
RDGC	Radial Distortion Generation Coefficient
RDTC	Radial Distortion Transfer Coefficient
TT	Total Temperature, Also Used as a Subscript
ΔPRS	Loss of Surge Pressure Ratio
b	Superposition Function for Combining Radial and Circumferential Distortion Components

Subscripts

C Circumferential
R Radial

INTRODUCTION

The development of short-haul aircraft for the 1980-1990 time period brings with it a concomitant set of new engine-airframe integration problems. Some examples of these problems are the control complexity introduced by new cycle concepts, geared fans, and thrust vectoring nozzles; the proper determination and allocation of surge margin for internal engine factors such as engine transients, control tolerances, and deterioration to mention a few; and the proper determination and allocation of surge margin for external engine factors such as the various types of flow distortion (nonuniformity). This paper addresses only the problems associated with the flow distortions to which engines in a short-haul aircraft will be exposed, although the importance of the other aforementioned problems is recognized. In the past, the flow distortion problems, with their associated impact upon engine stability, were generically described by the phrase "inlet-engine compatibility." As will be shown in this paper, this phrase is too restrictive when applied to many short-haul aircraft.

It is important to recognize that inlet-engine compatibility has been achieved successfully for a number of CTOL aircraft inlet-engine configurations where the challenging compatibility problems have been posed by the maneuvering and/or maximum Mach number requirements for the aircraft. Examples of recent aircraft-engine systems where successful compatibility has been achieved are the B-1/F101 and the YF-17/YJ101. During these development programs, the stability characteristics of fans and low pressure compressors were explored experimentally for a wide variety of inlet total-pressure distortion patterns, and the stability characteristics of core compressors were explored experimentally for a variety of total-pressure and total-temperature distortion patterns corresponding to representative fan discharge conditions. Thus, it was learned what the important parameters were that must be included in correlations which describe the distortion-sensitivity, distortion-transfer, and internal-distortion-generation stability characteristics of compression components. It is this depth of experience upon which one can build for developing methods for correlating the more complicated interactions between types of distortion which can occur in short-haul aircraft propulsion systems.

DISCUSSION

Potential Short-Haul Aircraft Compatibility Problems

Perspective of potential short-haul aircraft inlet-engine compatibility problems is gained by examining proposed short-haul aircraft configurations. The potential sources of distortion which may cause compatibility problems are illustrated for some aircraft concepts in Figures 1 through 4. These concepts include a V/STOL A aircraft with a lift fan, a V/STOL B aircraft with lift fan plus lift cruise engines, a V/STOL B aircraft with a remotely augmented lift system (RALS), and a helicopter.

As shown in Figure 1, the V/STOL A aircraft concept is subjected to the most severe distortions during V/TOL operations when ground effects are significant. The engine in the rotating nacelle will be subjected to inlet total-pressure distortion resulting from large mass flow ratios and total-temperature distortion resulting from ingestion of recirculated engine exhaust. The lift fan located in the nose of the aircraft will be subjected to inlet total-pressure distortion, inlet total-temperature distortion due to ingestion of the exhaust from the engines located in the rotating nacelles, and exit static-pressure distortion resulting from the exit flow control devices used for thrust vectoring. Because of this latter type of distortion, it is clear that the term inlet-engine compatibility is too restrictive when discussing short-haul aircraft. Inlet static-pressure lift-fan distortion is of concern for short length inlets as this aircraft configuration transitions to forward flight.

The sources of distortion to which a V/STOL type B aircraft with a lift plus lift cruise propulsion system would be subjected are shown in Figure 2. The cruise engines may be subjected to inlet total-pressure distortion resulting from high mass-flow ratios and inlet total-temperature distortion resulting from ingestion of the cruise-engine exhaust during V/TOL operation. Further, the lift fans may be subjected to inlet total-pressure distortion resulting from high mass-flow ratios, inlet static-pressure distortion resulting from the short inlet and crosswind flows occurring during transition to forward flight, and inlet total-temperature distortion resulting from ingestion of the cruise-engine exhaust gas during vectored lift operation. The exit of the lift fans may be subjected to static-pressure distortion due to the thrust vectoring which will be required for control purposes.

A V/STOL B aircraft concept with a RALS (Figure 3) presents a particularly severe inlet total-temperature distortion problem resulting from ingestion of the exhaust gases from the remote augmentor and the vectored lift exhaust from the cruise engine during V/TOL operation. The cruise engine will also be subjected to inlet total-pressure distortion associated with normal inlet operation.

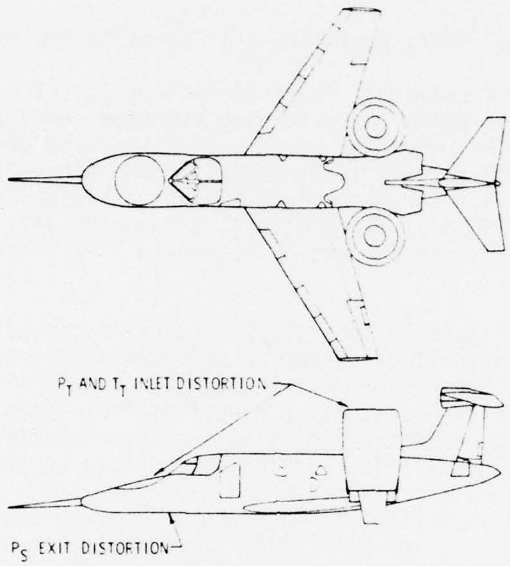


Figure 1. V/STOL A Aircraft with Lift Fan

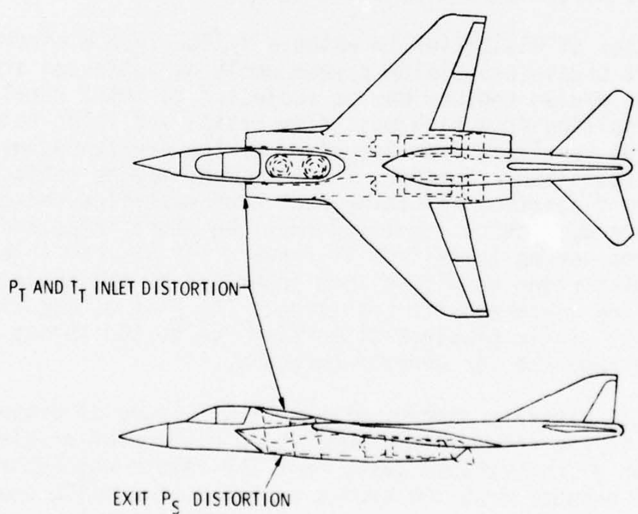


Figure 2. V/STOL B Aircraft with Lift Plus Lift Cruise

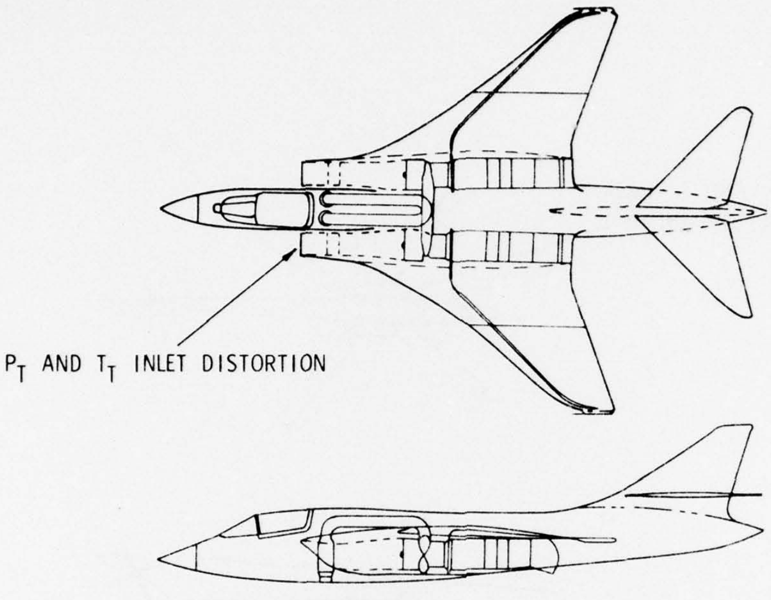


Figure 3. V/STOL B Aircraft with RALS

One type of short-haul aircraft, the helicopter (Figure 4) is beginning to experience inlet/engine compatibility problems as the turbine engine becomes more closely coupled to the rotor flow field. The external flow field swirl induced by the rotor leads to asymmetric engine-exhaust flows and can result in exhaust gases being ingested by the inlet and subjecting the compression system to inlet total-temperature distortion. In addition, there is some small degree of inlet total-pressure distortion.

This examination of potential short-haul configurations indicates that, in general, the external distortion problems can be classified into four common problem areas:

1. Inlet static-pressure distortion
2. Inlet total-pressure distortion
3. Inlet total-temperature distortion
4. Exit static-pressure distortion

During the remainder of this paper, our attention will be directed to the latter three types of distortion. While inlet static-pressure

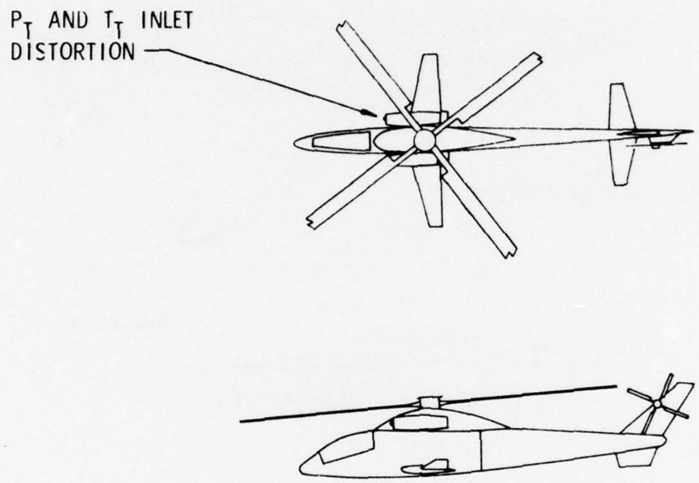


Figure 4. Helicopter

distortion is recognized as a potentially severe problem, it is hoped that adequate understanding and proper accounting for the combined effects of the other three distortions will permit ignoring the static-pressure distortion and, hence, will introduce only an insignificant amount of error. If this assumption does not prove to be valid, then it is anticipated that the methods for correlating the three types of external distortion discussed in this paper can be paralleled to provide a framework for handling inlet static-pressure distortion.

In the case of short-haul aircraft with multi-compression component engines, the engine developer is faced with additional internal distortion problems. In particular, consideration must be given to the transfer of inlet total-pressure distortion by an upstream compression component to a downstream compression component and to the total-temperature distortion that is created by the attenuation of the total-pressure distortion in the upstream component and is imposed upon the downstream compression component. A similar situation exists with inlet total-temperature distortion, the resulting total-temperature distortion transfer, and the associated total-pressure distortion which is generated.

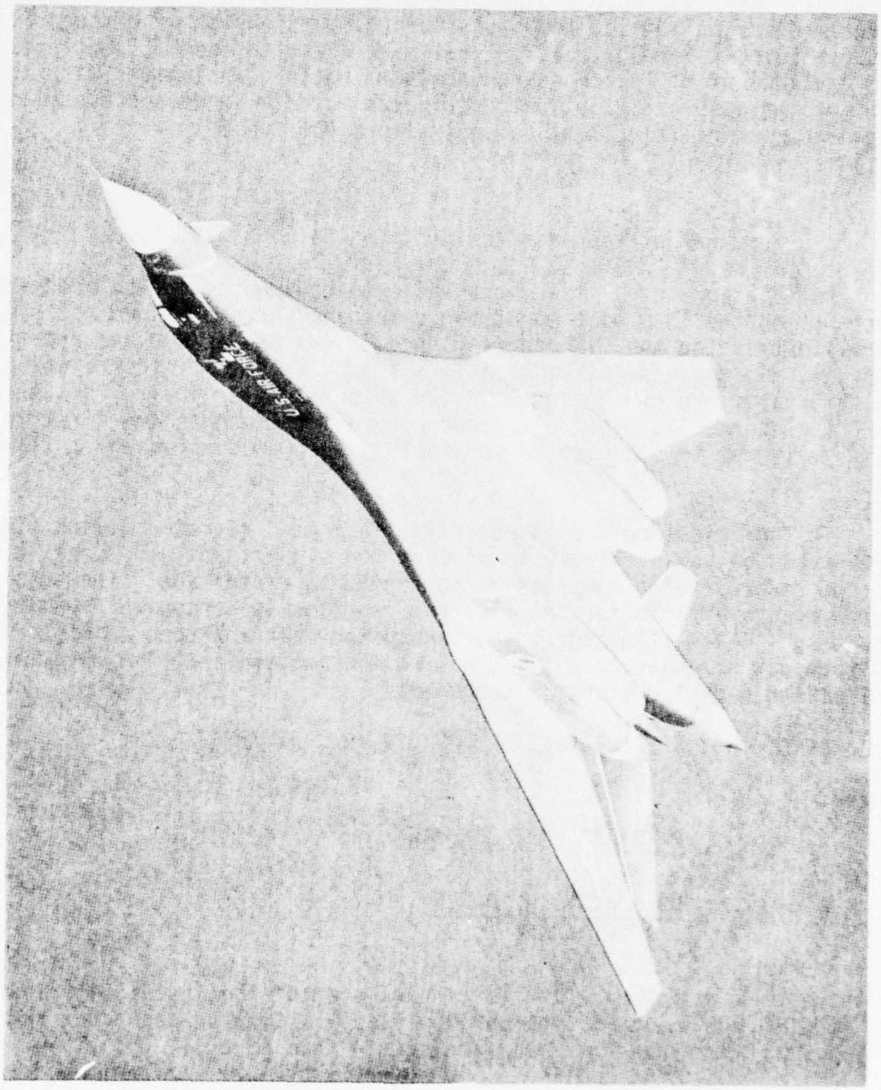


Figure 5. B-1 Aircraft

Although, this review of potential inlet/engine compatibility problems illustrates some of the complexities involved, a method for estimating the loss of engine compression component surge pressure ratio has been developed and successfully applied in a current engine program where only inlet total-pressure distortion was of concern. The program being referred to is the B-1 aircraft with the F101 engine. The development of loss in surge pressure ratio correlations with inlet total-pressure distortion contributed materially to successful inlet/engine compatibility flight demonstrations for the B-1 aircraft (Figure 5).

Current Approach to Inlet/Engine Compatibility

At this juncture, it is worthwhile to review the method used for correlating the loss of compression component surge pressure ratio due to distortion for the F101 engine, since it represents current practice. Further, all discussions of loss in surge pressure ratio correlations in this paper are consistent with the intent and terminology of SAE ARP 1420 (Reference 1). The referenced document represents the first industry attempt to standardize terminology when discussing inlet flow distortion.

The correlations are essentially linearized representations (in some cases piecewise linear to account for significant non-linearities) of the response of a compression component to distortion. The loss in surge pressure ratio for the F101 fan is given in conceptual form by Equation 1, which illustrates the assumption that any complex distortion pattern can be analyzed by decomposing the pattern into its circumferential and radial distortion components.

$$\Delta PRS = b_{PT}\ EX_{PT}(\theta)\ KC_{PT}\ \Delta PT/PT)_C + KR_{PT}\ \Delta PT/PT)_R \tag{1}$$

where:

$$\Delta PT/PT)_C = \frac{PT\ R\ AVG - PT\ MIN\ AVG}{PT\ F\ AVG} \tag{2}$$

$$\Delta PT/PT)_B = \frac{PT\ F\ AVG - PT\ R\ AVG}{PT\ F\ AVG} \tag{3}$$

The manner in which the level parameter of the individual rings is handled (selection of the maximum value) is engine dependent and is a detail that need not be addressed in this paper.

The factor $EX_{PT}\ KC_{PT} \Delta PT/PT)_C$ represents the loss in fan surge pressure ratio due to a circumferential distortion with level $\Delta PT/PT)_C$ and angular extent (θ^-). The factor $KR_{PT}\ \Delta PT/PT)_R$ represents the loss of fan surge pressure ratio due to a radial distortion of level $\Delta PT/PT)_R$. The loss in fan surge pressure ratio due to a complex total-

pressure distortion pattern is obtained by superposing the circumferential component of distortion using a superposition factor b_{PT} on the radial component of distortion.

In a similar manner, the loss in surge pressure ratio for the compressor can be obtained as shown by Equation 4.

$$\Delta PRS = b_{PT}\ EX_{PT}\ KC_{PT}\ CDTC_{PT}\ \Delta PT/PT)_C + KR_{PT}\ RDTC_{PT}\ \Delta PT/PT)_R$$

$$+ f[b_{TT}\ EX_{TT}\ KC_{TT}\ CDGC_{PT}\ \Delta PT/PT)_C + KR_{TT}\ RDGC_{PT}\ \Delta PT/PT)_R] \quad (4)$$

The first line of the right hand side of Equation 4 represents the loss in compressor surge pressure ratio due to a complex inlet total-pressure distortion pattern and is quite similar to Equation 1, except for the inclusion of two additional parameters. The total-pressure circumferential-distortion-transfer coefficient parameter $CDTC_{PT}$ transfers the circumferential component of inlet distortion to the plane of the compressor entrance and represents the effect of the fan (amplification or, more hopefully, attenuation) on the circumferential component of inlet total-pressure distortion. Similarly, $RDTC_{PT}$ represents the total-pressure radial-distortion-transfer coefficient and establishes the level of radial total-pressure distortion which enters the compressor. The second line of Equation 4 represents the loss of compressor surge margin due to the total-temperature distortion generated within the fan resulting from the work of the fan on the total-pressure distortion. The terms $CDGC_{PT}$ and $RDGC_{PT}$ are the circumferential and radial total-temperature distortion generation coefficients, respectively, and represent the generation of total-temperature due to inlet total-pressure distortion. The components of a complex total-temperature pattern are handled in the same manner as the components of total-pressure distortion. When both total-pressure and total-temperature distortions (combined distortion) are present, the function f accounts for superposing the two types of distortion and the angular displacement of one relative to the other.

The determination of the coefficients of Equations 1 and 4 necessary to correlate the compression component loss of surge pressure ratio within an accuracy of ± .02 ΔPRS units is not a trivial matter. Development of a distortion methodology, as implied by Equations 1 and 4, requires a substantial development program involving tests of the inlet, the engine, full scale inlet-engine testing, etc. and involves considerable communication and interplay between the airframe and engine manufacturers. The airframe-engine development program for the B-1 aircraft is shown in the time-line chart of Figure 6 and illustrates some of the efforts needed to establish a margin of compatibility. This margin of compatibility is attained when the engine will tolerate more than the objective level of distortion without surge, and the inlet will produce less distortion than the objective level.

During the steady-state component tests, the fan distortion

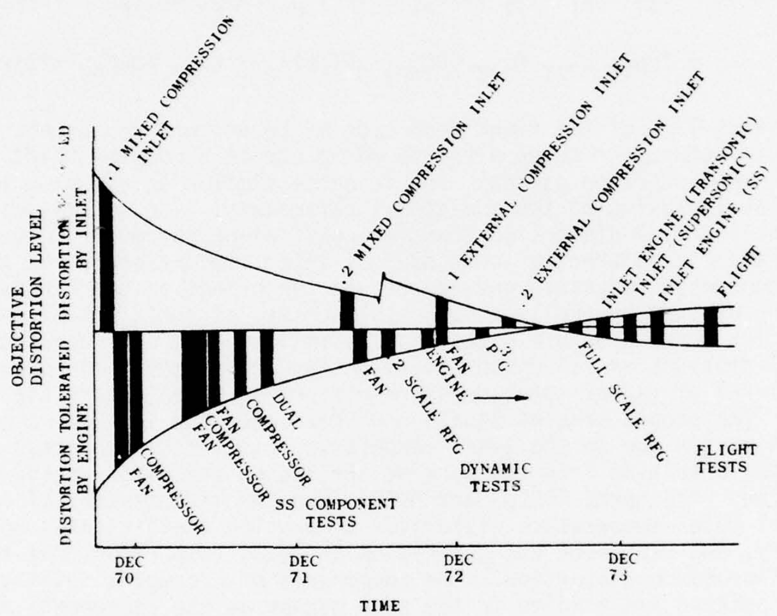


Figure 6. B-1 Aircraft/F101 Engine Program Development Summary

sensitivity, the fan distortion transfer, the fan distortion generation characteristics, and the compressor distortion sensitivity characteristics were determined. The equivalence between dynamic and steady-state distortion was determined, and the full scale engine was tested behind an RFG (Random Frequency Generator) (Reference 2). The RFG test provided validation of the equivalence between dynamic distortion and steady-state distortion and verification that stable engine operation was assured when it is subjected to distortion with dynamic content representative of conditions to be encountered during flight.

The types of screens that were tested during the development of the F101 fan distortion methodology are shown in Figure 7, as well as the resulting accuracy of the correlation provided by Equation 1.

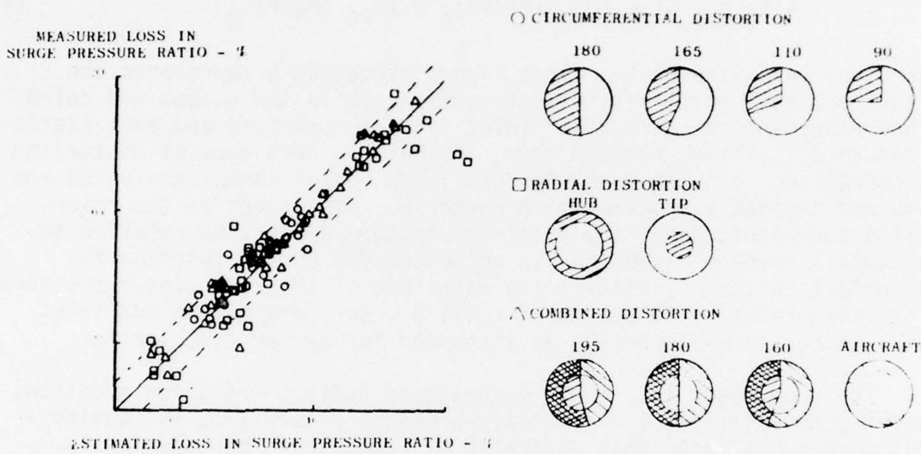


Figure 7. F101 Fan Surge Pressure Ratio Loss Correlation

Thus, a rational, consistent technique exists for correlating the loss in surge pressure ratio for current turbofan engines. However, there is no question that an extensive testing program is required to determine the necessary coefficients if the methodology is to produce reliable estimates. It is from this point of view that it is possible to look to future compatibility technology advances that will be required, based upon the problems identified during our review of potential short-haul aircraft configurations.

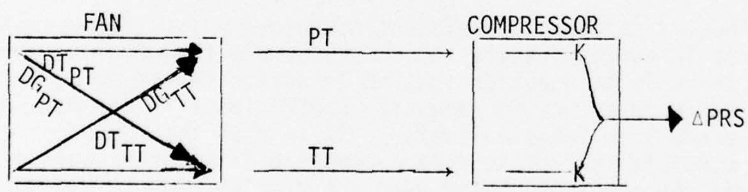
Proposed Approach to Compatibility

As one looks to the future when, in the general case, a fan is subjected simultaneously to inlet total-pressure distortion, inlet total-temperature distortion, and exit static-pressure distortion, it is anticipated that the loss in surge pressure ratio due to these distortions can be handled in a manner similar to and as an extension of Equation 1. This being the case, one might expect to correlate the loss in surge pressure of a fan using an equation of the form:

$$\begin{aligned}\Delta PRS = {} & b_{PT}\ EX_{PT}\ KC_{PT}\ \Delta PT/PT)_C + KR_{PT}\ \Delta PT/PT)_R \\ & + f_1[b_{TT}\ EX_{TT}\ KC_{TT}\ \Delta TT/TT)_C + KR_{TT}\ \Delta TT/TT)_R] \\ & + f_2[b_{PS}\ EX_{PS}\ KC_{PS}\ \Delta PS/PS)_C + KC_{PS}\ \Delta PS/PS)_R] \qquad (5)\end{aligned}$$

The right hand side of the first line of Equation 5 represents the effect of inlet total-pressure distortion, while the second and third lines represent the effects of inlet total-temperature and exit static-pressure distortion, respectively. As before, each type of distortion is decomposed into its circumferential and radial components which are combined through a superposition factor b. The effect of the orientation and magnitude of the total-temperature distortion relative to the total-pressure distortion is accounted for by the function f_1. The effect of the orientation and magnitude of the exit static-pressure distortion relative to the inlet total-pressure distortion and inlet total-temperature distortion is accounted for by the function f_2.

The compressor loss in surge pressure ratio correlation equation, assuming no significant exit static-pressure distortion, is considerably more complicated than indicated by Equation 4 when the fan is subjected to both inlet total-pressure and total-temperature distortion. The following sketch helps to illustrate the complexity of the problem and shows that there are potentially two sources of compressor total-pressure distortion and potentially two sources of compressor total-temperature distortion. When the fan inlet is subjected to both inlet total-pressure



and total-temperature distortion, the two sources of compressor inlet total-pressure distortion are the fan transferred total-pressure distortion DT_{PT} and the total-pressure distortion generated due to the fan working on the fan inlet total-temperature distortion (DG_{TT}). Similarly, the two sources of inlet total-temperature distortion are the fan transferred total-temperature distortion DT_{TT} and the total-temperature distortion generated due to the fan working on the fan inlet total-pressure distortion (DG_{PT}). Hence, the loss of compressor surge pressure ratio can be correlated with the following equation:

$$\Delta PRS = b_{PT}\ EX_{PT}\ KC_{PT}\ [CDTC_{PT}\ \Delta PT/PT)_C + CDGC_{TT}\ \Delta TT/TT)_C]$$

$$+ \ KR_{PT} \ [RDTC_{PT} \ \Delta PT/PT)_R + RDGC_{TT} \ \Delta TT/TT)_R]$$

$$+ \ f\{b_{TT} \ EX_{TT} \ KC_{TT} \ [CDTC_{TT} \ \Delta TT/TT)_C + CDGC_{PT} \ \Delta PT/PC)_C]$$

$$+ \ KR_{TT} \ [RDTC_{TT} \ \Delta TT/TT)_R + RDGC_{PT} \ \Delta PT/PT)_R]\} \quad (6)$$

The first two lines of Equation 6 represent the effect on the compressor of the circumferential and radial components of total-pressure distortion, respectively. The last two lines of Equation 6 represent the effect on the compressor of the circumferential-and radial components of total-temperature distortion, respectively. The terms within the braces are the components of inlet distortion, the fan distortion transfer coefficients, and the fan distortion generation coefficients and represent the distortion entering the compressor. The terms outside the braces are compressor coefficients completely analogous to those of Equation 4 except for the extent functions EX_{PT} and EX_{TT}. The extent functions take the following form:

$$EX_{PT} = EX_{PT}(CDT\theta_{PT}, \ CDG\theta_{TT}) \quad (7)$$

$$EX_{TT} = EX_{TT}(CDT\theta_{TT}, \ CDG\theta_{PT}) \quad (8)$$

where the CDTθ coefficients represent the extents of the transferred distortions and the CDGθ coefficients represent the extents of the generated distortions.

Even with only a cursory examination of Equations 5 through 8, it is obvious that an extensive amount of detailed testing of the compression components with distortion is needed to establish the necessary coefficients, functions, sensitivities, etc. Let us now turn our attention to some of the testing needs implied by Equations 5 through 8.

Future Testing Needs

Probably the major technology advance required in the area of compatibility technology will be the development of a method for measuring the static-pressure distribution of the fan exit flow and determining its effect upon the stability of the compression component. Such a determination requires not only evaluating the effect of the level of exit static-pressure distortion, but also evaluating the effect of the static-pressure distortion levels and orientation with respect to the inlet total-pressure and/or total-temperature distortion. The accurate measurement of stream-static pressures has always posed a difficult problem. In a distorted flow environment with its concomitant crossflows, it is necessary that static-pressure probes be

developed which are relatively insensitive to flow angle changes and/or can maintain close alignment with the flow.

Advances also will be required in designing efficient test programs which will provide the data necessary for determining not only the distortion sensitivities of the compression components, but also the superposition function for combining the radial and circumferential components of each of the three types of distortion, the superposition functions for combining the three types of distortion, and the distortion transfer and generation coefficients. These coefficients are nonlinear and are dependent on a number of parameters such as illustrated for the transfer of the circumferential and radial components of distortion for a recent fan (Figure 8). The total-pressure circumferential distortion transfer coefficient is dependent on the ratio of the level of circumferential component of total-pressure distortion to the radial component of total-pressure distortion, whether the circumferential distortion is located in the hub or the tip of the fan, corrected speed, and operating line. The radial total-pressure distortion transfer is not only affected by corrected speed and operating line, but also has a dependency on being located in either the fan hub or fan tip. This latter finding is not well understood.

The complexity of the correlation equations (Equations 5 and 6), and the extensive and complicated tests that will be required for obtaining the data necessary to develop the correlations, suggests that alternate methods for establishing the stability characteristics of compression components should be found. Because of the advances that have occurred in recent years in computer technology and in numerical methods, it is thought that analytical modeling of compression components may offer that alternate method.

Analytical Stability Studies - Current Capabilities

During recent years, sophisticated yet economical analytical techniques (References 3-7) have been developed for analyzing the stability characteristics of compression components. The success of these analytical techniques in simulating aspects of the aerodynamics important to stability characteristic predictions is taken as a portent of things to come. Namely, analytical predictions can be used so that the large amounts of testing can be reduced, and also predictions can be made for conditions which are not attainable due to facility limitations.

It is not the intent of this paper to review all the techniques or methods which have been developed for studying the stability characteristics of compression systems, but rather to merely mention and illustrate some of the results which have been obtained. A one-dimensional, pitch-line, dynamic digital blade row model of compression components has been constructed and used to predict with a high degree

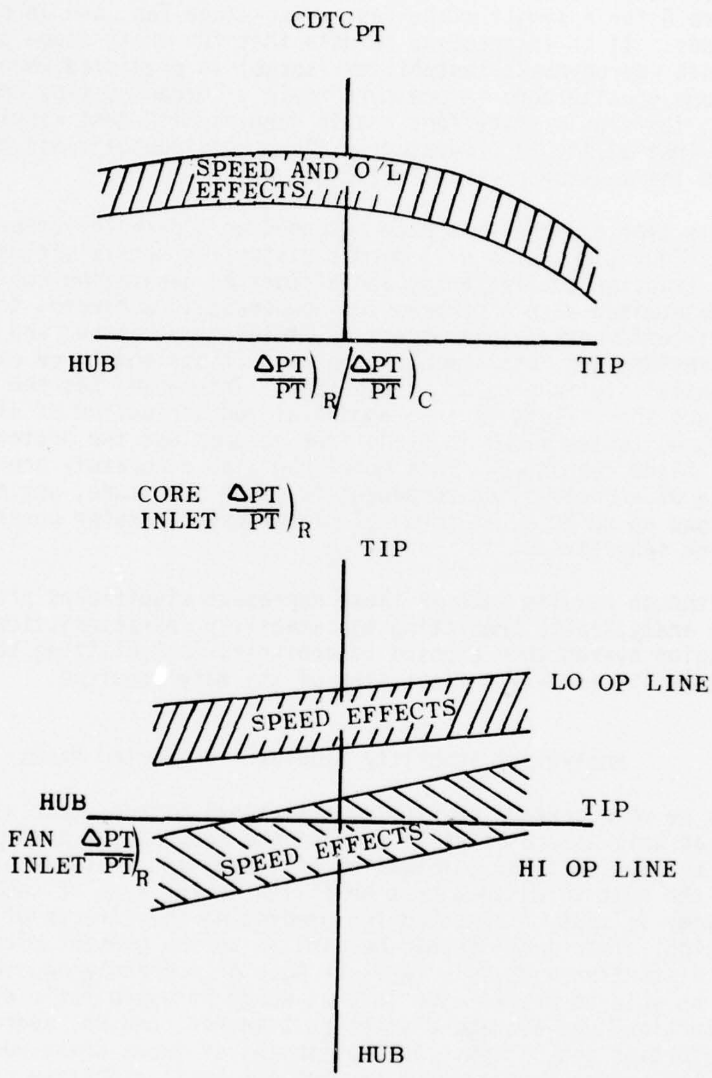


Figure 8. Example Distortion Transfer Characteristics

of success clean-inlet-flow surge lines, responses to 1/rev circumferential distortions, and responses to planar waves (References 3-6). Some results of clean-inlet-flow surge line studies are illustrated in Figure 9 for a single stage fan, a two-stage fan, and an eight stage compressor. It is interesting to note that for multi-stage compression components, aerodynamic instability (surge) is predicted when the speed line slope goes to zero in pressure ratio - corrected flow coordinates. However, for single stage fans and in keeping with test results, aerodynamic instability is predicted at flows considerably less than those at which the maximum pressure ratio occurs.

This type of model has been extended to a parallel-compressor type configuration which will permit distortion sensitivities, distortion transfer coefficients, and distortion generation coefficients to be determined when a compression component is subjected to 1/rev inlet circumferential total-pressure, total-temperature, and combined total-pressure and total-temperature distortions and 1/rev exit circumferential static-pressure distortion. This model has the capability to include the effects of the tangential redistribution of circumferentially distorted flows in blade free volumes and the unsteady aerodynamic blade responses. This model can also accurately predict the response of a compression component to large amplitude, unsteady planar wave flows up to 80 Hz in terms of planar-wave-transfer coefficients and surge sensitivity.

Although results such as these represent significant progress towards analytically predicting the stability characteristics of a compression system when exposed to arbitrary destabilizing boundary conditions, it is only a first step of the many required.

Analytical Stability Studies - Projected Needs

As we move towards developing analytical methods that will accurately as well as economically predict the response of a compression system to any arbitrary combination of steady and unsteady distortions, one of the next voids that must be filled in the area of compatibility technology is that of a method for predicting the effects of radial distortion. This model should be similar to the present circumferential distortion models in terms of ease of use and economics and should be able to predict the loss of surge pressure ratio due to radial distortion, the average distortion transfer, and the average internal distortion generation. Accomplishment of these goals would then permit the engine developer to predict the basic stability characteristics of a compression system when either pure circumferential or pure radial distortions were imposed.

Another technology void which is becoming increasingly apparent is what these writers refer to as "design for stability." The design guidelines for obtaining performance appear to be well established and

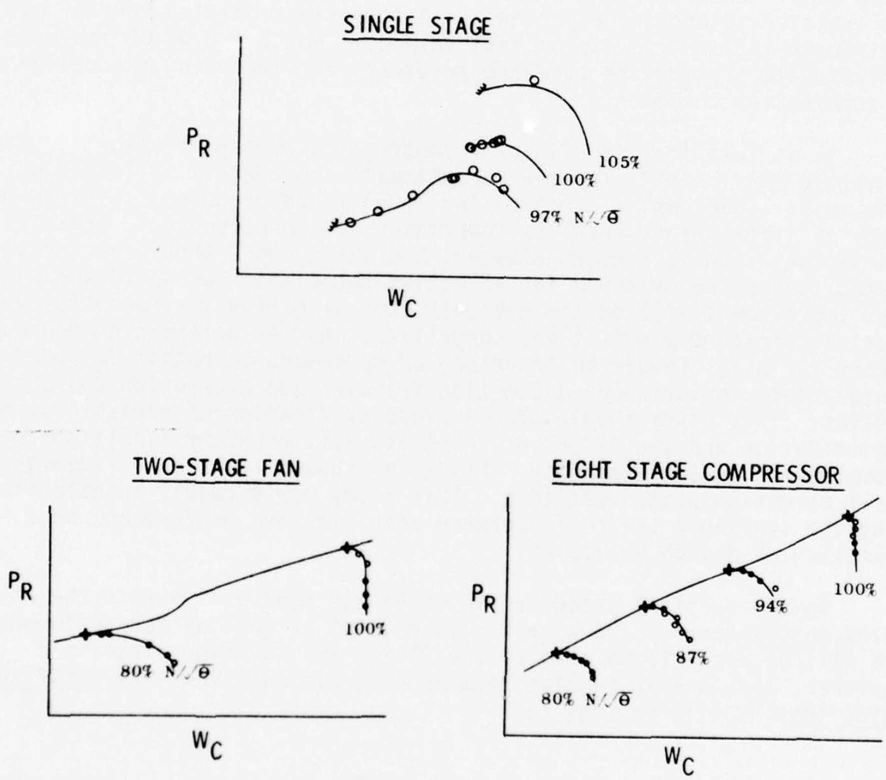


Figure 9. Compression Component Clean-Inlet-Flow Maps

can be carried out assuming steady flow. But because surge is an inherently unsteady event arising from the amplification of internal or external disturbances, the coupling between stages becomes important to predicting the stability characteristics of a compression component. Now that we are arriving at the point at which we can predict the conditions which will cause a compression component to surge, stability analyses personnel are being asked questions relating to how to design for stability in new components or what can be done to improve the stability of existing components. Unfortunately, at this time many recommendations are hit or miss guesses and are at best based on isolated islands of experience. A consistent unified body of parametric analytical results or test data does not exist which could be utilized to provide the guidance necessary for improving the design of a compression component.

It is felt that the field of compression component stability has advanced sufficiently far that a systematic parametric variation of design variables using a pitch-line model would produce a significant body of information useful to compressor design personnel. Geometry variations such as camber, stagger, and blade chord should be investigated to provide criteria for selecting the blade rows where the payoff for improvement will be the greatest, for selecting the variables which most significantly effect the surge line, and for determining how much these variables should be increased or decreased to obtain improved surge characteristics or distortion transfer and generation characteristics. This study should also include examination of variable geometry optimization and the effect of inlet and exit boundary conditions upon compression system stability. Attention should focus on off-design as well as design point operation. This study would result in establishing the available tradeoffs between stability and performance as a function of design variables.

Once a verified procedure is developed that will permit the engine manufacturer to "design for stability" as well as for performance, it will be possible to move to the flight verification phase in a quicker, less costly fashion because only minimal stability verification testing will be required.

CONCLUSIONS

Short-haul aircraft for the 1990s present the engine developer with some important challenging compatibility technology requirements. In particular, inlet total-temperature distortion and fan exit static-pressure distortion will present significant problems which must be addressed both by test and analytical methods. Although a format for correlating the loss of surge pressure ratio for compression components can be envisioned as a logical extension of a current correlation system, the complications introduced by including all known destabilizing distortion effects implies the need for even more extensive

testing than is currently being accomplished. It may be possible to circumvent some of this testing by resorting to analytical methods for predicting compression system stability. These methods, some of which are currently being developed, will present the engine designer with the ability to make fast, economical stability predictions that will, in part, obviate the need for some of the implied testing. Even more important significant economies in time and funds will be achieved as techniques are developed which will permit designing for stability with a concomitant recognition of the performance tradeoffs.

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DISCUSSION

WELLIVER: (Boeing Military Airplane Division)

I have just one question on what you said. You made a side comment there , really, that you input deviation into your analysis program. Does your analysis program take into account the details of the blade row geometry?

STEENKEN:

Yes.

WELLIVER:

If it does then I am lost as to why you use the deviation input.

STEENKEN:

The model that I was referring to is a one-dimensional pitch-line blade row by blade row model. It uses flow areas, the pitch-line geometry (radii, metal angles, and deviation angles), and rotor loss coefficients as input. The loss coefficients and deviation angles are correlated as a function of incidence angle. It is usually a straight forward task to obtain loss coefficients and deviation angles from clean-inlet-flow data if you make the appropriate measurements for each blade row within the compressor.

DENNING: (Rolls-Royce (1971) Limited)

It has been a long time since I worked in this field and computers have come in since then. The question I would like to ask you is, when you are dealing with distortion do you think we could continue along the lines of using regression analysis in conjunction with these tests on sectors, or should we get down to a better analytical model and really try to understand what goes on in the compressor of a fan when you have distortion?

STEENKEN:

I think the point is that the correlation method is something that allows you to communicate, and it is a very necessary tool right now. As you gathered from my talk, I am worried about its utility for the future if it gets much more complicated. I sincerely believe that if we learn or had learned how to develop and use proper models, then we would be much closer to the real answer. With the development of some of the microprocessors, it is not hard to believe that you could have a simple on-line model for evaluating the effects of distortion. In a sense we are almost there. As you may know, we are using analog computers during compatibility testing to accomplish

real-time, on-line analysis for evaluating the effects of dynamic distortion using the correlation equations. Hence, it is not hard to believe that by the time the 1990s roll around with improved analytical techniques and with the improvements that are rapidly coming in microprocessor technology, we might be able to make use of on-line models for evaluating the effect of distortion without having to resort to such correlations. Then we would have a more useful communication tool that the airframer and the engine manufacturer could work with.

SILVERSTEIN: (National Academy of Sciences)

I think in pointing out the problems you have with correlations, the same statement can be made about all types of correlations. That is, you cannot get anything more out of them than you put in. All that they are good for is interpolation. And I was wondering, have you ever tried to use results from correlations on one engine on another engine?

STEENKEN:

Well, that is not quite the situation. The correlation I have shown is the one that GE has been using for a number of years in one form or another. We have had a great deal of success with it, but let me point out that we are talking about measuring 40 probes, 5 rings with 8 rakes. We find that the manner in which you average the hub and tip or how big an angular sector you average over does change from engine to engine. This is a result of the engine response, not because the equation in its basic general form is any less important.

SILVERSTEIN:

Oh no, I do not question that. I simply say the derivatives are different for each engine.

STEENKEN:

That is correct.

SILVERSTEIN:

And the thing is of no use to you unless you know the derivatives. It is true of all correlations. You take the data you get experimentally and put them in an ordered form. You cannot get anymore out of that kind of analysis than another point between the two points you have measured.

STEENKEN:

That is exactly correct, and you have to measure each time you have a new engine.

SILVERSTEIN:

But now really what I am getting to, is your use of the words "designing for stability instead of performance." Actually are not you really saying that you have to design over the whole map? Is not that what we always have done? The fact is there is no way of designing just for performance? What you are saying is that you must get an intimate knowledge of the details of the flow within the multistage compressor at every point, radially and axially, in order to understand the conditions for which the flow will break down and surge results.

STEENKEN:

I hope I did not mislead by saying design only for performance, but to a large extent that is what happens. Then stability considerations get tacked on and you work with what you have. This is a situation where I think the stability people ought to become involved in the design loop much earlier and begin to work with the compressor designer, because usually he is charged with meeting design point performance as the main goal. The off-design performance often occurs much later in the program, and as an example you somehow find a way to get around some of the subidle region problems. However, I certainly don't disagree with your point, but what I am talking about is learning about the details of the flow at all points on the map, so you can better match the stages or match the blade rows. But the point of whether you need to know everything about the blade row all the time, radially as well as circumferentially, is not clear to me. I think it depends upon a given compression component and its sensitivities.

NAPOLITANO: (University of Naples)

I agreed with the remarks that everybody has been making, that correlation has drawbacks. However, the limitation is cost. But I would like to ask you one question. Given the number of data you had and given everything else, what is the reason for not trying a nonlinear correlation in the range where you have shown that things are really holding?

STEENKEN:

There is no reason for not trying a nonlinear correlation. It is one mainly of convenience and of ease of usage, and I think that as time goes on, there certainly will develop ways of accomplishing what you are saying. But that is a step beyond where we started.

We hoped we could do things with linear functions because that was a simple way. My contemporaries would probably be appalled to hear me say that I am willing to move away from the linear type functions, but I personally don't see any reason why I cannot do that.

NAPOLITANO:

I would like also to underline and stress, and this goes into the general overall strategy in which you see the future, there is a little bit more of interphasing between pure statistics like this is and physical insight and computing power. I mean, as Mr. Denning was saying, were computers not around, you had some constraints, but now that they are around, why don't you put the little slaves to work? I would also like to stress that, maybe you did say this in discussion, but there is a direction which I think is the most promising; it is the interacting of the two, the halfway, the hybrid between the two where the shortcomings of one are eliminated or overcome by the other, so you take a short cut and get to the point. You mentioned this a little bit in speaking of microprocesses, but the point is to see clearly what type of tests you have to do to utilize both as the function correlation, not the point correlation and the model correlation. That is the direction, I believe, you should go.

STEENKEN:

I think your point is well taken. Even if you don't use the correlation method in the future because it becomes so complicated, you will certainly be evaluating some of the derivatives in tests and tracking as to how they match with previous experience and whether you are making progress. You cannot forget the past, and you should build upon it.

MIKOLAJCZAK: (Pratt & Whitney Aircraft)

I was impressed with your ability to predict the stability of a single stage fan with uniform flow. I am wondering if in that prediction you were using the overall characteristic of the fan or whether you were doing some three-dimensional adding-up from root to tip?

STEENKEN:

We were doing no three-dimensional adding-up. It was strictly the one-dimensional model. We took the single stage map, derived the loss coefficients, the deviation angles for the rotor, and the loss coefficient for the stator and put it into the model with the geometry. The result is what I showed.

MIKOLAJCZAK:

When you say loss coefficient, you mean an average for the whole blade?

STEENKEN:

For the whole blade, yes.

MIKOLAJCZAK:

Have you used the measured shape of the characteristic along a constant speedline to do the prediction?

STEENKEN:

That is correct.

MIKOLAJCZAK:

This therefore implies that you will have to build the machine before you can make any kind of prediction about its stability.

STEENKEN:

That is an interesting point, and I would not disagree with what you are saying. The point I want to make about what we have done is that we predict where instabilities occur and that one-dimensional models can be of help. Now when you get to compressors which have a low hub-to-tip radius ratio, then what I think you are driving at is the fact that the hub characteristics are going to be a lot different than the tip characteristics. There is no reason, if you make measurements behind the appropriate hub, tip, and midspan regions, that you cannot play the same game by calculating the stability characteristics and in fact find out whether it is the hub- or the tip- or midspan region that is weaker. I think this becomes much more important when you talk about a multistage machine, since obviously we get into a lot of problems of determining how the loading distributions change through the machine. But the point is then that you can do something about it, once you figure out which stage it is that is the weakest from your time dependent model analyses. Does that tend to answer you question?

MIKOLAJCZAK:

I would like to make a general comment. There appears to be a desire to make more extensive use of computational capabilities to arrive at a more fundamental prediction of stability. A necessary prerequisite is to be able to predict the off-design performance from choke to stall, since subtle changes in the shape of the compressor

characteristic near stall influence strongly the prediction of the instability point. We shall have to model the off-design flow in the blade row very accurately before we can make significant progress.

WELLIVER:

When I worked on compressors, though they did have computers, they were not as big as they are now, so that shows I am younger than Mr. Denning. But one of the things that bothers me in this discussion and really has for years is that again, and I agree with a point made over here, that the aggressive analysis is glorious if you are going to interpolate, but you don't extrapolate with it. It seems to me that all these analyses are going to duck the issue until you input the actual hardware in there and you analyze the actual physical metal and you analyze through the blade rows. If you were to do a detailed analysis of some of the airfoil sections that are actually used in engines flying today, it would make your hair stand on end compared to what you might have an airfoil shape look like if you knew what the shape should be. You will never get there until you actually lay out the hardware and analyze through the passages, and that, by the way, I think gets you to the off design automatically. But somebody has to start out to do that. Here we are one more time with a one-dimensional pitchline regressive analysis, and I am not knocking it. I know that you need answers now, but it seems to me that, somewhere along the line, somebody ought to be working that problem pretty hard.

STEENKEN:

I think that that area is being worked. I did not address the problems that are associated with that complex area. But it leads to exactly the point that you were making earlier, that you can get one man who can only run that program and it is not user oriented. The point is this complication stands in the way of allowing us (stability design personnel) to work on a day to day basis with the compressor designer and to help improve things. I don't, by any means, imply one-dimensional models are the total answer, but I do think that there is a lot that can still be learned from one-dimensional models which has not yet been wrung out of them. I think that some of the results that we are obtaining bear that out. But I agree with you, the trend has got to be towards a more complete analysis. At the moment, I have not seen results that show we can do that on an economical basis for all machines. The other point, Burt, that I think you were driving at about analyzing the blade foil shapes is a good one. There is room for some improvement, but it is an area that I think is just beginning to emerge.

BRUNDA: (Naval Air Propulsion Test Center)

I would like to mention that the present state of the art in

distortion work is such that, even for a given existing compressor or engine, it is not possible to satisfactorily measure the distortion tolerance of the compressor, so that the one-dimensional approach that you have taken, I feel, is very important and continued work in that area is needed, even though some of the refinements that we would like to see in them are not present at this time. I would also like to ask a question though about your analysis. In your study of the effects of stagger angle and aspect ratio on the distortion tolerance. do you foresee the possibility of determining a limiting design parameter for a particular compressor?

STEENKEN:

That is an interesting question because, at this time, I believe we have identified the variables that are involved, but whether or not a limiting design parameter can be derived is not completely clear to me. Based upon the single stage fan study that I briefly discussed, it is probably possible to solve analytically the matrix of equations and determine what the important variables are and how they relate to the surge line. A number of similar studies have been done in the past using unsubstantiated simplifications and it is not clear to me that they had a validated dynamic model to start with as we would have. But again, as soon as you move to a multi-stage machine, the coupling and interaction effects between the blade rows probably would preclude determining a detailed stability parameter, because it would involve so many blade rows and the associated flow variables. I do believe that if you could do it for a single stage fan that it would tend to lead you in the right direction because we know right now that it is not simple slope or level effects. It is a combination of many things.

GOETHERT:

I understand that your calculations are done for both steady state distortions as well as dynamic oscillating distortions. I wonder, did you use also for the dynamic distortion the steady state characteristics, or did you consider the changes of the characteristics due to the nonsteady conditions? The ratio which is important here is the distortion wave length to the blade chord and, for high frequency distortions, this difference becomes important.

STEENKEN:

For the two-stage fan effort on which we reported, we used the steady-state correlations; that is, that the loss coefficient and deviation angle change and adjust instantaneously to new values associated with the new instantaneous incidence angle. That correlation works well for frequencies up to about 60% of the one per rev speed of the fan. At that point, we looked at the Schor and Reddy work. Based on it, we developed a technique for handling the

finite time for flow readjustment, and showed that if it was included in the model, we obtained better phase correlation between parameters. I believe, based on our results, that it is necessary, once the planar wave frequency reaches 60% of the fan speed, that you do have to account for the finite time for the fluid to readjust around the blade.

GOETHERT:

You mentioned earlier the work of Schorr and Reddy at the University of Tennessee Space Institute. The difficulty with that is that it was done for incompressible flow, and I think we need very badly an extension of this work for compressible flow.

STEENKEN:

That is right. There is no question that the time constants we used pushed us in the right direction. However, the magnitudes were not correct. This is an area we would like to pursue and to fine tune the time constants and determine the values for real blades in a multi-stage compressor.

GOETHERT:

I have another question. Let us consider again the dynamic input, for instance, the dynamic pressure oscillation with a certain amplitude. Do you find with your method that this pressure amplitude increases when you go through the compressor due to the fact that you have a volume of space between the individual stages which is compressed and expanded? You see, when you have a certain pressure distortion, Δp, of the inlet flow with a certain frequency and now you go through the individual stages then the pressure disturbance will grow due to the fact that you have a volume of air between the individual stages.

STEENKEN:

Are you talking about a steady state distortion now?

GOETHERT:

No, an oscillatory one.

STEENKEN:

Okay, the planar wave type of thing. To the best of our knowledge, it is handled rather rigorously in that we use the three conservation equations and the proper thermodynamics. We do not currently include the interaxial gaps between blade rows. They can be included, but it is expensive since the computing time is

increased. This is because you are dealing with small volumes that are characteristic of many of our modern compressors; that is, the gap length to radius ratio is very small. I believe it is a small effect, but there is nothing in our results right now that shows it to be significant. If it is, then there is nothing that would keep us from calculating and including that effect because we have a redistribution model for free volumes that we can use. We are currently using this model for blade free volumes upstream or downstream of the compression component, but not in the interaxial blade-row gaps.

PROPULSION INDUCED EFFECTS FOR VTOL AIRCRAFT IN GROUND EFFECT

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ABSTRACT

Several important effects occur when the jets from a VTOL aircraft strike the ground. Turbulent entrainment in the jet plumes and the wall jets spreading from their impingement leads to negative pressures on the vehicle's lower surfaces and a downward force known as "suckdown." Interaction of more than one jet with the ground results in reflections of exhaust gases upward toward the vehicle, producing "fountains" and "upwashes." The interaction of these upward flows of exhaust gas with the aircraft can produce favorable forces, but induced moments are often destabilizing, and dynamic instabilities can also occur. The hot exhaust gases reflected towards the airplane also cause skin heating and engine reingestion problems. The ability to predict these effects is necessary for efficient design of VTOL vehicles. This paper summarizes the state of the art in understanding and modeling these complex problems.

Grumman's prediction techniques are organized around a computer program which is comprised of subroutines utilizing various methods of calculating different aspects of these flows. This program is written so that portions of it can be continuously updated as the techniques are improved. Calculation of the flow outside of the outer jet boundaries is founded on a finite element panel method for inviscid flows with the jet plumes and wall jets represented by additional panels. The flow between the jets (inner region), however, is not as amenable to analysis because of its complexity. This region

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has been divided into segments which are modeled separately and fit together in a modular fashion. We have modeled the components of the inner region by analytical methods wherever possible and verified these results by experiment. Where analytical methods proved inadequate, we used correlations of the experimental data to develop empirical models.

This paper summarizes the methods used for modeling the inner region flows and discusses the aspects of these flows most in need of further research.

INTRODUCTION

Existing VTOL aircraft are severely restricted in performance, sacrificing either payload or range in order to achieve VTO capability. The engines are usually sized by the requirement of thrust equal to weight for vertical takeoff. The performance loss is traceable to the additional weight associated with the required thrust, thrust vectoring hardware for the primary propulsion system and separate lift engines, and to aerodynamic problems that are peculiar to VTOL aircraft design. These special aerodynamic problems are primarily caused by jet-induced effects that are basically different when the aircraft is close to or far from the ground (Ref. 1). Interference effects occurring out of ground effect involve lift losses that are associated with the entrainment of ambient air by downward-directed jets and the free stream flow field distortions produced by the exhausts as the aircraft begins transition to horizontal flight. In-ground-effects include stronger lift losses incurred by enhanced entrainment close to the ground (suckdown), engine thrust loss caused by inlet ingestion of hot exhaust gases, and the positive lift forces caused by fountain impingement on the aircraft underside.

VTOL aircraft are more strongly influenced by propulsion induced effects than are CTOL designs because of the strong interaction between the exhaust flow and the VTOL vehicle's environment. An example of the varied nature of these interference effects on vertical force is given in Fig. 1 (from Ref. 2). Different aircraft designs can produce force interference characteristics which follow very different patterns. Since the aircraft's propulsion system is generally sized by the requirement that it lift a fully loaded vehicle off the deck, a few percent loss of thrust because of these interference forces results in a much larger change in aircraft size. Prediction of VTOL aerodynamic performance characteristics requires an analysis that can account for many different flow field processes that are driven by turbulent mixing and entrainment of ambient air. Further discussions of the many problem areas, their importance to aircraft design, and methods for dealing with them is given in Refs. 3 to 8.

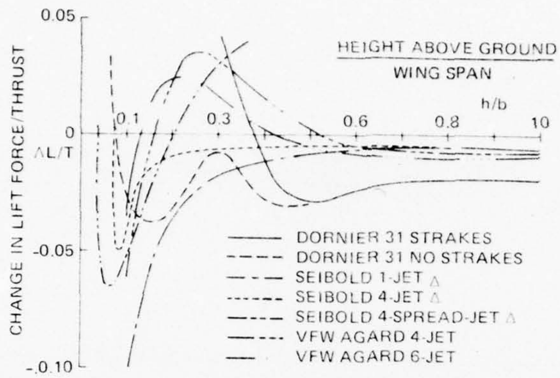
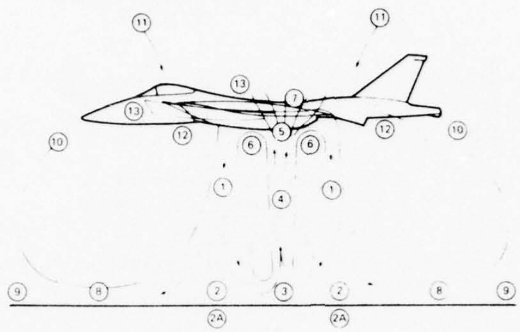


Fig. 1. Illustration of the Variation of Jet-Induced Lift in Ground Effect With Vehicle Geometry (From Ref. 2)

Refs. 6 and 7 describe the development of techniques used by Grumman to predict the interaction effects that occur for V/STOL aircraft. Calculation of the flow outside of the outer jet boundaries is founded on a finite element panel method for inviscid flows, with the jet plumes and wall jets represented by additional sets of panels The velocity has a component normal to these jet boundary panels, to account for the turbulent entrainment. This method works very well both in hover and with forward velocity. The flow between the jets (inner region), however, is not as amenable to analysis because of its complexity. A review of past investigations of such flow fields can be found in Refs. 3 and 5. Each of the impinging jets produces on the ground a flow that turns parallel to the ground to form a wall jet. Collision of two wall jets forms a relatively thin, fan-shaped upwash. This upwash formation process is very important, since it is the first major step in development (and modeling) of the inner region flows. Factors influencing the upwash between the impinging jets include the size and shape of the nozzles, the spacing between them, relative jet strength, ground plane distance, and angle of jet impingement. A concentrated, jet-like fountain flow can exist when three (or more) upwash flows intersect. The location, direction, and intensity of the fountain depend on the properties of the upwash flows that form between pairs of adjacent jets. In our treatment, this region has been divided into segments which are modeled separately and fit together in a modular fashion to compute the entire flow (Fig. 2). Many of the modules used to treat fluid processes such as ground impingement, fountain formation, and inlet ingestion are not amenable



1 JET MIXING AND MUTUAL INTERFERENCE
2 JET GROUND IMPINGEMENT & SPREADING
2A DECK HEATING
3 FOUNTAIN FORMATION
4 FOUNTAIN FLOW AND MIXING
5 FOUNTAIN IMPINGEMENT ON AIRCRAFT
6 SPREADING FOUNTAIN EFFECT ON BASIC JETS
7 FOUNTAIN FLOW AROUND AIRCRAFT
8 SPREADING GROUND JET AFTER IMPINGEMENT
9 OUTER SEPARATION OF SPREADING JET
10 RETURN FLOW OF EXHAUST
11 INFLOW DUE TO ENTRAINMENT AND RECIRCULATION
12 PRESSURES AND FORCES ON LOWER SURFACES OF AIRCRAFT DUE TO INFLOW
13 FOUNTAIN EXHAUST PRODUCTS SUCKED INTO INLETS

Fig. 2. Modular Elements Modeling Concept for Under-Aircraft Flow Fields

to direct calculation by conventional aerodynamic analysis because of the strong influence of turbulence and entrainment. The best current models of these fluid processes, which rely heavily on expermental data, have been incorporated in the computer program in modular form. This facilitates continuous improvement of predictions as more accurate data or new techniques become available.

In this paper we will discuss the state of the art and aspects needing further research for the inner region of a V/STOL vehicle operating in ground effect.

STATE OF THE ART FOR THE INNER REGION

Our basic modeling of in-ground-effect flows is built upon the modular, or building block, concept illustrated in Fig. 2. The flow is computed beginning from the nozzle and following the flow through the jet plume mixing, ground impingement, wall jet, upwash formation, etc. This method allows different modeling methods for each of the regions and a continuous updating and inclusion of added complexities. The results of this modeling determine the direct ground-effects on the vehicle and provide the boundary conditions for the inviscid panel method computations of the outer flow.

While there is always room for improvement in any technical area, we feel that free jet plume mixing is adequately modeled. (cf. Refs. 9 and 10). Modeling of the jet impingement and wall jet formation is based largely on the work of Donaldson and Snedeker (Ref. 11). A summary of their results for subsonic, circular jets indicates that: the extent of the impingement region is approximately twice the impinging jet diameter; the wall jet half velocity thickness grows such that $h_{\frac{1}{2}}/r = .07$; and the maximum velocity in the wall jet decays with distance from the center of the impingement raised to the -1.1 power.

We have extended this knowledge to include the effects of both noncircular jets and fan jet type plumes on the wall jets. The wall jets from a fan jet type flow (i.e., a core region having lower dynamic pressure than the outer ring or fan flow) were found to be

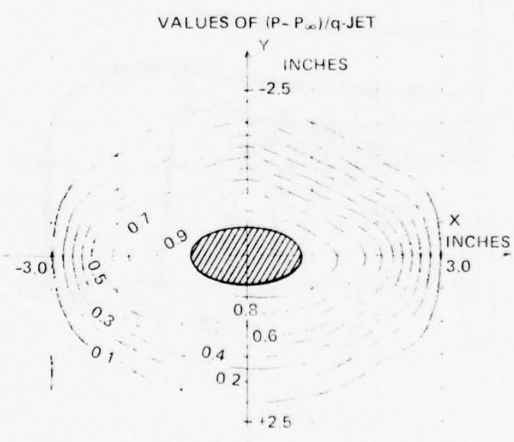


Fig. 3. Ground Plane Static Isopressure Contours For Aspect Ratio = 2 Elliptic Nozzle (Nozzle Dimensions Shown by Crosshatching)

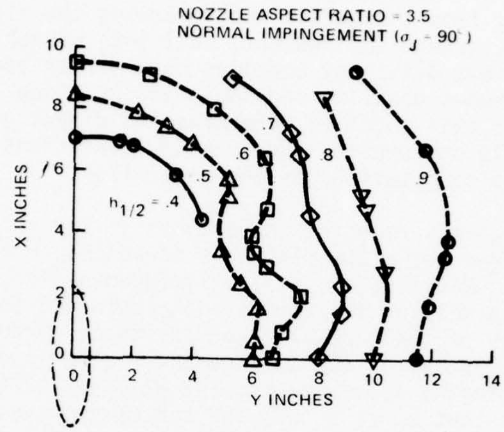


Fig. 4. Wall Jet Half--Velocity Thickness ($h_{1/2}$) Isolines

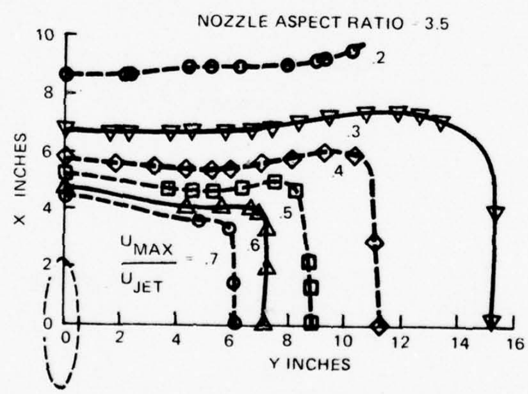
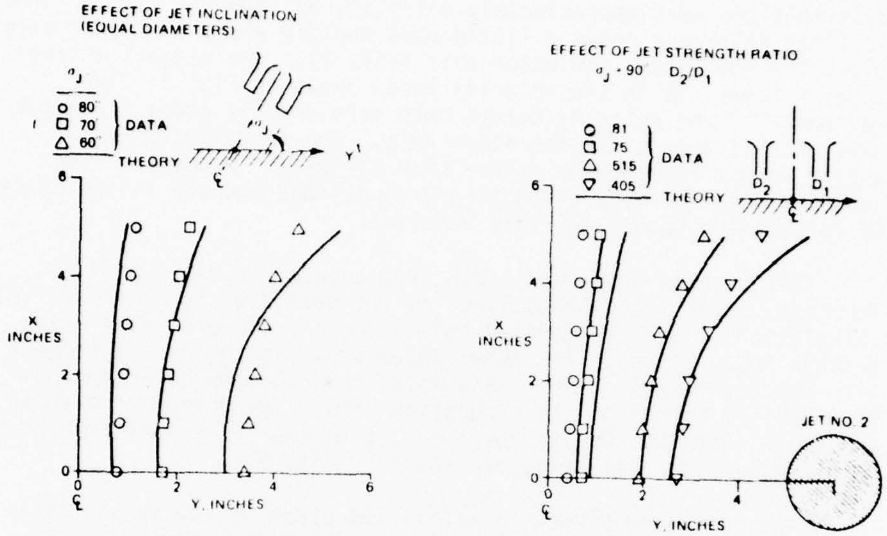


Fig. 5. Wall Jet Maximum Velocity (U_{MAX}) Isolines

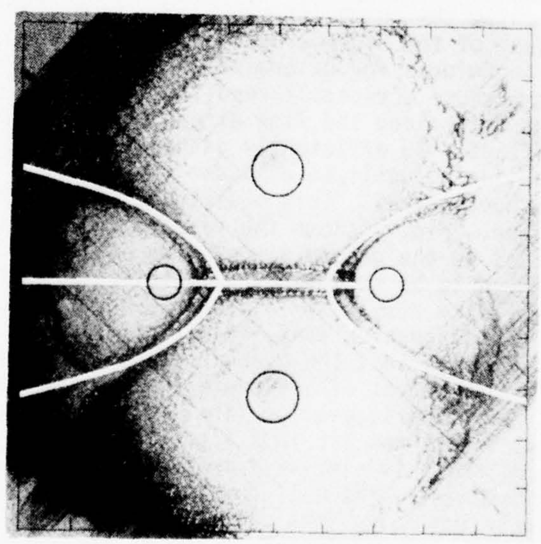
very similar to those from a simple circular jet. The flow resulting from jets of elliptical cross section showed greater differences from circular jet impingement flows. The ground plane pressure distributions were approximately elliptic, as shown in Fig. 3. The wall jet thickness grows a little more rapidly along the minor axis direction than along the major axis (Fig. 4). The biggest difference, however, is in the velocity decay shown in Fig. 5. The maximum wall jet velocity decays much more rapidly along the major axis than it does along the minor axis. The isovelocity lines appear almost rectangular rather than elliptic. We have a wide range of data for many elliptic jet conditions and are in the process of formulating models for this behavior.

The location where wall jets from adjacent nozzles meet and form the upwash (stagnation line) is the next step in the flow field calculations. We have modeled this process in several ways using various suppositions. The model which produces the best agreement with our experiments uses a balance of the component of momentum flux density normal to the stagnation line. The agreement between this model and the results for a series of two jet conditions and a more complex four jet array are shown in Fig. 6.

The local flow direction within the plane of the upwash is modeled by assuming that the component of momentum normal to the stagnation line is reflected to become the component normal to the ground plane while the component along the stagnation line is preserved. Because of the high turbulence level present in the upwash, conventional techniques for determining flow direction (tufts, smoke, etc.) did not produce acceptable results. After trying several new approaches, we determined the flow directions in that plane using small, stiff flags. To efficiently gather flow direction data with the flag technique, we utilized a ladder type array of flags and long time exposure photography. This restricted its use to upwashes lying in a plane. Fig. 7 shows the results from a pair of equal jets impinging normal to the ground plane. The agreement of the data with the model is seen to be excellent. Inclining the jets along the stagnation line also results in a planar upwash. With an inclination of 60° along the stagnation line, excellent agreement with the model is again found. Note that the model predicts the upwash flow inclinations to be independent of jet angle for impingements when the jets are inclined along the stagnation line, as shown here. For cases where the upwash does not lie in a plane, the flow inclination of the upwash surface itself is similarly modeled by momentum component balances. This model works well for the inclined upwash arising between normally impinging jets of unequal strength (Fig. 8) but not for equal strength jets impinging at angles inclined in a plane connecting the two nozzles. The reason for this is currently being investigated.



a) TWO JETS



b) FOUR JET ARRAY

Fig. 6. Comparison of Theory and Experiment for Ground Plane Stagnation Line Locations

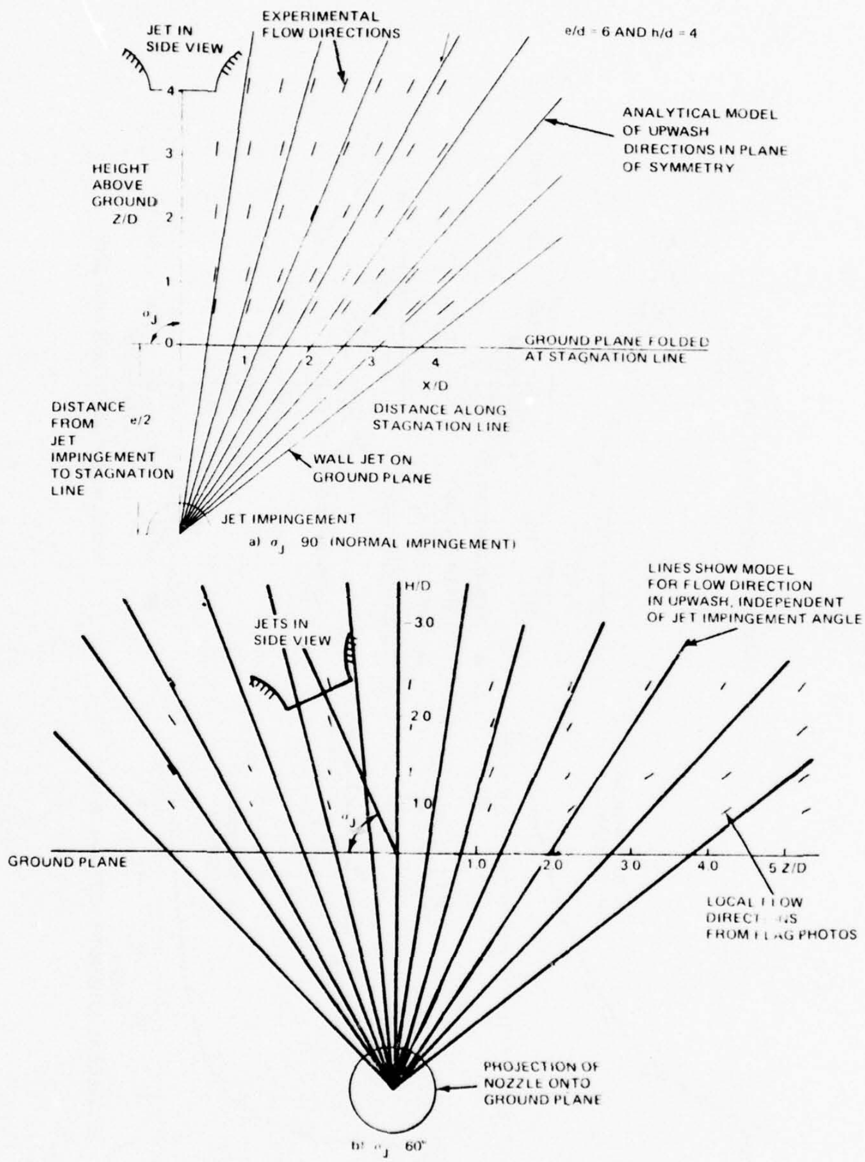


Fig. 7. Flow Direction in Upwash Between Two Equal Jets

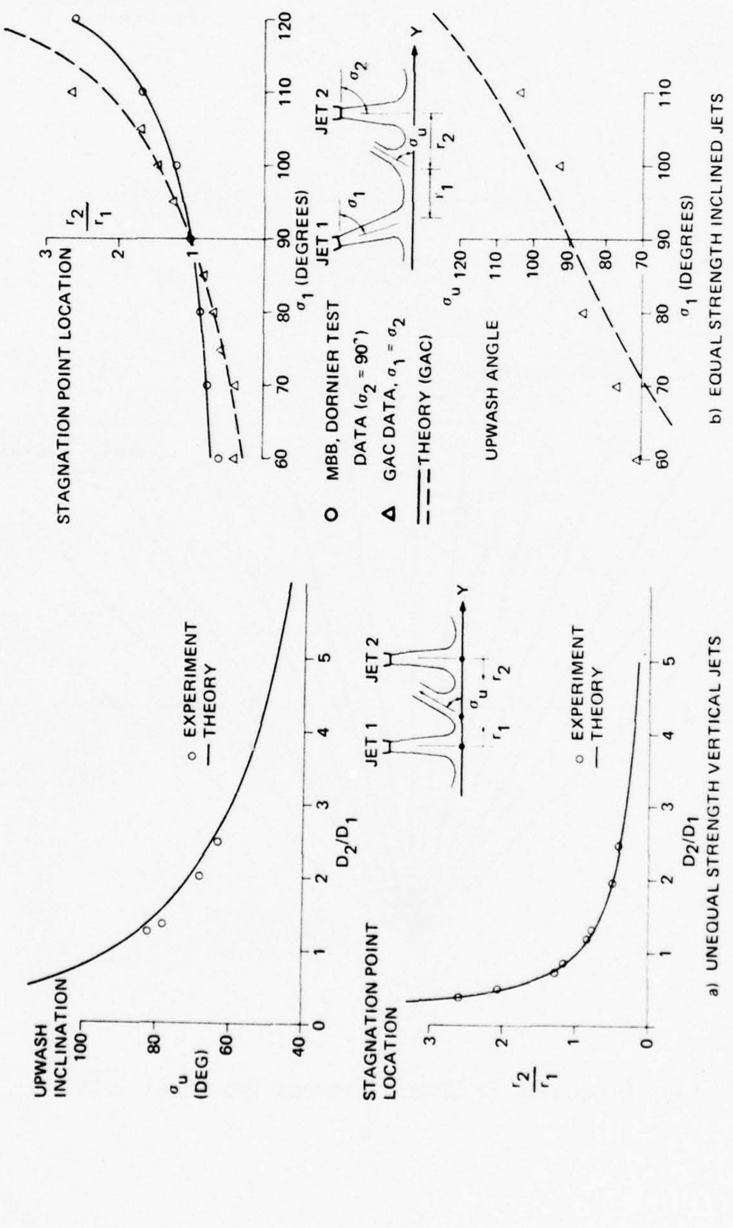


Figure 8. Comparison of Theory and Experiment for Upwash Location and Direction Between Two Jets

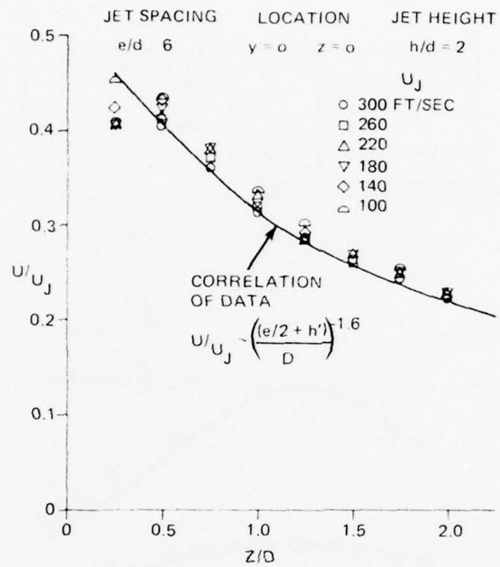


Fig. 9 Decay of Maximum Upwash Velocity With Height Above Ground Plane for Two Equal Jets Normal to the Ground Plane

The velocity on the center of symmetry of the upwash has been correlated as a power law decay, as shown in Fig. 9. The starting velocity is taken as that corresponding to a dynamic pressure equal to the maximum groundplane pressure. (This is also the peak dynamic pressure in the wall jet at the stagnation line location). The power law decay with an exponent of -1.6 is more rapid than that of a two dimensional jet. This is a product of both the streamline divergence in the pattern shown in Fig. 7 and the high turbulence level found in upwash flows. Using this decay law to predict the upwash dynamic pressure throughout the upwash produces very good agreement, as shown in Fig. 10. As the jets are inclined along the line of symmetry the same distribution of upwash pitot pressure is maintained, but location of the maximum pitot pressure moves along the stagnation line in the direction that the jets are pointed.

We do not yet have detailed data on the flow in a multiple jet fountain to complete this phase of the modeling. The success in modeling the stagnation lines for multiple jets and the details in two jet upwashes leads us to the conclusion that the same type of

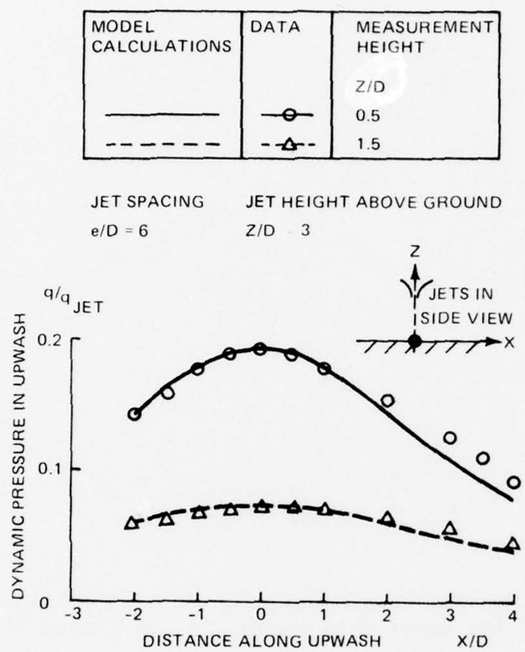


Fig. 10. Pitot Pressure Profiles Along Upwash For Two Equal Jets

approach will also be satisfactory for the central fountain.

Following the flow path shown in Fig. 2, we see that the next step is to model the fountain impingement on the vehicle and its subsequent flow around the vehicle. We have been moderately successful in predicting interference forces by assuming a total transfer of momentum from the fountain to the vehicle when the aircraft has a wide, flat bottom (Fig. 11). Methods are needed for dealing with the many new and varied classes of vehicles that are currently being designed.

AREAS MOST IN NEED OF FURTHER RESEARCH

The impingement of fountains and upwashes on the aircraft is the most important topic needing additional research. For some configurations where the fountain extent is small relative to the vehicle's planform, we have modeled the flow in a manner similar to a jet

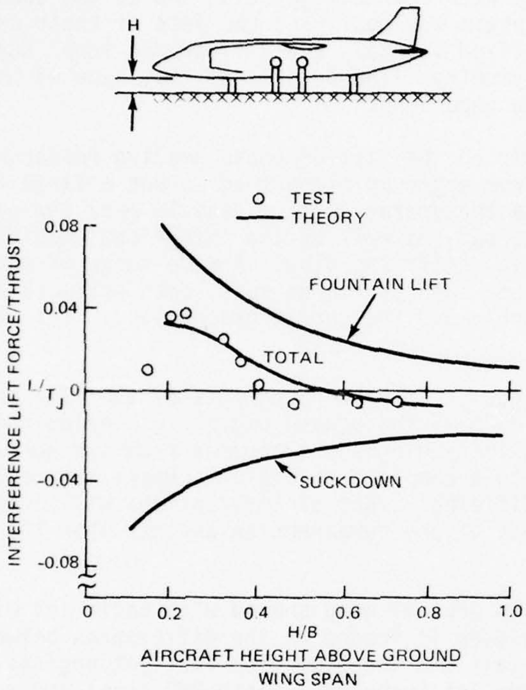


Fig. 11. Comparison Between Theory and Model Test For an Aircraft in Ground Effect

striking a flat surface. The vertical momentum in the fountain or upwash is transferred to an upward force on the aircraft, and the flow becomes a wall jet on the underside. However, many configurations will have fuselage or aerodynamic surface dimensions of the same order of magnitude as the upflow striking them. The understanding and modeling of this case will be very difficult.

A similar subject is the effect of the aircraft's presence on the development of the upflow. Our modeling of upwash flows has been quite successful using experiments with multiple free jets. When a large aircraft is present at the jet exit plane, it often strongly affects the flow. The fountain can be modified by a "back-pressure" effect, and the symmetry and stability of the flow can be destroyed. The backpressure effect should be similar to that occurring in an impinging free jet and, therefore, fairly easy to deal with. The

disturbance of symmetry and stability is a more difficult problem. Our own experience with one configuration led to the conclusion that the impinging fountain was modifying the jets at their exit plane. This led to a modified fountain and a "feedback loop" that led to instability or asymmetry. The details of this type of behavior are very configuration dependent.

The next topic on the list of those needing research is the flows resulting from a ground plane that is not a large flat surface. This would include the operation of a vehicle near the edge of a small flat landing pad, as well as the intentional modification of the ground plane to modify the flow. A wide range of modifications has been suggested, such as ribbed mats, open gridwork platforms, "holey" groundplanes, and the cupped groundplane. (cf. Refs. 3, 4, 12).

For certain geometries, the components of the flow interact with each other. The edge of the upwash begins to overlap the edge of a jet, or their proximity blocks entrainment from surrounding air and they are drawn into a complete collision. Means are needed for predicting when different types of interactions will occur and for modeling the effect of one component on another when the interference is small.

Finally, in an area of need shared with basic jet mixing technology, more knowledge is needed on the differences between the flow from scale model test and the flow from real jet engines. These differences include jet turbulence (both RMS level and scale), temperature distributions and fluctuations, and swirl.

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DISCUSSION

KOVEN: (Naval Air Systems Command)

You have described some studies of the vertical effects on the aircraft altitude developed at the suckdown and relative to the plume effects. Are you also examining the side effects as though this were an attachment that could drag the airplane sideways, so that you can examine the control forces that have to overcome that?

HILL:

Actual side force results? No. That ties into the fountain flow around the airplane configuration. We really have no way of dealing with that problem unless the airplane has a fuselage which is small compared to the upwash, so the upwash will surround it like

a body in a uniform stream, or the upwash is small compared to the fuselage bottom and it is essentially striking a flat bottom. But, for the case where the upwash is of the same dimension as the fuselage, which would lead to the kind of thing you are talking about now, we have no real general method of dealing with that at all.

SIEWERT: (Naval Air Systems Command)

What about the results you presented in Fig. 11?

HILL:

Well, they are moment data again, as opposed to side force. The moment characteristics of the configurations I have looked at also are generally very nonlinear, and they reverse directions several times making it very difficult. But, generally speaking, with any configuration, as you roll, the fountain that is coming back up moves towards the side that is lifting from the ground, which increases the disturbing moment in that same direction, so they are generally unstable due to the very existence of a fountain.

EMERSON: (Pratt & Whitney Aircraft)

My reason for asking this question is out of curiosity as to whether the effect was being accounted for in the control forces that the propulsion system has to provide in keeping the airplane stable?

HILL:

Well, I think Ray Siewert gave the message this morning that each configuration is a wholly different story, and there is no general answer. What I was trying to talk about was our efforts at developing the technology so that we do not have to build the model and run into that particular problem before we find out that it exists. That is the stage that we are looking at, and we do not have an answer to it yet.

WEINRAUB: (Naval Air Systems Command)

Did you also look at the decay of the stagnation line in the ground jet plane? In other words, if you get a stagnation line under a wing of an airplane, the amount of entrainment caused by this stagnation line could have an impact on the amount of suckdown and it would be of interest to understand the decay of the stagnation line in the ground plane.

HILL:

I have not run into any case where it appeared that the wall

jet seemed to be dependent on entrainment due to upwash itself. We found that pressure distributions were affected. You would find on the ground plane near the upwash, near the stagnation line, a region of negative pressure, while the sum of all the mechanisms, one by one, would lead to a positive or ambient pressure. When the upwash was close and surrounded the incoming jet, you would get a three-dimensional negative pressure field rather than any change in the velocity profiles or momentum content in the jet plume or the wall jet. It looks like that does not answer the question.

MURTHY: (Purdue University)

I understand that you are looking for methods to establish jet-interaction with the ground. Now, where do you include the interaction with the ground?

HILL:

We follow the flow around the jet model in a step by step fashion. The jets come down, impinge on the ground, and that impingement form's wall jets which spread out radially from each of the jet impingements.

MURTHY:

Do you have a model that does that?

HILL:

Yes, we have essentially relied upon existing models except for the noncircular jet cases. These wall jets then meet, and the location where they meet is the stagnation line location. They meet and they essentially stagnate, and their velocities turn to pressure. They they turn vertically, and the pressures convert to velocity in a vertical, or near vertical, direction, and that is the formation of the upwash. So the stagnation line location is the first step: where do they meet, what kind of conditions do you use to determine where they meet? The next step then is the local flow directions within the upwash, which I went through in the sequence. and then the velocities that occur in the upwash.

MURTHY:

I believe that level of results could be obtained with an inviscid analysis and, in fact, it may be adequate for many purposes.

HILL:

No, that is not true. The jet plume mixing is a turbulent

mixing process.

MURTHY:

Specifically, what aspects of mixing are included?

HILL:

We have used basic jet plume techniques that have developed for other reasons over the past 20 years. The wall jet impingement is essentially an inviscid process that we can treat directly. Again, we have used an empirical technique which matches observed results. We have not modeled the details within that impingement region. We have what comes out in the way of a wall jet versus what went in in the way of a jet plume model, which again is based on previously existing published data by other people. And for the wall jet flow, we have formed models for a radial wall jet flow; again the velocity profiles and the details are found to be the same as they are for classical wall jets.

WU: (The University of Tennessee Space Institute)

Is it not true that if you have a two-dimensional jet coming down, this will make the fountain jet even stronger? If this is a desirable case, your calculations should indicate that. Could you give me your comment?

HILL:

Yes. Within the wall jet itself, after your circular or finite elongated jet impinges, there is both turbulent entrainment and a radial diffusion; that is, the streamlines are spreading in a somewhat radial direction. If you had a two-dimensional jet, then you would not have that spreading component; you would only have the turbulent entrainment to weaken the jet, that is true.

WU:

Not only that is true, but in addition, if you have a two dimentional jet, the space is much more limited, so you are going to have a stronger effect due to that.

GOETHERT: (The University of Tennessee Space Institute)

I wonder, did you also do some calculations for more than two jets?

HILL:

Yes, both the stagnation line location and the upwash and fountain.

GOETHERT:

If you have more than two jets, but not aligned in a row but so that they tend to enclose an area, which might be a triangle or a square, then the situation is dramatically changed. You could create a positive pressure among the jets; you will still have the fountain effect, but very much weaker. So it is a step towards a ground-effect machine.

HILL:

There is a difference. It is not as dramatic a difference as many people seem to believe. The aircraft calculation I showed had multiple jets, the VAK 191. I do not have any surveys of the detailed upwash and fountain flow for multiple jets to do a comparison with, so I could not present that comparison. We do have the capability of calculating that.

GOETHERT:

Actually the basic idea is that you place the jets around the circumference of an area; you prevent air from flowing through the spaces between the jets. Then you get, more or less, a ground effect machine. Then you would have a positive lift force instead of a downward force. But the distance between the individual jets must not be too large.

HILL:

Well, that might be possible, but again that falls into the vehicle design category--how you want to design your airplane to try to get the best or try to get around this area of operation with the least negative effect, I guess, rather than attempting to model things in a manner that will let you compute what should happen for a reasonable airplane design.

GOETHERT:

I want to say that there should be more modeling for configurations with positive ground effect. For instance, in case that there is a full circle of individual jets, you have a really good, positive lift force. There is little sense in always looking at data where you have negative lift forces.

HILL:

That is true, but that kind of design does not usually turn out to be what is needed for the operational requirements once you take off.

HEISER: (Arnold Engineering Development Center)

The U.S. Army helicopter people succeeded in convincing me that heat transfer from the engine is a real problem. Have we already reached the point at which you need to know what the heat transfer coefficients that accompany the fountain effect are? And if not, when do we get there?

HILL:

Yes we do need heat transfer data also, but I guess my strongest point is, we have to walk before we can run. The vehicle design, being able to develop an aircraft that can take off vertically, is the biggest problem. And after we have overcome that obstacle, then we begin to worry about the temperature.

HEISER:

But that confuses me because you are going to try to augment the fountain effect in order to help the aircraft fly, and that is the very thing that helps it melt. At what point do you give up and rely upon the simple thrust of the engine to get it off the ground?

HILL:

Well, I am sure, every VTOL designer would be very happy to be able to just rely upon the thrust of the engine to get it off the ground because you still have to fly through that flight regime anyway. Any positive effect that you get close to the ground is of a very minor importance because you then go up a few feet higher and you lose that effect. So, really, what we are trying to do is develop models for understanding and being able to predict these effects so that we do not search in a blind testing program through all the many different thousands of parameters we can play with to design the aircraft, as opposed to trying to get the greatest positive effect.

THE AIRJET DISTORTION GENERATOR SYSTEM: A NEW TOOL FOR AIRCRAFT TURBINE ENGINE TESTING*

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ABSTRACT

An airjet distortion generator system has been developed to produce steady-state total pressure distortion at the inlet of turbine engines. The system employs a method of injecting controlled amounts of high-velocity secondary air counter to the primary airstream to effect a local total pressure decay. Digital computer control provides an on-demand distortion pattern capability. The AJDG system is described, and the pattern-generating logic is presented. Operational characteristics, turbulence, cycle times, and distortion pattern fidelity are discussed. An engine stability assessment with comparison of stability response to screens and airjet-produced inlet distortion is included.

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*The work reported herein was performed by the Arnold Engineering Development Center (AEDC), Air Force Systems Command (AFSC). Work and analysis for this research was done by personnel of ARO, Inc., a Sverdrup Corporation Company, operating contractor of AEDC. Further reproduction is authorized to satisfy needs of the U. S. Government.

Nomenclature

AJDG	= airjet distortion generator
EI	= local total pressure error at each spatial location at engine inlet
P2AVG	= average total pressure at engine inlet, psia
P2DIST	= overall distortion level, $\frac{P2MAX - P2MIN}{P2AVG}$
P3Q24X	= engine high-pressure compressor pressure ratio
PLPQ2	= engine low-pressure compressor pressure ratio
PRDI	= ratio of desired local to average total pressure at engine inlet
PRMI	= ratio of measured local to average total pressure at engine inlet
PSD	= power spectral density
RMSE	= overall pattern error, based on desired and measured total pressure values at N spatial locations, $\sqrt{\frac{\sum_{I=1}^{N}\left[\frac{PRMI}{PROI} - 1\right]^2}{N}}$
T2CLEAN	= average total temperature in the undistorted (high-pressure) area at the engine inlet,°F
T2DIST	= average total temperature in the distorted (low pressure) area at the engine inlet, °F
WA24R24	= corrected airflow rate at the hig-pressure compressor inlet, lbm/sec
WA2R2	= corrected airflow rate at the engine inlet, lbm/sec
WSQ2	= ratio of secondary (airjet) airflow to primary (engine) airflow

INTRODUCTION

The recent increase of emphasis on the effects of inlet total-pressure distortion on turbine engine stability and performance has resulted in a major effort at ground test facilities to improve the duplication of the inlet total-pressure profiles encountered during operation of engines over the aircraft flight envelope. An engine will encounter a variety of distortion patterns over a wide range of engine airflow rates. In order to adequately define the engine stability characteristics, testing with a large number of unique distortion patterns is required. The most widely accepted approach to producing the distortion patterns has been the use of complex assemblies of various porosity screens. The inherent inflexibility

of the screen configuration (single design operating point) and the extensive development effort required for each screen dictated the need for a more flexible method of producing total-pressure distortion. In response to this need, an effort to provide an alternate method has been in progress at the Arnold Engineering Development Center (AEDC) during recent years.[1-3]

The airjet distortion generator (AJDG) system is a method for producing steady-state, total-pressure spatial distortion at the inlet to a turbine engine. The airjet system (Fig. 1) produces steady-state distortion by injecting secondary air counter to the primary airflow. By injecting a controlled amount of secondary airflow in specific spatial locations, a wide range of inlet distortion patterns can be produced. Digital computer control of secondary airflow provides a dial-a-pattern capability that makes the airjet system a highly flexible and efficient test tool.

AIRJET DISTORTION GENERATOR SYSTEM

The AJDG system consists of two basis subsystems, an air supply temperature- and pressure-conditioning system and an airflow distribution system.

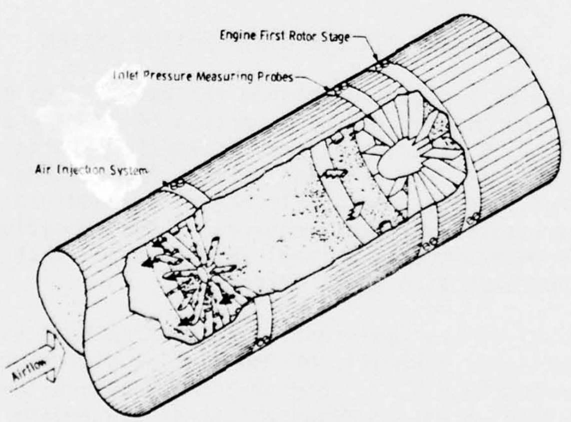

Figure 1. Airjet Distortion Generator Installation

The air supply temperature- and pressure-conditioning system conditions the secondary air to the temperature level required to match the primary engine inlet air temperature and throttles the secondary airflow to produce a desired pressure level at the supply manifold.

The airflow distribution system meters secondary airflow to each of 56 injection ports as required to produce the desired total-pressure decrement at each spatial location. A functional schematic of the AJDG system is presented in Fig. 2.

AIR SUPPLY TEMPERATURE AND PRESSURE-CONDITIONING SYSTEM

Filtered secondary air is pressure regulated through a flow-measuring system, then directed through either the high- or low-temperature-conditioning system.

Each temperature-conditioning system consists of an airflow leg through a heat exchanger and a by-pass airflow leg. Desired air temperature at the outlet of the air-conditioning system is attained by mixing air from the heat exchanger with ambient supply air. Secondary-air conditioning is accomplished using either steam or liquid nitrogen as the heat-transfer medium.

Temperature-conditioned air is then delivered to a distribution manifold at a specified pressure.

SECONDARY AIRFLOW DISTRIBUTION SYSTEM

Conditioned air is distributed from the manifold to the desired spatial location and counter to the direction of primary airstream flow using a set of 56 metering valves, connecting tubing, and drilled passage struts. The metering valves regulate the flow rate through each strut passage (spatial location) as required for the specific total-pressure decrement required in that location. Manifold pressure is maintained at a level which ensures that each strut discharge port operates as a sonic orifice.

The airjet valves are individually controlled by a digital computer. A functional schematic of the airflow distribution system control is presented in Fig. 3. Engine inlet pressure level is determined from total-pressure measurements at the engine face. The pressure levels measured at the engine inlet are transposed to equivalent locations (comparable flow area for each pressure value) at the plane of the jets and normalized by the face average pressure. The computer compares the actual pressure level at each spatial location to the desired level and commands the airjet valves to either open or close as required to establish the desired pressure levels.

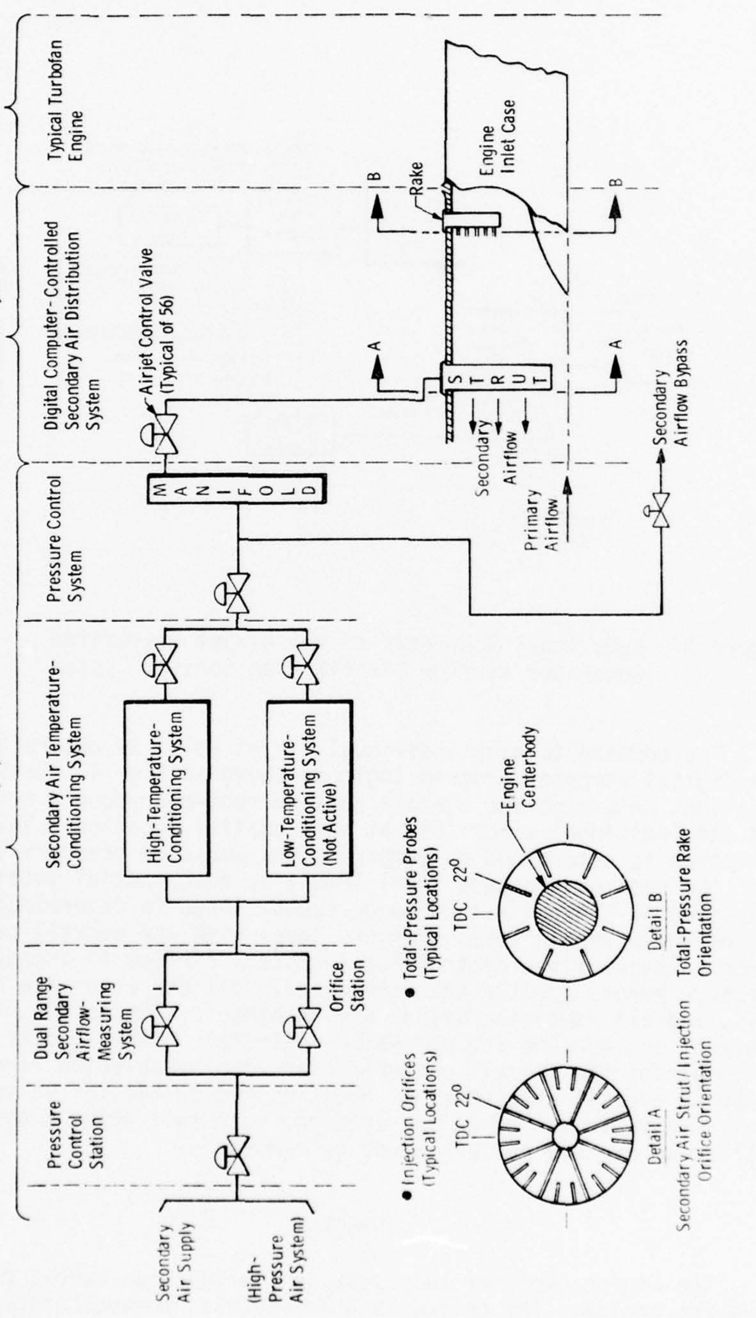

Figure 2. Functional Schematic of the Airjet Distortion Generator System

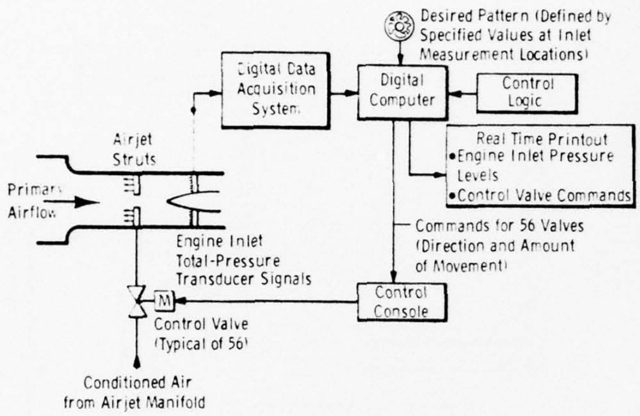

Figure 3 Functional Schematic of the Airjet Distortion Generator Airflow Distribution Control System.

The command to each individual airjet valve is determined by the digital computer program logic, as shown in Fig. 4. Basic logic functions determine the overall pattern root-mean-square error (RMSE) and the individual error (EI) at each spatial location. Valve direction is determined by comparing the measured pressure level (PRMI) with the desired pressure level (PRDI) at each spatial location. The selection of control valves to be repositioned is determined by comparing the error in local pressure level with the overall pattern error. Those valves controlling secondary airflow to areas with local pressure errors greater than the overall pattern error are directed to move, and all remaining valves are unchanged. The amount of valve movement is the same for all valves selected to move and is determined by comparing the overall pattern error with preselected ranges. The range of overall pattern error dictates the particular valve travel time. Valve travel times are selected such that valve travel becomes smaller as overall pattern error is reduced.

TURBOFAN

The engine used for this test is a production model, present-day turbofan engine. The engine is a low-bypass, nonaugmented turbofan

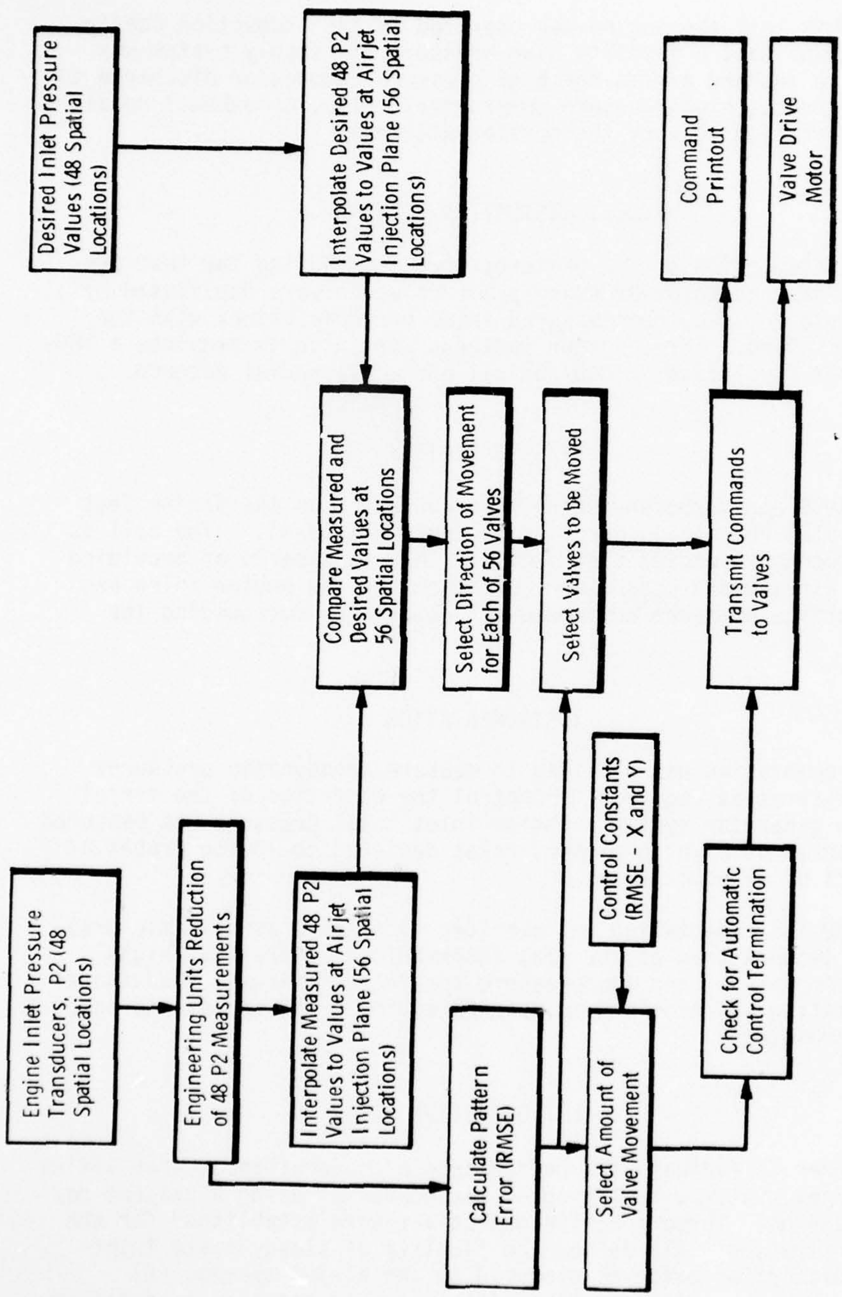

Figure 4. Compuger Control Logic for Airjet Distortion Generator Airflow Distribution System

engine of the 15,000-lbf-thrust class.

For this test the engine was operated in the production configuration, except that a facility high-pressure air supply system was installed to inbleed air to the high-pressure compressor discharge to back-pressure the high-pressure compressor. Also, an exhaust nozzle plug was installed to vary the nozzle exit area.

INLET DISTORTION SCREENS

Three distortion screen patterns were used during the test program. The desired total-pressure patterns which were duplicated by the AJDG were defined from measured inlet pressure values with the screens installed. Three screen patterns were used to simulate a 180-deg, one per revolution; a hub radial; and a tip radial pattern.

TEST FACILITY

The AJDG and turbofan engine were evaluated in the Engine Test Facility (ETF) Propulsion Development Test Cell (T-4).[4] The cell is a "direct-connect" engine test facility that is capable of providing specified airflow and stagnation conditions at the engine inlet and the appropriate pressure environmental conditions surrounding the engine.

INSTRUMENTATION

Instrumentation was provided to measure aerodynamic pressures and temperatures as required to control the operation of the airjet distortion generator system. Engine inlet total pressure was measured using an array of eight 6-element rakes designed to locate probes at the centers of 48 equal areas.

Engine instrumentation was provided to measure aerodynamic pressures and temperatures at the fan, intermediate compressor, high-pressure compressor, and low-pressure turbine discharges. Additional instrumentation was provided to measure engine rotor speeds and operational parameters.

TEST OBJECTIVES

In order to evaluate the performance and operational capabilities of the airjet system, a test program was conducted using a present-day turbofan engine. Three specific objectives were established for the test demonstration: (1) define the fidelity of steady-state inlet pressure distortion patterns produced by the airjet system, (2) determine the capability of the airjet system to maintain a specified

composite pattern over a range of airflow rates, and (3) document any differences in engine surge margin with the same inlet distortion produced by distortion screens and by the airjet system.

TEST PROCEDURE

All testing was conducted with the average engine inlet air total pressure and temperature defined for the flight condition - 45,000 ft altitude, Mach number 1.2 (Hot Day).[5,6]

The pattern capability of the airjet system was demonstrated by operating the engine at selected airflow levels on its normal operating line. With the engine stable at a normal operating point, the airjet system was activated and changed the inlet pattern from "clean" to the specified pattern.

For the engine stability test the engine was loaded using in-bleed air to the high-pressure compressor discharge and a remotely positioned exhaust nozzle plug. After the inlet distortion pattern had been established the loading systems were activated and the engine driven to surge while holding engine rotor speed match essentially constant. Small variations in engine power setting were used to maintain a constant engine airflow.

INLET TOTAL-PRESSURE PATTERN FIDELITY

The fidelity of the inlet distortion pattern produced by the AJDG system was evaluated for three parametric inlet distortion patterns (180-deg, one per revolution; tip radial; hub radial). Each pattern was first produced and measured using a distortion screen installed in the engine inlet ducting. The AJDG system was then used to reproduce the inlet pressure pattern measured with the screen installed.

STEADY STATE, TOTAL-PRESSURE DISTORTION

Steady-state, total-pressure distortion pattern quality can be described by the pattern characteristic appearance and distortion level, P2DIST. Typical pattern characteristics, as shown by the isobar map at the engine inlet, are presented in Fig. 5. For each pattern, the AJDG system produced similar areas of high and low total pressure and maintained area contours similar to those produced by the distortion screens. The distortion level of each parametric pattern produced by the AJDG system agreed with the screen-produced distortion level within three-percent absolute distortion.

Although pattern characteristics and distortion level are good

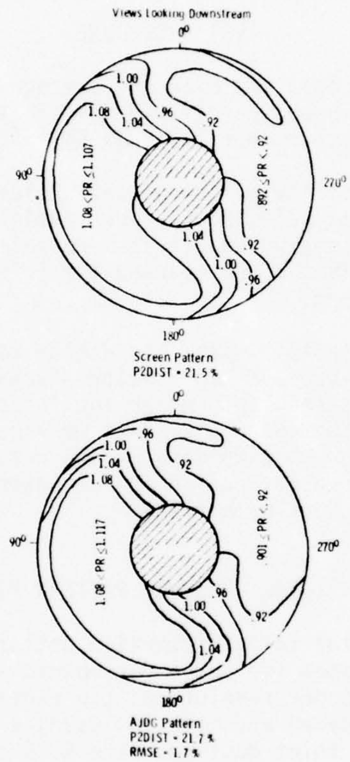

Figure 5. Engine Inlet Isobar Maps for Airjet and Screen Produced 180-Deg, One-per-Revolution Pattern

indications of pattern quality, the specific definition of each inlet pattern should be made on the basis of a comparison of individual pressure levels at the specific spatial locations. Individual pressure values for the 180-deg pattern are compared in Fig. 6.

The overall agreement between the measured and desired local pressure levels can be quantified by the RMSE. For the three patterns, the RMSE ranged from ± 0.7 to ± 2.3 percent. The largest RMSE generally occurred at the highest distortion levels.

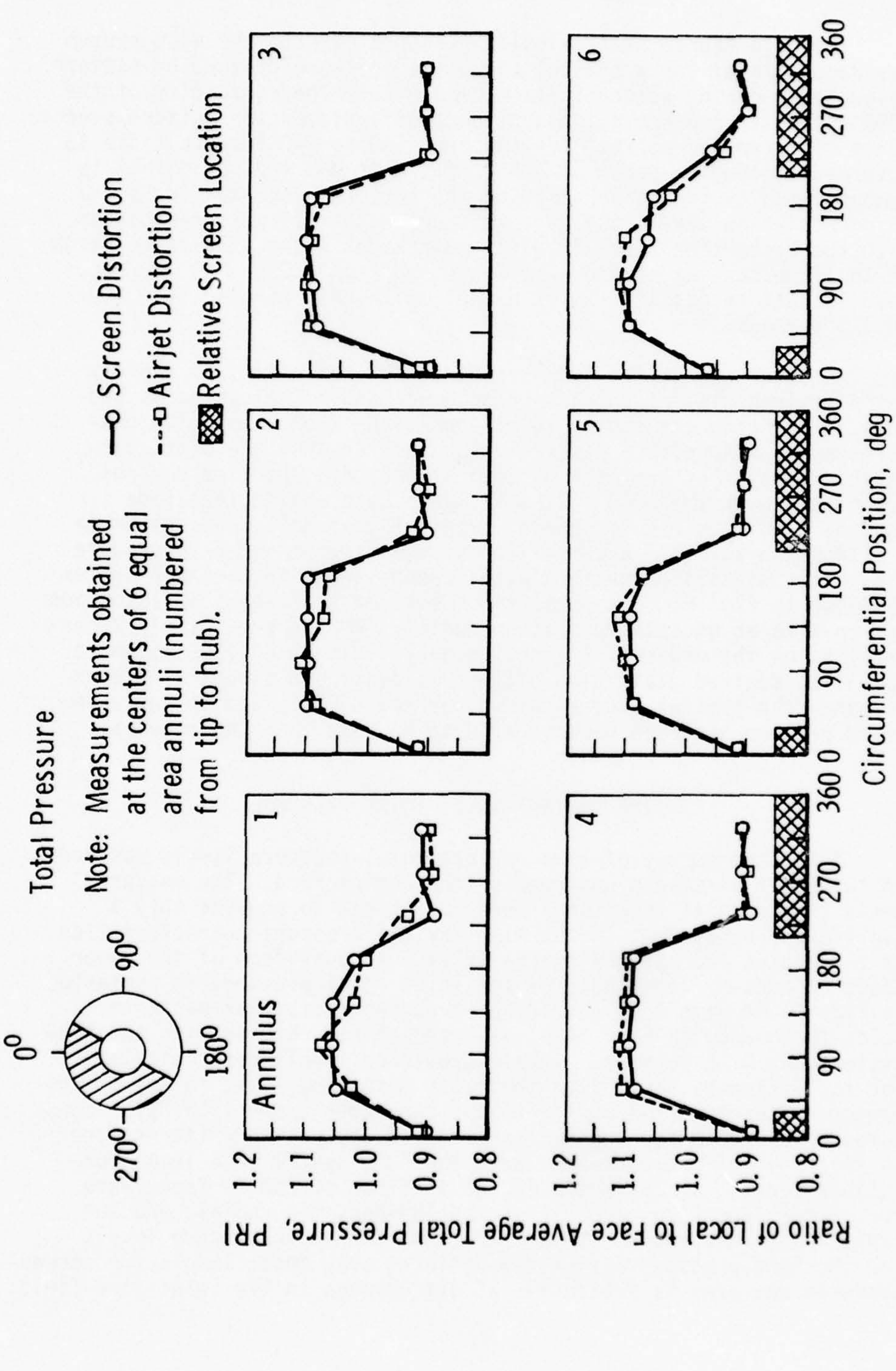

Figure 6. Inlet Pressure Profiles for the 180-Deg. One-per-Revolution Distortion Pattern (WA2R2 ~ 200 lbm/sec)

INLET TOTAL-PRESSURE PATTERN REPEATABILITY

The high degree of flexibility associated with the AJDG system was demonstrated for a typical composite pressure distortion pattern encountered during aircraft flight maneuver. The capability of the AJDG system to produce a constant composite distortion pattern over a range of corrected engine airflows from 160 to 240 lbm/sec (idle to intermediate engine power at 45,000 ft, Mach No. 1.2 condition) is demonstrated by the isobar maps of the pattern presented in Fig. 7. At each airflow level, the pattern characteristics were reproduced with the distortion level (P2DIST) maintained within the range from 11 to 15 percent as airflow was increased from 160 to 240 lbm/sec. Overall pattern quality was excellent, with RMSE ranging from 1.0 to 1.4 percent.

An advantage of the AJDG system is the capability to rapidly set a desired distortion pattern upon command. The system is capable of changing the engine inlet conditions from clean (low distortion) to a specified distortion pattern or from one distortion pattern to another during a given test period. The time savings associated with the AJDG system includes both engine test time and the time required to design, fabricate, install, and calibrate a distortion screen. A comparison of the time required to produce a specific distortion pattern with screens and with the AJDG system is shown in Fig. 8. The total time required to develop a distortion screen with an acceptable pattern quality (RMSE approximately 2 percent) is on the order of 12 working days. With the AJDG system, a specific, desired distortion pattern is available essentially upon command; the typical time required for the AJDG system to set a desired pattern has been demonstrated to be less than two minutes.

TIME VARIANT INLET TOTAL PRESSURE

A limited survey of time variant total-pressure levels was made in the inlet airstream upstream of the engine face. The measurements (two spatial locations) were sufficient to provide only a qualitative assessment of the time variant pressure characteristics at the engine face of the engine inlet. A comparison of the power spectral density functions for the inlet total pressure is presented in Fig. 9 for both AJDG and screen-produced distortion patterns. Inlet turbulence (ΔPRMS/P2AVG) was consistently higher with the AJDG system than with screens. Inlet turbulence levels were calculated for the frequency range from the lower measuring limit (5 Hz) to the frequency level corresponding to the fan rotor speed (160 Hz). Local turbulence levels for the screen-produced distortion patterns were on the order of one percent. With the AJDG system, the local turbulence levels, at corrected engine airflows of 200 lbm/sec, were two, three, and five percent for the 180-deg, tip radial, and hub radial patterns, respectively. The increase in turbulence levels for the AJDG-produced distortion patterns over those levels for screen-produced patterns is indicative of differences in the inlet flow field.

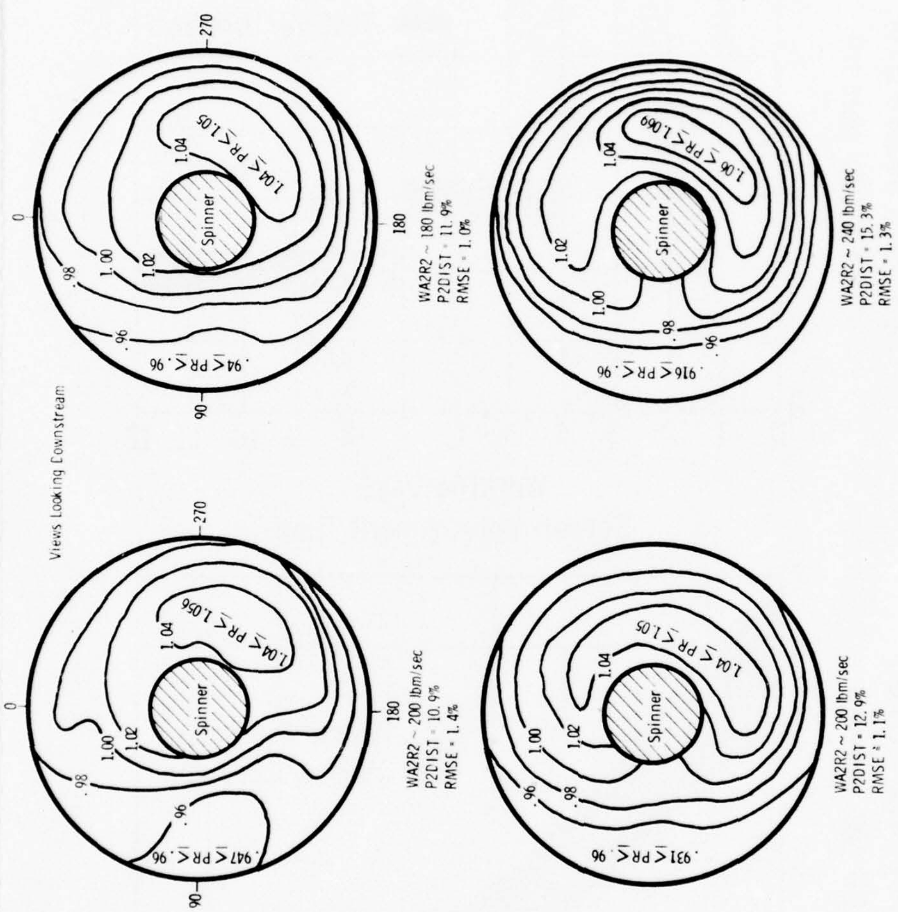

Figure 7. Engine Inlet Isobar Maps for the Airjet Distortion Generator-Produced Composite Pattern

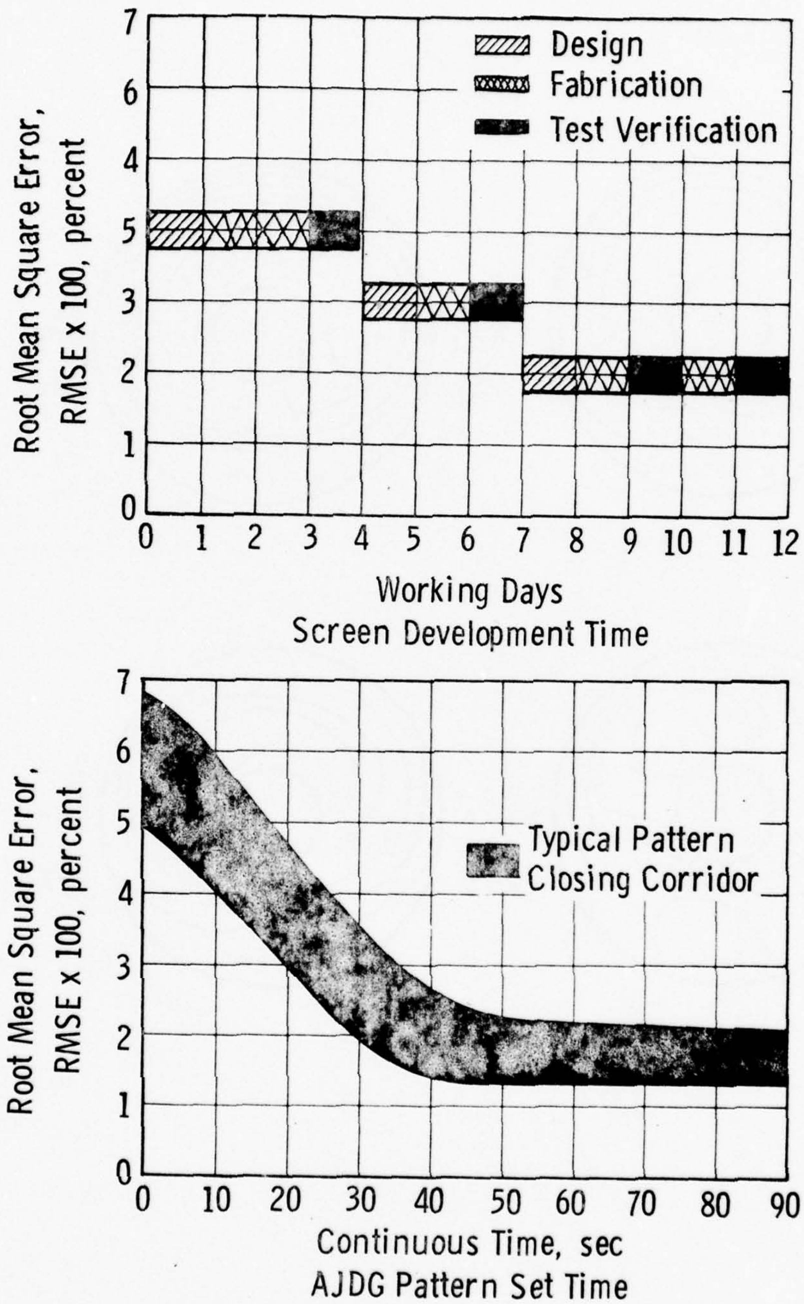

Figure 8. Relative Time Requirements to Produce a Specified Distortion Pattern with Screens and with the Airjet Distortion Generator System

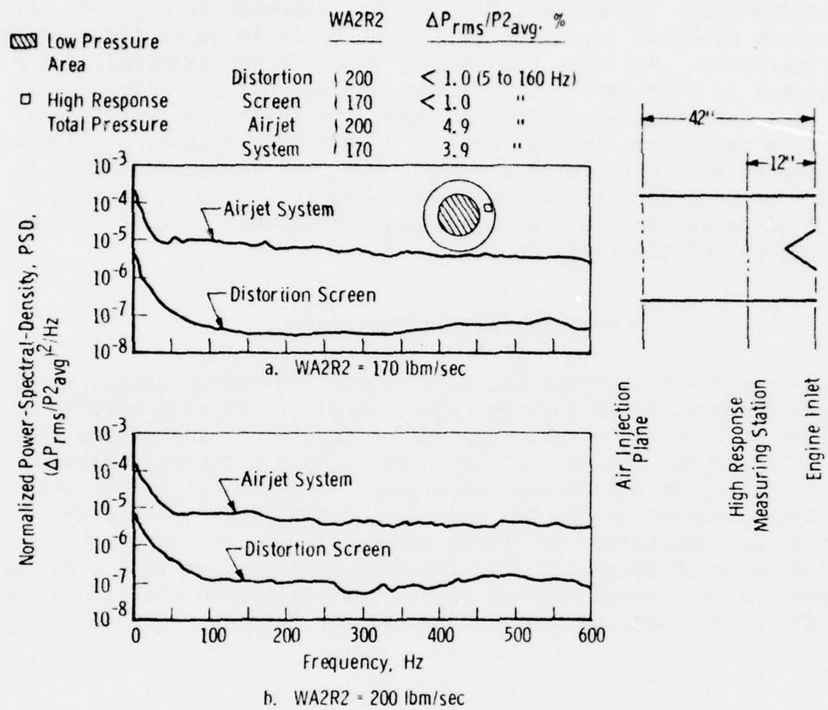

Figure 9. Power Spectral Density Characteristics Hub Radial Distortion Pattern

STEADY-STATE, INLET TOTAL-TEMPERATURE MATCH CAPABILITY

Testing of the AJDG system with the engine installed was conducted using secondary air that was temperature conditioned to match the primary engine supply air temperature within ± 3°F. In addition, an evaluation of the inlet temperature error (difference in measured temperatures downstream of flowing jets and nonflowing jets) resulting from primary and secondary air mixing was conducted during testing with an engine inlet simulator. In this evaluation, the primary and secondary air temperatures were intentionally mismatched in selected increments up to 15°F at various levels of secondary airflow. Engine inlet temperature error as a function of primary

and secondary air temperature mismatch is presented in Fig. 10. At temperature mismatch levels up to 3°F, no error in inlet temperature was discernible. An inlet temperature error of approximately 0.5°F was evident at a primary and secondary air temperature mismatch of 6°F. At a temperature mismatch of 10°F, the inlet temperature was 1.7°F at a secondary airflow level of 13 percent of total airflow. Inlet temperature error with a mismatch of 15°F increased from 1.1 to 3.1°F as secondary airflow was increased from 5 to 13 percent of total airflow and to 3.3°F as secondary airflow was further increased to 19 percent of the total airflow.

ENGINE STABILITY RESPONSE

The currently accepted method of producing steady-state, total-pressure distortion for turbine engine stability testing uses the technique of installing various porosity screens in the engine inlet. In order for the AJDG to be an acceptable alternate method, it is necessary to define any differences in engine stability with distortion produced by the two methods. During this test, engine stability was determined for three parametric distortion patterns produced by inlet screens and by the AJDG system. The test procedure provided a direct comparison of engine operation with the same steady-state distortion pattern produced by the two methods. Engine

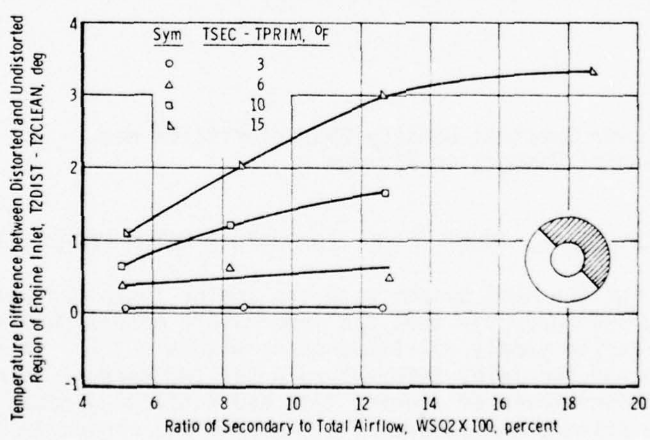

Figure 10. Effects of Primary and Secondary Air Temperature Mismatch on Engine Inlet Temperature

stability was also determined with and without the airjet struts installed (no secondary airflow) to evaluate the effects of the struts on the clean inlet flow pattern.

AJDG INSTALLED, NO SECONDARY AIRFLOW

In order to increase the test configuration flexibility, it is desirable to have the capability of conducting either clean inlet or distortion testing with the same hardware installation. Baseline engine stability was determined without the airjet struts installed and with the airjet struts installed, but with no secondary airflow.

The steady-state, inlet distortion level was not affected by the installation of the AJDG struts. For both installations, the inlet distortion (P2DIST) was nominally 0.5 percent at the highest corrected airflow (WA2R2 = 200 lbm/sec). Inlet total-pressure turbulence was less than one percent for both installations. There was no detectible difference in engine surge margin for the two clean inlet configurations.

PARAMETRIC INLET DISTORTION PATTERNS

The engine stability response to three basic parametric inlet distortion patterns (180-deg, tip radial, and hub radial) was determined. Each pattern was first produced with an inlet screen, and then the screen pattern was reproduced with the AJDG system. Engine stability testing was accomplished with the three basic patterns at nominal engine airflow rates (WA2R2) of 170 and 200 lbm/sec.

180-DEG DISTORTION PATTERN

High-pressure compressor surge occurred with the 180-deg inlet distortion pattern and the fan and high-pressure compressor loading. The operating map for the high-pressure compressor with the 180-deg engine inlet distortion pattern imposed is presented in Fig. 11.

The HPC surge point repeatability with distortion produced by each method (screen and AJDG) was approximately two percent. The average HPC surge pressure ratio with the AJDG produced 180-deg distortion patterns was one percent lower than the surge line defined with screen-produced distortion.

TIP RADIAL DISTORTION PATTERN

Fan surge occurred with the tip radial pattern imposed at the engine

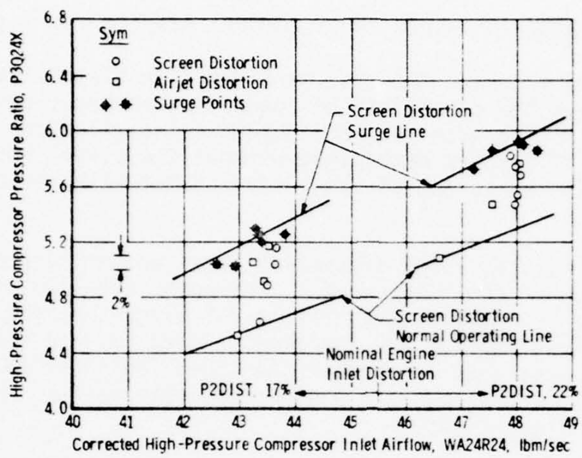

Figure 11. Comparison of High-Pressure Compressor Performance with Screen and with Airjet Distortion Generator 180-Deg Engine Inlet Distortion Pattern, 45,000 ft, Mach No. 1.2

inlet. The operating map for the fan with tip radial engine inlet distortion is presented in Fig. 12.

At the lower engine airflow (WA2R2 $\approx$ 170 lbm/sec), the fan surge pressure ratios with AJDG distortion ranged from one to three percent lower than the fan surge line with inlet screen distortion. At the higher engine airflow (WA2R2 $\approx$ 200 lbm/sec), the fan surge pressure ratios with AJDG distortion agree with the surge line defined with screen distortion within one percent.

HUB RADIAL DISTORTION PATTERN

The combination of the hub radial engine inlet distortion pattern and the engine loading resulted in an HPC surge. The high-pressure compressor operating map with hub radial engine inlet distortion is presented in Fig. 13.

The HPC surge pressure ratios were lower with AJDG-produced distortion than the surge line defined with screen distortion. At the lower engine airflow (WA2R2 $\approx$ 170 lbm/sec), the HPC surge with AJDG distortion occurred at pressure ratios ranging

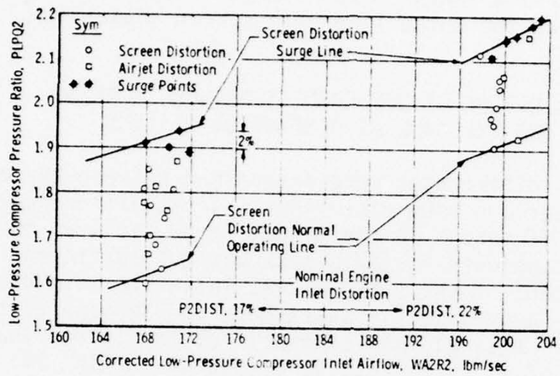

Figure 12. Comparison of Low-Pressure Compressor Performance with Screen and with Airjet Distortion Generator Tip Radial Engine Inlet distortion Pattern, 45,000 ft, Mach No. 1.2

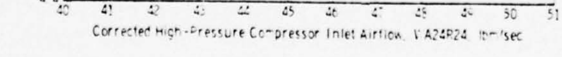

Figure 13. Comparison of High-Pressure Compressor Performance with Screen and with Airjet Distortion Generator Hub Radial Engine Inlet Distortion Pattern, 45,000 ft, Mach No. 1.2

from four to seven percent below the surge line defined with screen distortion. High-pressure compressor surge pressure ratios at the higher engine airflow (WA2R2 $\approx$ 200 lbm/sec) were seven to eleven percent lower than the surge line with screen distortion.

EVALUATION OF DIFFERENCES BETWEEN SCREENS AND THE AJDG AS DISTORTION SYSTEMS

Significant differences were determined between the engine surge margin with screen and AJDG distortion. These differences are probably the result of differences in the dynamic flow field at the engine inlet. The instrumentation used for this test was not sufficient for a quantitative assessment of the time variant inlet flow-field characteristics; however, a qualitative assessment can be made of the dynamic characteristics. A simplified schematic of the inlet flow field which categorizes the inlet turbulence is presented in Fig. 14. By associating relative turbulence intensity with inlet flow area, a qualitative evaluation may be made of its impact on engine stability margin.

The flow field downstream of the AJDG may be associated with three flow zones: (1) uniform mixing zone-- that area downstream of the counterflowing jets that is affected only by the pressure loss mechanism of counter-flow, (2) nonmixing zone-- that area downstream of the nonflowing jets that is unaffected by the flowing jets, and (3) nonuniform mixing zone-- that area that encompasses the transition from the uniform mixing zone to the nonmixing zone. The relative extent of each flow zone for a specified distortion level (P2DIST) is dependent on the pattern characteristics (shape and proximity of low-pressure boundary to duct wall), primary air velocity, and secondary airflow rate. The progressive increase in the size of the nonuniform mixing zone that occurs with increasing secondary airflow and decreasing primary air velocity is shown schematically for a hub radial pattern in Fig. 14. The nonuniform mixing zone can range from a small part of the area (Fig. 14a) to the extreme condition at which the nonuniform mixing zone becomes large enough to completely eliminate the nonmixing zone (Fig. 14c).

The size of the nonuniform mixing zone is reflected by the extent of total-pressure loss across the inlet duct in the area of nonflowing airjets. For the hub radial pattern produced by the AJDG system, a total-pressure loss extended across the entire radius of the inlet duct. The flow field associated with the AJDG hub radial pattern then conformed to the dynamic flow field in which nonuniform mixing occurs over the majority of the flow area, and an area of nonmixing does not exist.

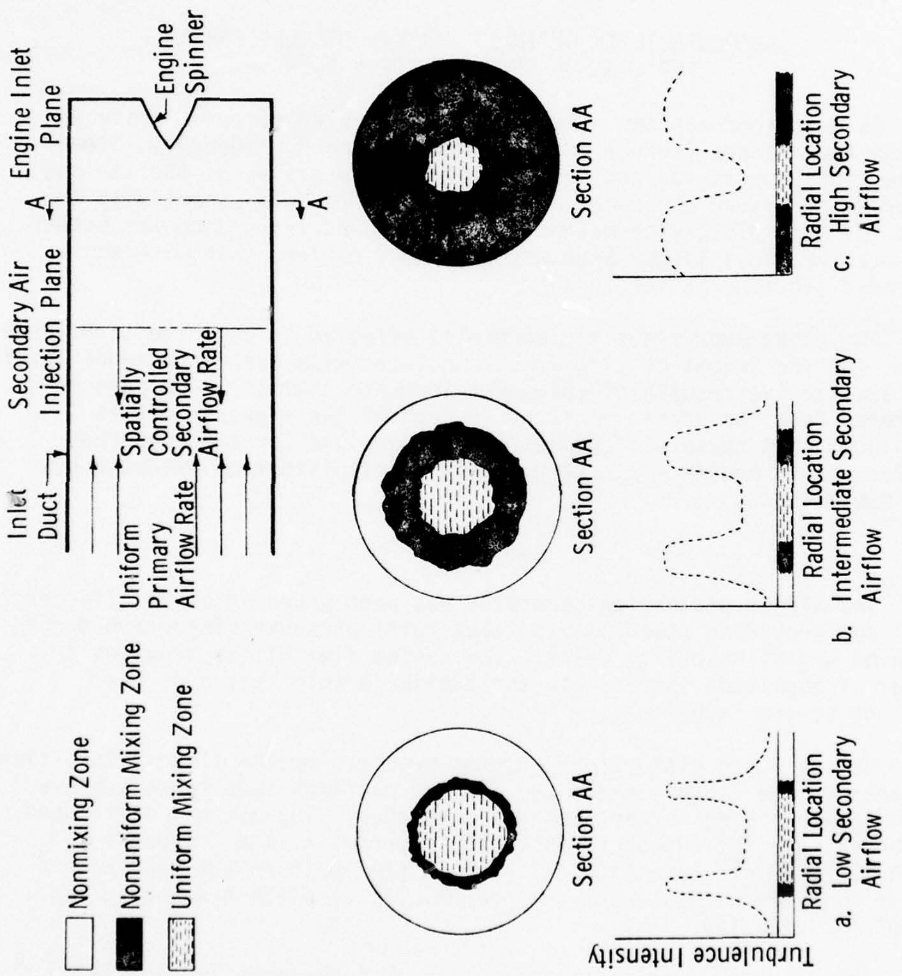

Figure 14. Schematic Representation of a Planar Turbulence Distribution Existing with Airjet Distortion Generator Hub Radial Distortion Patterns

For this inlet flow field, it is surmised that the highest turbulence levels occur in the nonuniform mixing zone. The AJDG system then produced a hub radial pattern that included a relatively large area with the highest turbulence. This is dissimilar to that obtained with the distortion screen. The extent of the high shear zone associated with the screen was small, relative to that for the AJDG as reflected by the absence of total-pressure loss in the undistorted (clean) area.

ACCOUNTABILITY OF INLET PATTERN DISTORTIONS FOR LOSS OF ENGINE SURGE MARGIN

As described earlier, the steady-state inlet patterns produced by the two methods (screen and AJDG) were in good agreement. Steady-state, total-pressure, and total-temperature profiles at the various compression system component measuring stations were essentially the same for both distortion methods. The measured turbulence was higher and occurred in a larger area with the AJDG pattern than with the screen distortion pattern.

The compressor stability margin is affected by both the turbulence level and the amount of flow area associated with each turbulence level. The qualitative results of this test indicate that the AJDG patterns differed from the screen patterns in both of these areas. It is concluded that these differences are responsible for the measured difference in engine surge margin with inlet distortion produced by the two methods.

SUMMARY

The airjet distortion generator has been shown to be an efficient tool for providing steady-state inlet total pressure distortion during turbine engine stability tests. The system flexibility provides an order of magnitude increase in the available inlet patterns over current screen techniques.

Steady-state distortion patterns produced by the airjet distortion generator more closely reproduce desired patterns than those achieved with screens using current design techniques. The use of conditioned secondary air introduced no discernible error in total temperature. Turbulence levels associated with the airjet pattern are higher than those measured with screens and are related to pattern severity and inlet air velocity.

Engine surge margin was responsive to turbulence levels. With the present airjet system, the inlet flow field must be completely defined, i.e., both steady-state and time-variant pressure will be defined on an instantaneous rather than a time-averaged basis.

Further refinement of the system is desirable to reduce the turbulence to a negligible level. This would allow the use of far less

area with the highest turbulence. This is dissimilar to that obtained with the distortion screen. The extent of the high shear zone associated with the screen was small, relative to that for the AJDG as reflected by the absence of total-pressure loss in the undistorted (clean) area.

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Further refinement of the system is desirable to reduce the turbulence to a negligible level. This would allow the use of far less

complex inlet instrumentation and associated conditioning and reduction equipment.

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DISCUSSION

REESE: (Purdue University)

It has never been clear to me why total pressure distortion is the right way to determine what happens to the compressor. It has been my understanding that it really is the magnitude and the direction of the velocity as opposed to the magnitude itself that is of importance. I know the total pressure correlates those cases that do have bad dynamic total pressure distortions, but it has never been clear to me what the relationship is between a stall in the compressor and total pressure distortion.

OVERALL:

Well, I would really rather defer that to a compressor man. I guess I am not that familiar with the mechanics of the stall.

STEENKEN: (General Electric Company)

I think the answer to that question is partially tied up with the fact that the engine pumping characteristics, in conjunction with the total-pressure distortion pattern, uniquely determines the static-pressure distortion and swirl velocity distributions. Given a steady-state pattern, it fixes these other components of the flow, and therefore if you fix one you have fixed all. Hence if you measure one (component), it then turns out to be, as a general approximation, that you are measuring a quantity related to the end effect. We have been doing some flow field calculations as an extension to some of the work I discussed yesterday. When you impose a total-pressure distortion, you can follow the development of the swirl and static-pressure distortion that ensues through the flow field. I think this is probably the reason that total-pressure distortion is related to stall. However, your point is well taken. If you think in terms of flow processes and you want to define the state of the fluid, it normally involves specifying three variables of the flow. The fact that we have been so lucky (to correlate stall with total-pressure distortion), so to speak, is probably the result of the relationship which exists between the three flow variables and the boundary conditions.

DENNING: (Rolls-Royce (1971) Limited)

Is it not true to say that until now we have tested engines, we have used gauzes, rotating gauzes, and inlets? It has been a standard procedure that you took an engine on the test bed and you had a device which, early in the development, could determine whether that engine would stand a certain total pressure distortion. As I understand it, what you are trying to do is to make a more elegant device for producing total pressure distortion. It occurs to me that in the context of this meeting, we have identified a need for a total temperature distortion device and the sort of system that you have pursued, where you inject air into the inlet, might well be a way of producing a total temperature distortion which can be set up rapidly. Have you thought of that?

OVERALL:

Yes, the potential is there in the system. Of course we are still in the development stage of the system. This will require basically a more complicated computer control system, and that is really, I guess, where our hangup will be. The system capability is there.

DUPCAK: (Lockheed Aircraft Company)

Could you give us an idea of what kind of flow ratios of primary and secondary air flows are available, and if they are low, how do you propose to create temperature distortion? In other words, would you not need a lot more air flow?

OVERALL:

For these patterns, we were running at about 20 percent distortion level. The Mach number was about three-tenths; for that we were running normally a secondary-to-total ratio of about 15 percent.

GOETHERT: (The University of Tennessee Space Institute)

I would like to point out that this is really a method which is much more flexible than the screen method. You see, when you want to change the distortion pattern of the screens, you have to shut down usually and go to a pattern which you have determined before. Here with this new device, you do have the great flexibility, without shutting down, of going on from pattern A to pattern B or whatever you want. This is a tremendous advantage, but it also introduces a new problem. When you go from pattern A to pattern B, how do you accomplish the changing? You want to be sure that in between two desired patterns, a transient pattern is not established which might be worse and might cause a preliminary surge. I wonder whether you have any experience along that line?

OVERALL:

Well, the program has been modified to look at the desired pattern and that area that has high pressure and to direct it to close those valves, so that we have overdrive, so to speak, going from one pattern to another. Now in these cases, we did execute the program with, say, a tip radial pattern to take us from that pattern to a hub radial pattern. So we could get in a problem depending upon the extent of the pattern. There could be some air jets in there that were being commanded to move in the wrong direction! The program logic has been modified; we execute a program always to take us in the right direction, so that we will be constantly decreasing the distortion on the engine when we are in a transient mode.

LOTH: (West Virginia University)

Is it possible to suppress the turbulence behind the screens by adding another screen between the ejector and the transfer?

OVERALL:

That is a consideration, yes. The turbulence problem sneaked up on us. We had looked at turbulence behind an active air jet, and it was fairly small, on the order of 2 percent of the inlet total pressure. But it appeared that we were not getting in this periphery zone, that if you get buried behind the active jets, you do not have a turbulence problem. A lot of people document turbulence for air jets by turning them all on, and it is really low. But when we get this interaction around the periphery of the active air jets, that is when we get high turbulence. But that is a consideration to use an attenuation screen.

SECTION IV

GENERAL DISCUSSION

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GENERAL DISCUSSION

Panelists

W. H. Heiser (Arnold Engineering Development Center; Panel Coordinator)
R. H. Petersen (NASA, Ames Research Center)
B. A. Reese (Purdue University)
R. F. Siewert (Naval Air Systems Command)
A. Silverstein (National Academy of Sciences)
H. von Ohain (Wright-Patterson Air Force Base)

S. N. B. MURTHY: (Purdue University)

We now come to the last part of this workshop, the panel discussion. Now, the panel discussion is, from various points of view, a very important part of the workshop. What we hope will happen is an assessment of the kind of basic research areas that need emphasis in the next five to ten years. Of course, immediately, the question can be asked, what kind of an answer do you want? Because, for any question, I suppose, we identify the man who asked the question and then give the answer that should satisfy him. I think, in this case, let us not worry about any particular organization, but let us just assess the subject matter and let us see what kind of basic research work should be undertaken in this area in the next five to ten years. And I think in the last two days, we have been talking about basically, again as I said in the introduction, computer experiments, physical experiments and large testing facilities. And I hope the discussions now will center somehow around those in relation to any grand ideas that might develop.

W. H. HEISER: (Arnold Engineering Development Center)

As the coordinator, I think my chief contribution is to make some attempt to get us finished in a reasonable time!

I understand that the panel discussion is a heavy part of the payload for Project SQUID and other research funding agencies because you folks are trying to look years down the line and shape your

investment strategy intelligently. I have tried to listen to the two days of presentations with as open a mind as possible and start all over again on the business of V/STOL, and I am also going to try to resist the temptation to summarize all the bits and pieces that I have seen. In fact I am going to make two general observations and come to a few conclusions which I hope are specific enough for the funding agencies, primarily Project SQUID.

Both of the general observations start off in the form of questions to myself. The first question is: what is really going on in the V/STOL business; what makes V/STOL different from what we have been doing before, if, in fact, anything? I have to say that this reminded me happily of an incident from my own past with the Air Force which does bear upon the issue. When I was first with the Air Force I was working with Dr. Lovelace on a study to see how composites could be used confidently in airplanes, and I was a rather junior and naive member of the team. It was at one of his meetings that it finally dawned on me that the engine was regarded by everyone else as a subsystem of the airplane. When everybody realized that that was just coming to my mind as a brand new concept, there was general laughter at my expense. I had never relegated the engine to the same category as wheels and drag chutes and radios and things like that, but apparently everybody else in the Air Force did. From that embarrassing moment, I come to today. I would say that what is really going on here is that the engine is less and less a subsystem as we move towards STOL and, more importantly, towards V/STOL. Instead, I think we have to regard it as closer and closer to a full partner with the airframe and everything else that flies that has to be integrated from scratch. In addition to some technology, which I will come to, this is going to require significant changes in attitudes, changes in attitudes on the part of the airframer and the engine-man and everybody else in between. I don't think this runs counter to the present trend, however, as you will see from a few observations about the development world. If you look out to Wright Field, for example, you see that the Air Force has reorganized its engine procurement business so that all of the engine work is being done under a single ASD organization which is a propulsion kind of system program office. That recognizes at least that the engine is not purely a subsystem in the old sense of the word, even if it is not quite the whole system itself, as it should not be. Also, the Air Force has decided to invest $437 million down at AEDC, where I come from, to build an aeropropulsion system test facility. The word that matters most in that title is system. The whole point of the massive facility is to test all the parts of the engine in a coordinated way in the most realistic environment that you can provide. So I think the trend is there and V/STOL is a continuation of that trend.

The second question I have asked myself is: what does all this mean to the research community and, in particular, to SQUID? I guess

that I should be a little sorry to say it, but it is clear in my mind that I have not seen any generically new engine problems in the two days that I have been here. They are all either the same as we had dealt with before or their extrapolations. It looks to me like business as usual, but more so in many ways. Perhaps what encourages us to go on to V/STOL is that there are no inherently new dangers that we have been able to identify. But that does not mean that there are not going to be areas in which there are very large technological leaps required, even if we can see the direction that we have to go. Therefore, I have made an approximate rank-ordering, from what I have heard, of the technological areas in terms of the size of the leap that, I believe, is going to be required to get, say, all the way to a V/STOL type of airplane. The first is the effect of the engine aerodynamics on the airframe, both the desired ones, such as the fountain effect, and the undesirable ones, such as the engines getting all mixed up in the flight controls in a first order of magnitude way as they never have before. The second, and I think Pratt and Whitney will be happy with this, is control complexity in the fullest sense of the word, namely integrating a control into the entire system and giving it adequate computational strength. The third is the question of safety of flight. I believe that never before has the engine been quite so pivotal to holding the airplane up because these cannot even glide. That will bring up questions of reliability which I think will be reminiscent of the space age. And fourth on my list are the effects of the airframe aerodynamics on the engine, which are, I take it, possibly an order of magnitude more significant than they have been in previous installations. I am talking here, of course, about inlet distortion and things of that sort. Finally, I have a four way tie for fifth place. It is obvious to me that component performance is going to be pushed because these engines are going to have to get the maximum thrust and fuel consumption out of the smallest possible space and weight. Structural durability will be pushed harder than before because the cycling of these engines per hour, or per whatever, is going to be more than we have seen before and perhaps even made worse, I am sorry to say, if the engine controls are turned over to a computer, because the pilot will not even know how many times the engine is being cycled as it comes down. The cost of ownership models and everything that goes with them and techniques of reducing cost of ownership will be utilized more than before. Also in a tie for fifth place are questions of noise, which have not only community importance but a military importance in keeping the pilot from going crazy and keeping the airplane from shaking apart.

By means of these two observations, I have come to a few conclusions that I hope will help in focusing the meager resources for research. If I were running a program, after seeing these two days of presentations, I would first look for some more novel and ingenious ideas and put the community to work trying to think of new ways to do the things we want to do. I believe they are out there, and I am

sure we have not seen them all. At this point I would rather see the ideas drive the application than have a number of specific applications drive the research program along. Secondly, I would invest my money in the foundations that are required for good analytical models, including improved phenomenology and suitable computer programming. Thirdly, I would look for the classical, relevant experiments to back up the phenomenology or even to provide new models. With that, I will turn the floor over to Ray Siewert.

R. F. SIEWERT: (Naval Air Systems Command)

As an aerodynamicist in this group of propulsion people, it is really great to see that you are now considering the engine as part of the control system! In 1968 I tried to convince Pratt and Whitney that in carrier airplanes the engine was indeed part of the control system,but they would not buy it. They would not change the engine to help the airplane fly better. I certainly agree with a lot of Bill Heiser's comments regarding where we go. I guess first on my list, from an aerodynamicist's standpoint, are the thrust-induced effects that Bill Hill talked about yesterday. Most of the methods that we have right now, including the Grumman methods, are based mainly on empirical data gathered from small models. So right here we have a problem because we don't really care what happens to the model as much as we care what happens to the airplane. And the need to correlate the model to full scale data is that we don't have a lot of we see as a pressing issue. The problem is we don't have a lot of full scale data. Now to give you a little insight into the full scale data, as you may know, the Navy entered into a program with the VFW in West Germany, about two years, ago to conduct a series of flight tests on the VAK-191B, which is a lift-plus-lift-cruise,jet-lift V/STOL high performance airplane. I have some data back at the office from some of those tests. It is interesting because we show excellent model-to-full-scale correlation, as well as with the theoretical methods, so long as we use the proper correlation coefficients. Model-to-full-scale is well correlated on one run and completely useless on the next run. Well, we are talking about a phenomenon here that, as we showed yesterday, is equivalent to about four percent to six percent of the net thrust available for the aircraft to lift-off, and we find that we can only measure thrust within ± five percent on the airplane. So there is a plea here to start working on thrust measuring instrumentation on the airplane if we are really going to be able to do any correlations in this area.

Another area is the reliability of power transmission systems. I am speaking here about mechanical systems. The Navy actually does operate one other S/STOL airplane besides the Harrier, which is called the X22A. This is a four ducted-propeller aircraft. It is a research aircraft in that it is a variable stability airplane wherein we can vary the control system characteristics, the frequencies, and the damping characterics about four axes. The principal

concern here is that it is a four-engine, four ducted-propeller airplane that runs on 11 gear boxes. With the exception of the four engine gear boxes being able to tolerate a possible single non-contaminate failure, failure in any of the other seven gear boxes is critical to the safety of flight. If we look at some of these mechanical powered transmission systems, the same problem is facing us. The Navy was involved in a flight test program on the CL84, which is a tilt-wing propeller airplane. We lost one of the test airplanes because the gear box failed Fortunately, it was in conventional flight. I say fortunately because, in conventional flight, the pilots were able to eject safely.

I talked a bit yesterday about control power. I think it is our responsibility in the flying-qualities-community to define the control power required for these airplanes. However, I think that we have to do some work with regard to obtaining the necessary control power without unduly penalizing the propulsion system. Work on augmenting reaction-control systems, improving nozzle designs for systems that operate at present at very high pressure ratios is badly needed. These are things, I think, that can and should be done.

Finally, I feel that we have to have a better basic understanding of such phenomena as turbulent mixing. Max Platzer from the Naval Postgraduate School had a little program with Rockwell where they were trying to enhance the mixing of primary jets in augmentors by adding swirl to the flow. It seemed like a reasonable thing to do. Low pressure ratio tests indicated that you could enhance the mixing quite a bit, but with the increase of pressure ratios to values that we expect to use in actual aircraft applications, the mixing dropped off considerably, which basically indicates that there is a lot about the phenomenon we don't know. Now whether we do it through computer analysis or testing, I think a lot of work is in order to get a better understanding of this phenomenon.

A. SILVERSTEIN: (National Academy of Science)

This whole area of STOL, V/STOL over the years has persisted with us as a very nasty problem. We are not the first group who have tried to figure out how to do a better job in this area. As I look back over 25 years of efforts of various groups to create V/STOLs and STOLs, outside of the helicopter which has been an outstanding success, there is very little that I can see that has been effective. I think it causes us to ask why. The answer, I think, is that no one airplane will do all of these different things that you might want it to do. If you want to hover for a long time, you cannot do it unless you go to something like a helicopter. Or if you want to go really fast, you can hover for a while, perhaps, and then, by some configuration changes to the aircraft, go fast. But somebody has got to decide at the beginning what it is you really want to do. I was

pleased by Murthy's original statement. He listed a number of things that this session was about, and one of the things he suggested was that maximum Mach numbers in the range of 0.6 were reasonable ones to examine. Now that is a great deal easier problem than if $M = 0.9$ or if supersonic speeds were suggested. What I think we have to do first, if we are ever going to be successful in this whole area, is to write a set of specifications for these airplanes that make sense.

I have been discouraged by the fact that there is not enough really creative thinking in this area of engine-aircraft integration. Engines of the jet and fan type may be applied flexibly. You can take them apart in pieces. The compressor set can be separated from the turbine set; if you care to, you can separate out the power turbine from the rest of the engine. Power can be transmitted with shafts or with gases. I don't think our approach is adequately imaginative when nacelle installations as large as the fuselage are used. The base drag could be very large if the flow separated on the rear of the nacelle. Base drag has reduced the performance of current airplanes more than anything else. Also, what is the interference effect of big nacelle bodies sticking up over the top of wings?

I think we must examine more carefully possibilities for enclosing engine systems in the aircraft wings and fuselage. There are all sorts of configuration possibilities. Some of these have already flown. We saw a picture here of an airplane flying in 1984 with distributed fans.

When an airplane flies, a wake exists behind it. In the wake, a part of the air that the airplane is flowing through is moving in the direction of the airplane and the other part is moving oppositely. The thrust pushes the air backwards and the drag pulls the air in the flight direction. What you would like to see in the wake behind the airplane is the air at rest (except for its vertical velocity). Then you would have minimum drag. Now how do we do this? I think some of the ideas that Goethert was discussing here today are very good. That is, put the jets out the trailing edge of the wing or at the trailing edge of bodies. The jets and boundary layers interact and tend to reduce the energy losses in the wake. These configurations need to be examined in much more detail. You can separate the engine into components and locate jets where they can counteract some of the wakes that are being created.

The conceptual part was the part I missed most in this meeting. I think we had a very good meeting and discussed many important aspects of the problem. In particular, I believe considerations of an integrated control system involving the propulsion system and the airframe is an absolute necessity. The safety considerations that were mentioned by Heiser are also extremely important. As an ancient aerodynamicist, my eye rejects the crazy looking configurations with enormous bodies sticking on the fuselage and above the wing. I would like to see something that is more appealing to the eye.

H. VON OHAIN: (Air Force Aero Propulsion Laboratory)

First of all, I was quite surprised about Bill Heiser's comment that it took him quite a while to find out that propulsion is a subsystem. Coming from the same lab as Bill, I had just the opposite experience, namely, to learn that the airplane is a subsystem of the propulsion system! Now, seriously, I believe Heiser has an excellent point when he commented that engine performance and operational characteristics become more conspicuous and important as we move from conventional aircraft toward STOL and V/STOL. This is well in line with Professor Murthy's initial comments on functional integration where the functions of both an airframe and an engine with very flexible characteristics (e.g., Variable Cycle Engine) are blended in such a manner that strong synergistic effects are produced.

As Abe Silverstein had pointed out, I too believe that new concepts as well as basic research are required. In basic research, "mixing" would be one topic of great importance. Specifically, studies of phenomena in "hypermixing," according to Dr. Brian Quinn, AFOSR, seem to be highly promising.

Now I would like to make some comments with respect to new concepts and ideas. A systematic investigation of new concepts by actual hardware programs is too expensive and time consuming. In this connection, it was very interesting to hear in this conference that analytical methods are being developed which will enable a relatively quick evaluation and comparison of new concepts. I would like to categorize new concepts in the following three groups:

1. Aircraft concepts where the V/STOL or STOL capability represents an added feature to an otherwise normal airplane. The added hardware to achieve the V or STOL mode is in operation only during the takeoff or landing phase. During normal flight operation, this added hardward constitutes added weight and drag penalties. Also, the propulsion system may be overpowered for cruise conditions as a consequence of the high power requirements for takeoff. This would result in a penalty of high fuel consumption in cruise.

2. Aircraft concepts where the V/STOL or STOL capability does require only negligible or no specific additional equipment. A typical STOL aircraft configuration of this type is the upper wing-surface blowing such as the Boeing AMST YC-14. For such an aircraft type, added weight and drag penalties for achieving the STOL capabilities are comparatively small.

3. A somewhat futuristic STOL or V/STOL aircraft category is the following: the equipment needed for the "V" or "STOL" mode is also useful in cruise conditions for obtaining lift and drag advantages over conventional aircraft systems. Such aircraft types are

not as yet in existence; they would be examples of "total functional integration" of aircraft and propulsion system and would potentially offer overall advantages over conventional systems without paying a penalty for the V/STOL or STOL capabilities.

In this connection, new concepts or methods of propulsion by boundary-layer acceleration may become important. The basic principle of abolishing the airframe wake by the propulsion system was mentioned by Abe Silverstein. The Navy makes use of this principle in ships by placing the propellers in the wake of the ship. Various attempts to utilize this method for aircraft have not been successful so far, probably due to the very complex structure of the wake generated by fuselage, wings, and stabilizers.

I believe that in the future, through research and new concepts, fully integrated aircraft systems will materialize which will bring us very substantial improvements over current systems, with respect to both operational and performance characteristics.

A. SILVERSTEIN:

I just want to mention, with respect to the high propulsion efficiency, that at one time we took the Akron airship model and put a propeller on the rear of the body. We measured 115 percent propulsion efficiency.

R. H. PETERSEN:

I am also an aerodynamicist like Ray Siewert. If Dr. Hans Mark were here, we would have a physicist on the panel. He would introduce a very strong pitch for computational fluid dynamics as the wave of the future.

I will take a little different track, since I operate a number of large wind tunnels at the Ames Research Center. We are pushing very hard to use the new computational capabilities in conjunction with our experimental capabilities to produce new kinds of design [illegible]

[illegible]

information rather than Navier-Stokes solutions that give you the flow field but no design information. So that is one of our thrusts.

The other one that we are quite excited about, that I would like to recommend that other people look into also, is what we call hybrid experimental and computational techniques. With all of the new experimental measuring techniques and sensors that we are getting, specifically the laser Doppler velocimeters and laser holography, we have tremendous new capability to map flow fields rather quickly and very accurately in the tunnel. The LDV's give us nonintrusive velocity measurements in any direction we want and very good statistical information about the statistical variation in air flow at any point where we want it. We see a real opportunity to combine these with the computational techniques, use the experiment to find the things that are toughest to find in the computation, and generate a lot more design power that way. Actually we are working at Ames mostly in the aerodynamic field, say over wings. I tried to think of an example and perhaps the one Leroy Presley described -- trying to couple the flow field calculations on the nose and the inlet - is a good example. It is not too difficult to think of going in and mapping with an LDV the velocity flow field directly ahead of the inlet and then using a computational technique to compute the result of that inlet flow field. Within the inlet, you would get all of the advantages of computation which gives you complete flow field information, and you wouldn't have to go into the inlet experimentally and measure all that information. That presumes that once you knew the boundary conditions ahead of the inlet, you could make an accurate computation through the inlet. I would like to put in a pitch for a lot harder look at some of these hybrid techniques in the next few years.

B. A. REESE: (Purdue University)

Being last, all the good ideas have already been used up. However, I would mention a report that I heard the other day from the Dean of Students at Purdue, who reads different technical journals than I do and probably you do. She had one in which they had reported a student essay on Socrates. And it seemed the student had read that Socrates lived in ancient Greece and spent every day, all day long, going around giving everybody advice, so they poisoned him. And I think you run a hazard here as you give advice!

I think the words "engine-airframe integration" mean something different to everybody. Each of the speakers has illustrated that by his view of the engines and his view of their problems. A lot of the problems or concerns are engine-specific and mission-sensitive; these I believe are difficult research topics. That does not mean that there are not vital problems; it just means that they are different for the academic or pure research individuals to work on. An example of the "engine-specific" was the report of the difficulty in finding a good place to put the accessories on a given engine, and it is not clear how that problem is done in the abstract. It was mentioned

that engines are very sensitive to the application, and I think they are becoming more sensitive as we "tune" the engine to the mission. At a review of Air Force engine problems, it was found that the TF30-P-100 had, to that point in time, a reasonably good performance record. The Air Force mission changed, and the engine performance became satisfactory. When the mission profile was examined, it was found that engine cycles had increased to the point that engine time would be significantly reduced; the engine was designed to fail at the time it was failing if used in that mission. It should not have been a surprise, but apparently it was.

In addition to the engine specific problems, it is necessary to consider the problems that have been brought up in the last two days which are more research oriented, for example, the aerodynamic questions that Welliver discussed, predicting the flow around the aircraft and the distortion of the inlet. I remain convinced that an investigation of the form of inlet distortion encountered in the engine does need investigation. This is in addition to measurement of the total pressure distortion.

Another important integration problem is associated with the flow over the aft end of the aircraft. As Dr. Silverstein noted, it has been one of the things that has hurt us very badly in our aircraft, and is true for both military and commercial aircraft. Research is being conducted on predicting separation. I think we need to work harder (and longer) on the interaction of the jets and the flow over the aircraft. Variable cycle engines will obviously help in the aft end flow because they can hold the engine mass flow constant over much wider range of engine operating conditions. Work on the separation problem includes not only the airframe-engine integration, but also separtion in the engine components.

Research is required on the near and far field ground-jet interactions. Purdue worked on the problem several years ago but are no longer doing research in that area. However, it appears to me that there needs to be a great deal more work on that subject. I am sure that Mr. Hill was aware of the problems of cross flows, the wind problem. He did not mention it as a special problem, but we found it to be important. I suspect he included those effects in the problems of instability with his wall jets, such-down jets, and the fountain effects. Jet flap-like flow, ejectors and jet mixing, hot gas re-ingestion, all in this highly turbulent and complex flow, require more understanding.

A listing of fluid mechanics research problems, made after a similar meeting in 1975 by Professor Jim Skifstad of Purdue, does not seem to have changed very much, and it is presented for your consideration.

"1. Complex Duct Flows. There is evidently a need for even the most elementary type of information related to flows in the types of duct systems employed in these aircraft. Standard head loss

correlations are not applicable or of sufficient accuracy. Sensitivity of the flows to inlet profiles, Mach number, duct geometry, types of branching configurations, etc., and to the turbulence level could be examined to some extent, independent of specific airframe considerations. Three-dimensional, viscous flow solutions are evidently being explored (hyperelliptic, curved ducts), but there remain the ogres of turbulence models and separation which compromise their merit.

2. Jet Aerodynamics. There remains rather fundamental research to be conducted on jets subject to aerodynamic interactions (cross flow, etc.). In spite of several investigations supported in the recent past, little detailed understanding of these complex flows persists. The more complex treatments are evidently not used, presumably because they do not offer sufficient improvement over the simpler models (line singularities). The structure of these jets remains largely an open question (detailed properties, including turbulence attributes), and even the entrainment rates are not known accurately.

While the current turbulence theories are probably not adequate to tackle the 3-dimensional shear flow involved in these problems, there could be reasonable attempts to build a data base for some of the simpler ones. The task has been started but seems to be faltering for lack of funding or, possibly, of interest on the part of domestic researchers.

3. Confined Jet Mixing. This topic has found many applications in propulsion and other flow systems. Ejector technology remains rather an art despite an enormous "bone-pile" of literature on the subject. The difficulty, of course, lies rooted in turbulence. Progress has been made, of course, in predictive capability for the simpler systems, with or without combustion. But the most effective ones introduce essential 3-D effects (angled jets, vortex generators, acoustic interactions, etc.) to enhance fast mixing. Some fundamental problems could be identified here for research. But most of the work will probably remain semi-empirical until turbulence theory offers additional capability for 3-D shear flows.

4. Wall Jets, Coanda Effect, Fountains, Hot Gas Ingestion. These problems remain largely open, although some analytical and experimental research continues to be reported. Semi-empirical theories and the like seem to be of value to the designer, at least where the flows are stable. Unsteady phenomena and sensitivity to small configurational changes plague the aircraft applications and weaken the merit of simple systems investigated in research programs for design purposes.

5. Diffuser Flows. These seem highly related to specific configurations. There may be studies of a fundamental character which

could be defined for this purpose (selected inlet profiles, turbulence levels, geometry, etc.), but they are of limited generality

6. External Aerodynamics. This problem, during transition flight, must offer a significant challenge to the theorist in view of the essential 3-D, nonlinear character of the problem. There was little mention of it, insofar as tunnel testing and modeling on the basis of decoupled jet/airfoil flows seem adequate. Perhaps the level of interest and the precision of prediction methods may be enhanced when (and if) a new aircraft requirement and its funding arrive. Much reliance on empiricism presumably will remain a part of the problem, the large computer programs in existence notwithstanding.

In summary, it is probably useful to point out that many of the problems involved in these aircraft involve 3-dimensional, turbulent shear flows, and that some of the effects considered important (e.g., entrainment) are to be regarded as second order properties of these flows. The accuracy of the theory attainable for prediction purposes will be conditioned primarily by empirical correlations rather than by theories based on first principles for the foreseeable future, barring some breakthrough in turbulence theory. This situation calls for perceptive, well-defined experiments, coupled as closely as possible to what can be done theoretically.

Airplane designers will remain airplane designers in the classical tradition, it appears, particularly with respect to these V/STOL systems. Where the dollars and schedules are involved, there will be a fairly heavy reliance on tunnel testing. The airplanes are likely to fly long before we can fully predict their behavior."

W. H. HEISER:

Well, the panel has lived according to its rules, and now it is up to you. I ask for any questions you have, and I know there are people out there who have strong opinions about things who have not had an opportunity to voice them in the last two days. I welcome them to take their turn now.

J. LOTH: (West Virginia University)

I appreciated the comment about the importance of trying to make the axial velocity field behind the airplane more uniform so as to reduce the drag. I would like to point out that the induced drag is due to the low pressure region created by the radial pressure gradient associated with the tangential velocities in the wing tip vortices. So if the wing tip blowing could be so used as to eliminate the tangential velocities in the wing tip vortex, one might be able to combat the induced drag component.

H. VON OHAIN:

You are saying we could do something to help the induced drag for a given aspect ratio, physical aspect ratio, to be reduced. Is not that what the winglets are doing?

J. LOTH:

True, I was thinking more of wing tip blowing in a whirl manner so as to counteract the wing tip vortices.

H. VON OHAIN:

Oh, I did not see your point. Thank you.

A. SILVERSTEIN:

I think the fastest way to reduce the induced drag is to grab more air, that is, increase the wing span. The cruising part of the flight is the part where the induced drag will be important and when you need span.

L.G. NAPOLITANO: (University of Naples)

I think the coordinator in his closing remarks pointed out several things. I am speaking also of the ideas he referred to as second and third. He sees that fundamental work is required, foundations as he mentioned. My knowledge about foundations is that they are required in order to build up good models. And then he also mentioned that, equally important, there are the classical, crucial experiments needed to check this out. Well, in this connection, I would like to offer a comment and then ask some questions. I think that one would also be interested in experiments which would clearly and unequivically invalidate classical or conventional models. I say this, it looks like a joke or a paradox, but it is what I call the cost effectiveness of complexity. We have seen here that there are two groups. The one motivated, with different cross strains of course, and then this person who has to do things as simply as possible and as quickly as possible and maybe, has to have done it yesterday. And the other is the theoretically prone or fundamentally prone person who is interested, rightly, in trying to get this complicated phenomenon as complicated as possible in order to picture the reality. So there is a cost to both trends, the cost of one who wants to do things too simply and the cost of one who wants to do things too complexly. There is still a bridge to be built between them and also true trade-off. I have not seen any of those things being spoken about here. I can quote some examples. It was mentioned that turbulent mixing is important. Now, everybody knows that from Prandtl's intuition to now, it has been a long way in modeling turbulence, and there are still

longer ways to go. But you also know that the models can be very, very complicated and one key question is: when do you step up in complexity to get closer to reality? This is a big question because maybe failure to describe reality is not crucial to turbulence modeling, but it may be in something else. That is why I say it may be an important point. That is my comment. My questions now. As the organizer and director of Project SQUID, Professor Murthy pointed out that these excerises are useful in developing new ideas, not only for the management of research activity. I would like to have some indoctrination myself. In this respect, it would have been surprising if we did not conclude that we need more fundamental ideas and more experiments to check those. The point however is, what type of fundamental work or, if you like, what kind of priorities is this particular application demanding? That is, what are the areas in which specific investigations are required? One area was mentioned, turbulent mixing, and another one was also mentioned, but as an application, not as a need, a breakthrough perhaps. And yesterday I think, the problem was mentioned, namely the combination of experiments and computation, whether (and if I understood you properly, then I would agree with that) the experiments should give you initial and/or boundary conditions and then you can complete the loop, also get the exit condition and then you can sort of feedback and see who is wrong, whether your model of calculation or the experiments. But then the problem arises, would your input be of the same accuracy as your computing accuracy? There is all this interphasing problem. And the third point I would raise is whether, in hybrid or interacting processes, I would like to see human beings somewhere there. Because it must not be something push-button where you would push the button and, whether it would be only the wind tunnel or the wind tunnel and the computer talking together, that would be enough. We would like to have hybrid and interacting experiments, and at this point I would like to ask for instance: do you think that simple, nondissipative three-dimensional flow fields are of any interest? What is their priority? You killed, and I agree with you, the simple exercise in Navier-Stokes formalism, which can be a hobby or a love, but how about going to the other extreme and just throwing everything out? Would that be of any priority interest for you people?

W. H. HEISER:

I don't believe anybody can answer all those questions. Let me talk about a few of them, and I think I can come closest to answering them by just sharing some of my philosophy and observations with you. First of all, the notion of some experiment or an analysis that completely invalidates the need for something, I think, is a very useful concept to carry around. I believe that to be the province of companies and of government laboratories, to determine, even though they cannot get the performance of something precisely, when they can bracket something well enough to convince themselves that it is worth going ahead and leaving the exact answer to some later analysis or

experiment. I believe that is already under the control of the right partners in the game. Now as for where modeling needs to be improved, my concept is that the computing machine has so far outrun us, that the computational plenty is so enormous and is getting ahead of us at such a tremendous rate, that it is the physical models that are inadequate. If they are not inadequate now, they will certainly be inadequate before anything we can do will be available to be used. I give you one, I am sure, very fine example of where we need to improve things. That is in our ability to predict transitional boundary layers and the nature of turbulent boundary layers, particularly in transonic flow regions and where three-dimensional effects are to be included. The most often stated example of where that lack of knowledge causes us to get into trouble is in the base drag problem. Over and over again, the base drag of airplanes is excessive in the transonic region because all of the above factors come into play. The flow is compressible, there are shocklets, it is three-dimensional, and it is transitional. It will be extremely difficult to handle those problems on any _ad hoc_ basis until we have a model that incorporates all of the basic phenomena that are at work. I would consider it to be a very fertile area for people carrying out classical work. By classical, I mean that iterative process where somebody does experiments to supply models, and the models are extended with the help of existing equations and well-understood phenomena into analytical tools that cover a large range of aerodynamics, and then that is verified by an experiment in which all of the phenomena are known to be at work. I feel that the aerodynamicists are now behind the power curve, as it were, with respect to the computing machine, and we are going to have to do something to improve on that situation.

B. A. REESE:

In the context of the questions that were mentioned in regard to the vertical part of this short-haul transport we are discussing, there are some things which are very important, and one of those is the thrust to weight ratio. It is extremely important to increase it so decent payloads can be achieved in these vehicles. The mission constraints were discussed, e.g., you don't want to hover if you want to do the things that von Ohain was talking about; it is necessary to get up and get going if you are going to have these multiple capabilities.

The other item that was discussed in some detail, and I think there is still reason for research, is the ground effect of these vertical take-off systems. I think those are very specific kinds of research projects that need more research.

R. H. PETERSEN:

With regard to the computational area, I would say I wholeheartedly agree on the interactive nature of both the hybrid methods and any of the computational methods. I personally see a process of

using whatever method is the simplest way of computing flow in that region, which means there is plenty of place for inviscid, linearized solutions and plenty of place for boundary layer theory, and there are some places where you absolutely have to have Navier-Stokes equations. So the trick is to use the aerodynamicist to figure out where to patch things together. I would add that, if we are stuck and we cannot compute accurately through an area, we should use the new experimental techniques like LDV's and go in and find out what is going on.

A. SILVERSTEIN:

If you visualize the landing conditions for some of these aircraft that we saw pictures of during the presentations, you really wonder whether this flow system can be calculated.

R. H. PETERSEN:

Somewhat facetiously, I am tempted, when I see the picture of a V/STOL aircraft with its fountains and so forth, to give my computational friends that problem to work as a full Navier-Stokes solution. I think that would keep them busy long past my retirement.

W. H. HEISER:

One thing that has not been discussed in detail here is whether the ground test facilities will be up to testing these V/STOL configurations with the jets running. The answer today is frankly no, and, somewhere along the way, somebody is going to have to consider that. As for Abe Silverstein's last remark, it may be that we cannot do many of these calculations properly today, but, the way things are going, people are going to be doing the calculations whether the models are right or not. It would therefore seem wise to provide them with the right phenomenological understanding.

A. D. WELLIVER: (Boeing Military Airplane Division)

Abe Silverstein finally reminded me of some key points I would like to make. One of the issues here is, first of all, as somebody commented on the SST, that we know, in my opinion, a lot more about how to design the aerodynamics of an SST than we do about the aerodynamics of a V/STOL airplane. That is the first point. We know how to build a good efficient supersonic transport. Secondly, when I was talking about the problems of fluid mechanics, I was thinking seriously about this fact, and it was the reason I emphasized the fluid mechanics area. There are other areas I am interested in besides this one, but the reason I brought it up was that I think that this is a very, very difficult area that we really do not know much about. All you have to do is go look at the McDonald-Douglas chart on the huge arrays of V/STOL airplanes we have tried already, none of which has worked. And so I believe that America and the technical community of the world know how to build supersonic airplanes and get at it, but it is

pretty obvious that we don't know how to build these vertical take-off type machines and then have the same machine fly in forward flight efficiently. Now I come back to a comment that Abe Silverstein also made, which was that everything we talked about here was technical in nature or basic, but he did not see a lot of good configurations. Now the concern that I have is that in this V/STOL area, where you are trying to hang the airplane on a couple of posts scattered around the CG, one of the very difficult things is to assess, early on, which of these airplanes makes sense. It is as we flew the augmentor wing airplane, you know, which is probably the limit of stretching the gas out along the wing. But the point of fact is, to really find out how an airplane like that worked, you had to put the whole thing together, all kinds of rig tests, and all that, and that is what happened in all those airplanes that McDonald-Douglas is pointing to. I think one of the key points is, we really need to know how to get down to the facts of whether these airplanes, plus or minus ten percent or even twenty percent or something, are resonable, and that is what we really don't know, in my opinion.

W. H. HEISER:

I think we have come to the end of the panel discussion. Let me put in a closing note. I have enjoyed being here in Annapolis enormously, much more than I imagined before I came, and I believe that you folks have too. I wanted to point out something I ran into today that shows you how well the Navy has its ducks in a row, as they are reputed to do. We stopped off at the tomb of John Paul Jones and found that 199 years ago he said, "I wish to have no connection with any ship that does not sail fast, for I intend to go in harm's way." I feel he must have had our trip home in mind. Thank you and bon voyage.

SECTION V

SUMMARY AND RECOMMENDATIONS

SUMMARY AND RECOMMENDATIONS

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1. SUMMARY

The objective of the Workshop was an assessment of the basic research needs in the area of engine-airframe integration pertaining to short-haul aircraft. Such aircraft have important civil and military applications. While actual development and use of civil and military aircraft would be based on somewhat different considerations, it became clear early in the organization of the Workshop that the basic research needs in the development of short haul aircraft, more especially the V/STOL systems, would very largely be common in the two applications. It was also felt that the Workshop should concentrate on V/STOL systems although conventional systems were also of interest for short-haul applications. The Workshop also included a paper on the problems of engine-airframe integration in helicopters, although conventional definitions of short haul aircraft do not include helicopters, once again to emphasize the many common, basic problems needing attention by the research and development community.

Developments in V/STOL and related technology have been rather uneven over the years. There is still a considerable debate, even in the Navy programs, as to requirement and feasility of V/STOL systems. The Naval Air Systems Command has examined, according to published reports, the V/STOL developments in relation to the following.

(a) dispersal of air assets over a broad geographic range;
(b) reducing reliance on relatively few ships and large permanent bases; and
(c) reducing the cost of maintaining an adequate air arm.

However, it is recognized that economics and the philosophy behind overall strategy for any civil or military effort will dictate the emphasis to be placed on any one branch of aircraft technology. Economics should, of course, include the cost of research and development, first costs and operational costs.

There is considerable agreement that current levels of developments in the following areas have the potential for adaptation to meet the requirements of V/STOL systems, although not immediate feasibility.

(i) propulsion technology and flight dynamics,
(ii) airframes,
(iii) avionics and integrated control and
(iv) reliability and life cycle costs.

However, it is felt that considerable basic research is required in a number of areas before large scale development, leading to procurement of aircraft, can be envisioned.

The V/STOL systems should be understood here to refer to aircraft that always have a VTOL capability unless otherwise stated. While supersonic V/STOL aircraft was mentioned in the Workshop, the principal interest was in subsonic flight.

The Workshop contributions have been divided into four sections, namely (i) system requirements, (ii) system studies, (iii) special problems and (iv) general discussion. They cover the following subjects.

1. Overall system design for V/STOL
   (Siewert, Kishline, Acurio, Denning, Beam, Welliver)

2. Examples of V/STOL development.
   (Kishline, Denning, Beam, Acurio)

3. Propulsion systems
   (Siewert, Allen, Kishline, Acurio, Denning, Beam)

4. Flight/propulsion control systems
   (Siewert, Kishline, Acurio, Beam, Welliver, Miller)

5. Fluid mechanical aspects of integration
   (Welliver, Presley, Putnam, Ryle, Goethert, Brimelow, Hill)

6. Hybrid experimentation and testing
   (Welliver, Presley, Overall)

The names under each subject appear, for convenience, in the order of the appearance of their papers in this volume. The General

Discussion, of course, deals with each of the above subjects and, in addition, raises some important questions related to the philosophy of approach that may be necessary in the final evolution of this technology.

It may be pointed out here that the proceedings of two other recent conferences are very valuable in obtaining a full appreciation of the needs in basic research in V/STOL technology.

1. The Naval Airsystems Command Conference on Jet V/STOL Propulsion Aerodynamics held on July 28-31, 1975 at Washington, D.C. (Refs. 1 and 2).

2. The AIAA/NASA Ames V/STOL Conference held on June 6-8, 1977 at Palo Alto, California, (Ref. 3).

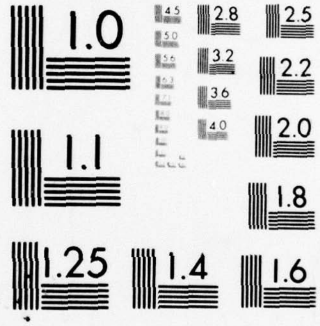
1.1. Overall System Design

The most crucial aspect of V/STOL system design, including helicopter design, is the integration of propulsion, aircraft and control. Such an integration should be generically and functionally established. While the power plant, structure and flight control will remain three major considerations in the design of a flight vehicle, the V/STOL system can be optimized only by taking the interaction between the three fully into account. For example, stability, balance, control and near-ground operation have to be examined together and that requires a total integration of the system.

A V/STOL system should function adequately in the take-off mode as well as in the cruise mode. One has to consider, therefore, four operational aspects of V/STOL flight in the design of the system, namely (i) VTOL and hover, (ii) transition to forward flight, (iii) impact of VTOL requirements on conventional flight and (iv) combining the desirable requirements of VTOL and cruise requirements in one system while providing useful payload capability.

The performance of a V/STOL system should be assessed by giving adequate recognition to its VTOL capability. That capability should be recognized as the principal demand in the specification of the system. However, it is not enough to emphasize VTOL capability without considering range, speed and maneuver requirements.

In concept, V/STOL capability may be looked at in three ways: (i) as an added requirement to conventional aircraft capability and therefore obtained through an additional subsystem; (ii) as a further step in the development of conventional aircraft capability and therefore obtained through air flow control; and (iii) as a requirement to be met through <u>total functional integration</u> (in Dr. von Ohain's [illegible]



MICROCOPY RESOLUTION TEST CHART
NATIONAL BUREAU OF STANDARDS-1963-A

that could extend as far as total elimination of aircraft wake and other methods of reducing various forms of drag and of increasing propulsion efficiency (Sir Frant Whittle's ideas expressed informally during this Workshop).

As we move progressively towards the third option or level of sophistication, the nature of system studies itself will change in the areas of testing as well as design calculations, for example, the calculation of the flow field about the system in ground proximity and in transition. In both of those areas a variety of uncertainties will have to be accommodated. Thus one, in fact, finally faces the question of what uncertainty is acceptable in regard to a chosen aspect of performance or operational capability.

1.2. V/STOL Developments

The development and testing of YC-14 (with the upper surface blown flap) and of YC-15 (with externally blown flap) were discussed (Kishline) in some substantial detail at the Workshop. The four principal features of the AMST which relate directly to integration are the super-critical wing, boundary layer control, blown flap, confluent-flow exhaust nozzle, thrust reversers and digital flight control. The engine is a high bypass turbofan. The philosophy in evolving the system and managing its development was also discussed. It is important to note here that this development was based on the use of existing technology and in many cases existing subsystems.

In the area of military applications of short-haul aircraft, a short discussion was presented (Denning) on a method of approach to the design of such aircraft using as an example the U.K. government sponsored variable-pitch fan demonstration program which utilized the M455D-02 engine. From the point of view of thrust management and reducing frontal area, the use of separate lift engines was compared with the use of cross-shafting.

A brief mention was also made of the flight trials on the VAK-191B in West Germany, which is a jet lift airplane (Siewert) and the early NASA studies on the Navy lift-fan ASW airplane.

Another development discussed in broad terms pertained to the Navy Type A V/STOL requirement. A shaft-coupled propulsion system was used as an example to illustrate the nature of development problems. Thrust management and reliability, especially under engine-out condition, were shown as major considerations.

In the application of short-haul aircraft for civil transport, the importance of operational economics was made the basis of a discussion (Denning) of the choice of power plant and its installation.

Finally, in the area of military helicopters, in addition to several recent development programs, the basic needs in engine-airframe integration in this class of machines were discussed (Acurio, Fig. 10). One of the critical areas of development obviously pertains to the low speed hovering condition: the structural-fluid dynamical coupling under dynamic conditions and the effect of download. The Army Heavy Lift Helicopter, in particular, the XT701 engine and its advanced version, were discussed briefly as representing current helicopter studies (Beam).

1.3. Propulsion Systems

The requirements of the propulsion system in a V/STOL aircraft are obviously complex. Some features of the propulsion system that were regarded as particularly significant are the following: (1) disc loading or thrust per unit frontal area, (2) fuel consumption, (3) thrust per unit installed weight, (4) noise generation, (5) first cost and direct operating cost of engines in relation to size, thrust and payload, (6) power sharing and thrust management, (7) variables in engine geometry, (8) overall lifting efficiency and (9) power for transition from hover to cruise.

From the point of view of installation of the propulsion system, several factors were pointed out as important: (1) heat loading, (2) engine face distortion, (3) ground effect, (4) noise footprint, (5) thrust augmentation, (6) thrust vectoring, (7) thrust reversal, (8) installation losses, (9) balance and stability and (10) impact on control system requirements.

The special aspects of helicopter installation were described in detail (Acurio).

The thermodynamic cycle parameters will need further examination in view of propulsion and installation requirements, for example the air flow path in the gas generator and thrust generator, the pressure ratio and the turbine inlet temperature.

1.4. Flight/Propulsion Control Systems

The control system for V/STOL aircraft needs to be an integrated flight and propulsion control system and therefore cannot be separated from the aircraft and the engine in integration studies. One of the basic questions in evolving a control system is "how much control?" This can only be answered based on the "extent" of integration.

The YC-14 and YC-15 control systems were evolved from the point

of view of operations requirements (Kishline). The impact of introducing full authority electronic control on the integration problems of helicopters was shown to be substantial (Acurio). Fly-by-wire and control configured vehicles were described as demonstrating the advantages of electronic flight controls while pointing to the problems of complexity, cost and reliability (Welliver).

The various aspects of the evolution of future control system technology were dealt with thoroughly from the point of view of systematically improving reliability through a methodological approach to incorporating applied technology (Emerson and Miller).

2. Recommendations

The outcome of the Workshop may be summarized under the following.

1. Need for conceptual changes
2. Research in thermo-fluid mechanics.

2.1. Need for Conceptual Changes

(1) Integration of the aircraft, power plant and control system is the most urgent need in a V/STOL system which can only be designed on a functional basis.

(2) The development of the methods of obtaining vertical take-off and landing, transition from hover to cruise and meeting the range, speed and payload requirements should be pursued on a unified basis.

(3) The traditional approaches to the problems of weight, drag and propulsive efficiency must be drastically altered so that the system can be developed in a unified manner.

(4) Similarly, flight and propulsion controls should be unified for each critical range of operation.

(5) A fundamental change is required in the method of conducting and utilizing analysis, experimental simulation and tests by adopting hybrid techniques of advancing the generation and synthesis of information in each.

2.2. Research in Thermo-Fluid Dynamics

The subject of basic research in thermo-fluid mechanics became in many ways the central theme of the Workshop, except for research in controls which was not, however, discussed in depth at the Workshop.

The major areas that were identified as requiring basic research studies are the following.

(1) Thermodynamics of variable cycle and variable geometry engines.
(2) Internal complex flows.
(3) External complex flows.
(4) Noise generation and footprint.
(5) Component performance under dynamic inlet and outlet conditions, in particular the fan with different types of shrouds and ducts.

(6) Angle of attack and maldistribution at engine inlet and fan and compressor inlet.

A word of explanation may be useful in regard to the so called complex flows. Complexity should be understood as arising out of one or more of the following: (1) geometry, for example, in internal flows, (2) interaction, for example, between a wing and a body, an inlet and a body, a jet and ground, a nozzle or augmentor and a surrounding (base) or locating body, a rotative or curved flow and a sheared flow, (3) turbulence and its effect on mixing and entrainment, and (4) distortion in entry flow. Under such conditions, there are two options available to the designer: (1) correlation of test data on the basis of generally simple analysis and (2) undertaking hybrid analytical-computational-experimental studies. The latter obviously is the more universal option, both from the point of view of the nature of problems and from that of establishing solutions to classes of problems.

Finally, concerning the subject of angle of attack and maldistribution, it is clearly insufficient to correlate test data in terms of simple distortion indices, for example, one based on total pressure distortion. The designer needs information regarding velocity distribution changes axially, radially and tangentially at all times for the onset of a given maldistribution at the inlet. This problem, when examined from the integration point of view, requires taking into account also the inlet vortex and the inlet-induced lift and drag forces on the airplane.

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