NASA SP-292

VEHICLE TECHNOLOGY FOR CIVIL AVIATION

The Seventies and Beyond

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A conference held at
LANGLEY RESEARCH CENTER
Hampton, Virginia
November 2-4, 1971

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

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Prepared by Langley Research Center

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PREFACE

The proceedings of the NASA Conference "Vehicle Technology for Civil Aviation – The Seventies and Beyond" held at Langley Research Center, Hampton, Virginia, on November 2-4, 1971, are reported in this NASA Special Publication.

The purpose of the conference was to emphasize what can be expected in the future by highlighting promising avenues of research that offer the potential of improving both current and future civil aircraft. The improvements were considered in the context of how they can change the current civil aviation picture by providing the aircraft required to satisfy the goals of the future. The following topics were covered in this unclassified conference:

(1) Aerodynamics
(2) Propulsion
(3) Structures and Materials
(4) Operational Aspects
(5) Technology Application

Contributions to this compilation were made by representatives from Ames, Flight, Langley, and Lewis Research Centers.

A non-NASA Panel Review was held at the conclusion of the conference. The deliberations of this panel are not included in this compilation. The panel was moderated by Mr. Oran W. Nicks, Deputy Director, Langley Research Center, and comprised the following six members:

Brigadier General Gustav E. Lundquist
Associate Administrator for Engineering and Development
Federal Aviation Administration

Mr. Robert C. Collins
Vice President – Engineering
United Air Lines, Inc.

Mr. Richard E. Black
Director, Advanced Design
McDonnell Douglas Corporation
Douglas Aircraft Company

Mr. William H. Sens
Chief Engineer – Advanced Gas Turbine Engines
Pratt & Whitney Aircraft Division
United Aircraft Corporation

Mr. Louis Achitoff
Chief, Aviation Technical Services
The Port of New York Authority

Mr. Calvin F. Wilson, Jr.
Chief of Aerodynamics, Structures and Flight Test
Piper Aircraft Corporation

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STRUCTURES AND MATERIALS

Session Chairman: Richard R. Heldenfels, Assistant Director for Structures, Langley Research Center

OPERATIONAL ASPECTS

Session Chairman: De E. Beeler, Deputy Director, Flight Research Center

TECHNOLOGY APPLICATION

Session Chairman: Oran W. Nicks, Deputy Director, Langley Research Center

VEHICLE TECHNOLOGY FOR CIVIL AVIATION: THE SEVENTIES AND BEYOND

KEYNOTE ADDRESS

By E. M. Cortright
Director, Langley Research Center

This conference on vehicle technology for civil aviation in the next decade and beyond represents somewhat of a departure from past NASA aeronautical conferences. For one thing, the emphasis on civil aviation is new. In the past, military aviation has paced the technology with civil aviation as the beneficiary. Today, however, we find civil aviation pressing technology heavily in its own right. Civil aircraft have their own unique set of technical and operational problems in flight regimes that were once the exclusive domain of the military. This trend may be expected to continue. And since the economy of this country is so dependent on the quality of our civil aircraft, there is a new urgency to look to the future of our commercial aircraft and transportation industries. This conference acknowledges that fact.

We are also deviating from past practice to some extent by placing the emphasis on future trends and opportunities rather than the known realities of experimental data. This caused us to reach a bit, and it required considerable homework to assess vehicle technology in the light of the total air transportation system. In this regard, I would like to acknowledge the assistance of DOT, FAA, aircraft-company, and airline advisors. A special panel of representatives of these groups will have an opportunity to critique our efforts at the close of this conference. What we are seeking is a clearer understanding of the best course to follow in civil aeronautics, and the most effective working relationship to realize future opportunities.

The growth of air transportation is one of the most remarkable technological developments of the twentieth century. It has provided almost unlimited mobility to hundreds of millions of people all over the world. In so doing, it has opened up new vistas of economic development and has offered the promise that one day nearly everyone will have his chance to see the world. This growth is highlighted in figure 1, where the Western World air-traffic forecast indicates a quadrupling of travel by 1985 – if we can overcome current technological constraints on this growth. The projections for the growth of air freight are equally impressive, and many feel that this will constitute an even larger business opportunity than the transportation of people. If this projected growth comes to pass, a vast market for civil aircraft will emerge, as shown in figure 2. While I do not place too much stock in the precision of these estimates, several significant observations can be made. A total sales potential of over $100 billion by 1985 is indicated. About 60 percent will be for aircraft not yet certificated. These new opportunities are being vigorously pursued by other countries – with the Concorde and the twin-jet airbus nearing certification. If we are to realize this potential growth and retain our share of this market, we must look to our technology.

The two main technological constraints on the growth of air transportation are noise and congestion. As in the case of other pollutants, public awareness of aircraft noise is bringing about stringent measures to curtail and control it. Until noise abatement is accomplished, airlines will find their operations increasingly restricted, and aircraft manufacturers will feel the effect in a loss of sales. Fortunately, we can see our way through the worst of the noise problem. Although some of the noise reduction will come with improved flight-path control and steep approaches, the main requirement is for quiet propulsion systems. Figure 3 shows what we can expect in the next 15 years. The approach noise level of first-generation jets can be reduced by retrofitting them with acoustically treated nacelles. Unfortunately, this is very costly. The second-generation wide-bodied jets are meeting FAR 36 today. NASA's current Quiet Engine Program is expected to demonstrate in the 1975 time period the technology for engines about 10 EPNdB quieter than current engines, without major performance penalties. Still further engine improvements, including jet-noise suppression, may be expected. When quiet engines are combined with steep approaches, it should be possible to achieve noise levels of about 90 EPNdB in the 1980's for most subsonic transports. Noise levels for long-range supersonic and hypersonic transports are apt to fall only slightly below FAR 36, however.

If the air terminals open their arms to quiet airplanes without some other improvements to the total system, the congestion problem of today can only get worse. Already terminal-area delays are estimated to cost over $160 million per year, as shown in figure 4. This not only represents increasing fares to the passenger, but is symptomatic of the inability of many terminals to handle more traffic. There are a number of ways to alleviate this problem. Larger aircraft carry more passengers without increasing the traffic density – but they reduce the flexibility of the transportation system. Secondary airports serving short-haul routes with suitable aircraft could off-load the most congested terminals. Also, improved ground and airborne electronic systems, coupled with acceptable aircraft characteristics, would improve all-weather operations, permit a wider variety of approach paths and closer spacing, and thereby increase airport capacity by about 100 percent if dual runways were provided.

Thus, quiet airplanes with advanced avionics and ATC equipment should alleviate the prime vehicle constraints on traffic growth and permit the market expansion projected earlier. This improvement in the ability to handle aircraft must be accompanied by an improvement in the ability to handle the passengers on the ground, but this problem will not be treated at this vehicle-oriented conference.

What, then, is the requirement for new aircraft types? I doubt if anyone at this conference feels that aircraft development is so mature that its future will parallel the past few decades of the automobile industry – that is, style changes without many basically new ideas or major advances. As good as today's aircraft are, they will ultimately be replaced by much better ones. The market will go to whoever has the best products to offer at the right times. Only by introducing better equipment can we maintain the record of holding down air-transportation costs in the future as we have in the past. As shown in figure 5, air fares have increased only 20 percent in the last 20 years, as compared with 60 to 100 percent for rail and bus fares and the consumer price index. At the same time our transportation has made dramatic advances in speed, comfort, and safety.

Improved aircraft types must be based on improved technology. Figure 6 lists some of the most promising opportunities for such improvements. Quiet propulsion leads the list. Higher engine temperatures will compensate for efficiency losses due to noise suppression. Composite structures can reduce overall structural weight by 20 percent. Powered lift will permit approach lift coefficients of 4 to 5 and field lengths less than 610 meters (2000 feet). Supercritical aerodynamics will enable aircraft to cruise efficiently at Mach numbers close to 1. Advanced supersonic arrow-wing configurations will produce cruise lift-drag ratios approaching 10 at Mach 3. Actively cooled structures may make possible hypersonic cruise with aluminum or titanium aircraft. Control-configured vehicles which depend on reliable SAS may improve efficiency by as much as 10 percent. And various advances in integrated avionics will reduce pilot workload, relieve terminal congestion, improve all-weather operation, and increase comfort and safety.

To achieve these technological advances will require a shot in the arm for civil aeronautics R & D. The application of these advances to operational aircraft probably will require increased government participation in limited partnership with the aircraft and airline industries in order to accelerate the introduction of advanced aircraft and transportation systems. As a step toward realization of the benefits of technological advances, increased use of research and experimental aircraft is anticipated.

One plan developed by NASA, in coordination with DOD and DOT, involves the design and manufacture of two STOL experimental transports. Although these aircraft probably will be smaller than the anticipated production transports, they will be well suited to the development of promising STOL augmented-lift concepts, control systems, and cross-wind landing gear. They will be capable of operating within acceptable noise limits to and from STOL strips about 610 meters (2000 feet) in length.

A second plan that has been developed jointly by NASA and the Army is expected to lead to advanced helicopter and tilt-rotor research vehicles. Although oriented primarily toward military applications, the research results also will be applicable to commercial VTOL transports.

Some of the advanced transport aircraft which might be forthcoming in the future are listed in figure 7. The sizes are merely representative of the results of some current studies. The first three vehicles are vertical- and short-takeoff machines for the short-haul market. Advanced subsonic transports of the 1980's will incorporate most of the applicable advanced technology just described with significant gains in productivity. An SST with longer range and higher capacity is needed for both the transatlantic routes and the longer routes across the Pacific. And a hypersonic transport (HST) is a possibility that cannot be discounted for the 1990's. Before examining some of the projected characteristics of these transports, just a word about general aviation.

Transition from the realm of hypersonic flight to the world of general aviation is a contrast in technology. Yet the opportunities associated with general aviation are as diverse and as challenging as any considered in this conference. In sheer number of aircraft, general aviation represents the largest sector of the U.S. aircraft fleet and runs the gamut from personal owner aircraft to corporate jet transports. The problems and opportunities are manifold, but the overriding concerns are for safety, economy, and all-weather utility. New technology in the form of advanced airfoils and flap systems, rotary-combustion engines, low-cost gas turbines, and improved control and navigation systems offer opportunities for significantly increasing the overall utilization of general aviation aircraft. The problem is cost, and this becomes the major challenge to the ingenuity of the designer.

Turning back to the commercial transports, figure 8 illustrates the possible evolution of rotorcraft transports. By 1980 we could see the widespread use of quiet and efficient compound helicopters having 250-knot cruise speeds and DOC's about half that of current helicopters. Sufficient avionic improvements would be incorporated to permit all-weather operations. By 1985 it should be possible to introduce a tilt-rotor transport which would be both faster and more efficient than the compound helicopter. Fully automatic flight with pilot monitoring and override is desirable and should be possible for these short-haul machines where minimization of flight delays is at a premium. As mentioned earlier, the NASA/Army rotor test vehicle and tilt-rotor research aircraft should contribute heavily to these new transport types.

Large-scale STOL operation with jet transports such as that shown in figure 9 should be possible by the 1980 time period. Powered lift will permit us to retain high cruise speed while reducing field lengths to less than 610 meters (2000 feet). Noise should be held to 95 EPNdB or less at 150-meter (500-foot) sideline for these first-generation STOL transports. The NASA STOL research aircraft are expected to evaluate at least two powered lift techniques, which will be discussed in a later paper.

By 1985 we could see VTOL operations (figure 10) with a lift-fan system. If successfully developed, it seems likely that an aircraft of this type could displace rotorcraft from the longer range VTOL routes, and could make substantial inroads into the STOL route structure because of its inherent ability to operate in either mode. An experimental aircraft will again be required to pave the way for a commercial transport.

It is in the area of medium- and long-haul transportation that the most advanced equipment is available today. With the first-generation jets still relatively productive, the second-generation wide-bodied transports are now entering service. The three new aircraft now available will soon be supplemented by the European twin-jet airbus and perhaps a U.S. counterpart. How, then, could we possibly need more of this type of aircraft? The answer probably is that we don't for quite a few years. But by 1980 even some of these fine aircraft will start being replaced, and the market will go to the best equipment then available. As we see it (figure 11) this new equipment will fly about 15 percent faster, will be at least 10 EPNdB quieter, and will have about 10 percent lower equivalent DOC's. The aircraft will be considerably advanced in terms of integrated avionics and controls. We may see the introduction of a 450 000-kilogram (million-pound) aircraft with dual passenger/cargo roles in a day/night shift arrangement.

The supersonic transport has just emerged and taken its first bow (figure 12). The Concorde and the TU-144 are both impressive first steps. And although they tend to be somewhat noisy, and have relatively high DOC's, it seems probable that an appreciable number of passengers will pay premium fares, if necessary, for the large time saving on the important North Atlantic route. In other words, the aircraft may be expected to sell and to earn a profit. At the same time it is clear that the second generation of SST's will be appreciably better. If we can successfully integrate all of the foreseeable advanced technology, we can expect cruise speeds of Mach 2.7 or higher, twice the current payload fraction, ranges of 5000 n. mi., FAR 36 noise levels, DOC's approaching today's subsonic jets, and an appreciably lower level of sonic boom. The second-generation SST will not only be more profitable on transatlantic runs, but will open up the Pacific to such rapid transportation.

While most of the technology for second-generation SST's up to Mach 3 is within reach, still faster transportation will require dramatic technical advances. Should we push on to Mach 4, or should we reach beyond to the hypersonic transport (HST) in the Mach 7 range? Or should we borrow space shuttle technology and jump to the suborbital transport? The HST appears to be a possibility worthy of concentrated research (figure 13). It is conceivable that in the 1990's we could have hypersonic transports flying close to 4200 knots at altitudes that would reduce the sonic boom to less than 48 N/m^2 (1 lb/ft^2) – possibly an acceptable level for flight over land. DOC's would be only slightly greater than those of today's aircraft. The hypersonic transport would use liquid hydrogen fuel which would cool both the engine and structure in one current concept. If this approach succeeds, conventional structural materials could be used and the passengers might even be afforded a normal view through a cooled window. To get from here to there, however, will require a hypersonic research airplane which is now under study. The new development which has sparked increased confidence in eventual hypersonic transportation is our successful experience with hydrogen fuel in the space program.

This has been a very brief overview of some of the new civil aircraft which might appear on the scene in the years ahead. During the balance of this conference you will learn more about these aircraft and the technology that would make them possible. Some projections may seem conservative; some very far out. Whatever your reaction, I hope it will be one of interest.

WESTERN WORLD AIR TRAFFIC FORECAST

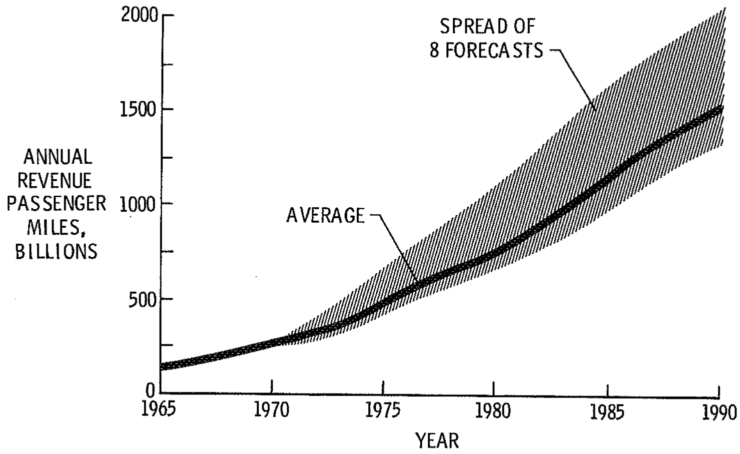


Figure 1

PROJECTED AIRLINE PURCHASES THROUGH 1985

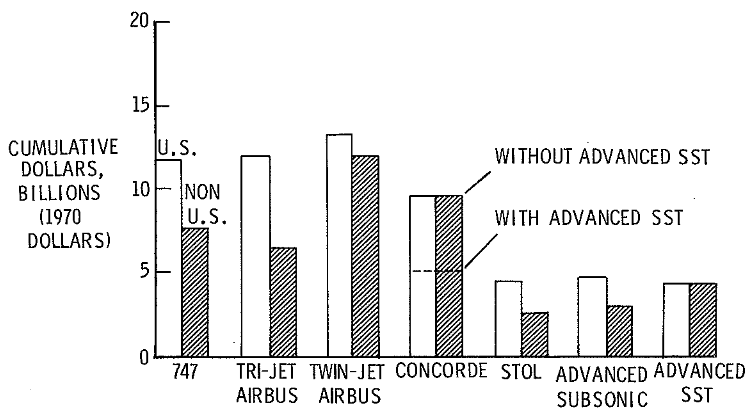


Figure 2

AIRCRAFT NOISE TRENDS

MULTIENGINE CTOL AIRCRAFT

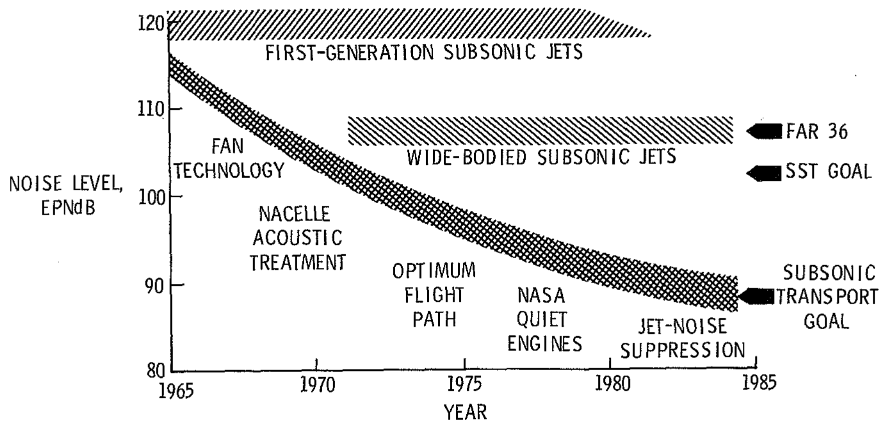


Figure 3

ESTIMATED U.S. AIRLINE COST FOR TERMINAL-AREA DELAYS

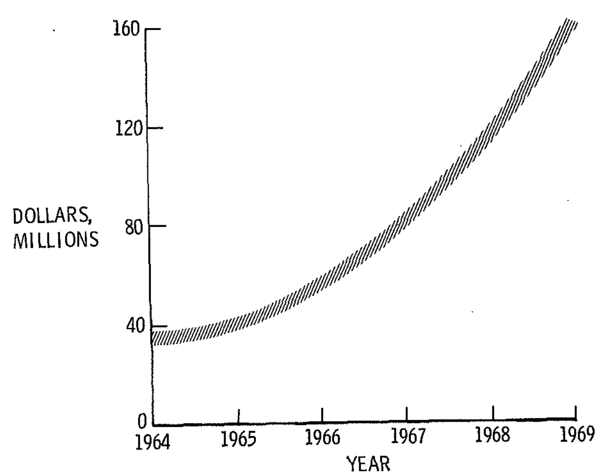
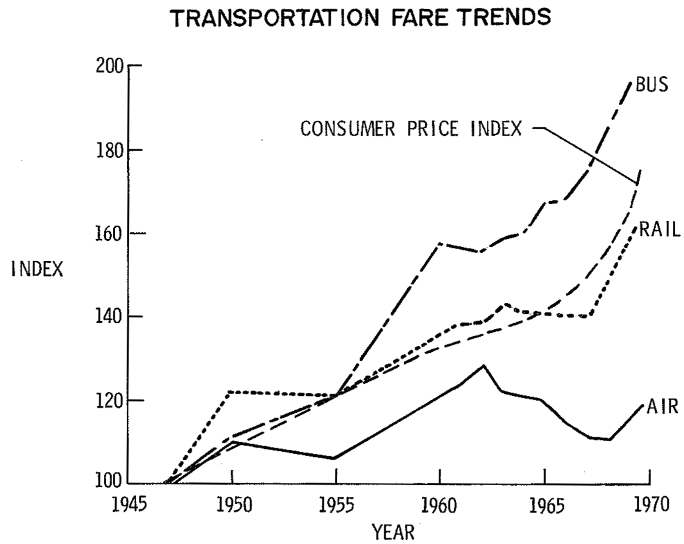


Figure 4



TRANSPORTATION FARE TRENDS
200
180
160
140
120
100
INDEX
1945
1950
1955
1960
1965
1970
YEAR
BUS
CONSUMER PRICE INDEX
RAIL
AIR

Figure 5

FUTURE TECHNOLOGY FOR CIVIL AVIATION

TECHNOLOGY	RESULT
• QUIET PROPULSION	90 EPNdB SUBSONIC TRANSPORTS 108 EPNdB SST
• HIGHER ENGINE TEMPERATURES	HIGH EFFICIENCY
• COMPOSITE STRUCTURES	20% WEIGHT DECREASE
• POWERED-LIFT AERODYNAMICS	APPROACH C_L OF 4 TO 5
• SUPERCRITICAL AERODYNAMICS	$M_{CRUISE} \rightarrow 1$
• ADVANCED SUPERSONICS	$L/D \rightarrow 10$ AT $M = 3$
• ACTIVELY COOLED STRUCTURES	HYPERSONIC CRUISE
• CONTROL-CONFIGURED VEHICLE	+10% RETURN ON INVESTMENT
• INTEGRATED AVIONICS	
AREA NAVIGATION	ROUTE VERSATILITY
AUTOMATED ATC	CONGESTION RELIEF
MICROWAVE ILS AND ATOL	CATEGORY III OPERATIONS
DIGITAL FLIGHT CONTROL	REDUCED WORKLOAD
ADVANCED ACTIVE CONTROLS	COMFORT AND SAFETY

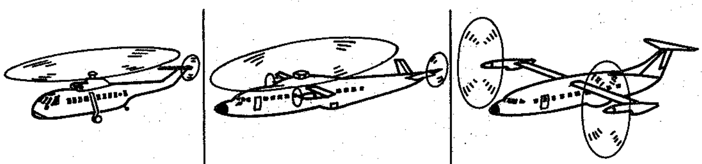
Figure 6

TRANSPORT AIRCRAFT OPPORTUNITIES

- 100-PASSENGER ROTORCRAFT
- 150-PASSENGER TURBOFAN STOL
- 100- TO 150-PASSENGER LIFT-FAN VTOL
- ADVANCED TECHNOLOGY SUBSONIC TRANSPORTS
- 5000-n. mi. SST
- HYPERSONIC TRANSPORT

Figure 7

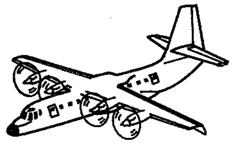
ROTORCRAFT TRANSPORTS

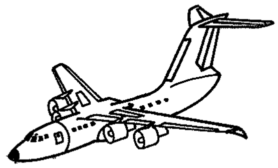


	CURRENT	1980	1985
SPEED, knots	130	250	350
PASSENGERS	25	90	110
PAYLOAD RATIO	0.24	0.32	0.38
GROSS WT. kg	9 500	27 200	27 200
GROSS WT. lb	21 000	60 000	60 000
NOISE, EPNdB	110	85	85
RELATIVE DOC	1.0	0.5	0.33
CONTROL	MANUAL	FLY-BY-WIRE	AUTOMATIC

Figure 8

STOL TRANSPORTS

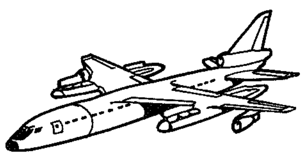




	CURRENT	1980 TURBOFAN
MACH NO.	0.35	0.75
PASSENGERS	50	150
GROSS WT. kg	22 700	59 000
GROSS WT. lb	50 000	130 000
RANGE, n. mi.	600	600+
FIELD LENGTH m	<610	<610
FIELD LENGTH ft	<2000	<2000
NOISE, EPNdB AT 150-m (500-ft) SIDELINE	109	95
RELATIVE DOC	1.0	0.85

Figure 9

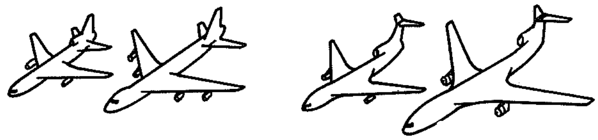
FUTURE VTOL TRANSPORT



MACH NO.	0.8
PASSENGERS	100 TO 150
NOISE	95 EPNdB AT 150-m (500-ft) SIDELINE
COST	COMPETITIVE WITH STOL AND CTOL SYSTEMS
CONTROL	AUTOMATIC TAKE-OFF AND LANDING LOW-SPEED CONTROL FROM THRUST MODULATION

Figure 10

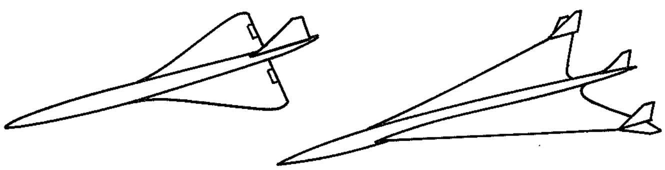
ADVANCED TECHNOLOGY TRANSPORTS



	CURRENT	1980's
MACH NO.	0.85	0.9 TO 0.98
PASSENGERS	260 TO 370	200 TO 500+
GROSS WT. kg	181 000 TO 340 000	109 000 TO 450 000+
GROSS WT. lb	400 000 TO 750 000	240 000 TO 1 000 000+
RANGE, n. mi.	3000 TO 5500	3000 TO 5500
NOISE, EPNdB	104 TO 108	90
DOC, cents/seat-mi.	0.75 TO 0.85	0.65 TO 0.80

Figure 11

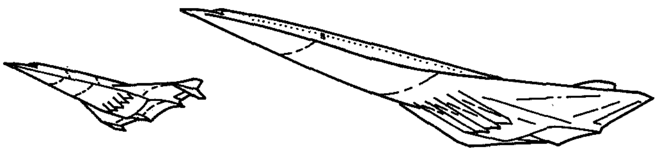
SUPERSONIC TRANSPORTS



	CURRENT	1980 SST
MACH NO.	2.1	2.7
PASSENGERS	100+	350+
PAYLOAD RATIO	0.055	0.10
GROSS WT. kg	175 000	363 000
GROSS WT. lb	385 000	800 000
RANGE, n. mi.	3200	5000
NOISE, EPNdB	111 TO 115	108
DOC, cents/seat-mi.	1.5 (?)	≈1

Figure 12

HYPERSONIC TRANSPORT



	RESEARCH AIRPLANE	1995 TRANSPORT
MACH NO.	6 TO 12	7 (4200 KNOTS)
GROSS WT. {kg / lb}	36 300 80 000	204 000 450 000
PAYLOAD {kg / lb}	680 / 1500 } EQUIPMENT	28 000 / 62 000 } 300 PASSENGERS
RANGE, n. mi.	1500	7000
CRUISE BOOM {N/m^2 / lb/ft^2}	--- ---	<48 <1
FUEL	LH_2	LH_2
PROPULSION	ROCKET/SCRAMJET	TURBO/SCRAMJET
STRUCTURES	PASSIVE OR ACTIVE COOLING	ACTIVE COOLING (ALUM. OR TITANIUM)

Figure 13

HIGH-LIFT AERODYNAMICS

By Alexander D. Hammond
Langley Research Center

2

INTRODUCTION

There is a continuing interest in the development of high-lift devices for improving the take-off and landing performance of civil transport aircraft. The purpose of this paper is to present some of the important aspects of the development of high-lift systems. The high-lift devices to be discussed include both powered- and unpowered-lift concepts. The landing performance that can be attained with the use of present-day state-of-the-art lifting systems as well as the predicted improvements in performance is related to some of the limitations imposed by operational constraints, particularly the limitations imposed on the usable lift of powered-lift concepts. The relative noise of powered-lift concepts for propulsion systems is discussed in paper no. 7.

SYMBOLS AND ABBREVIATIONS

C_L	lift coefficient, $L/q_\infty S$
$C_{L,max}$	maximum lift coefficient
c	chord, meters (ft)
L	lift, newtons (lb)
q_∞	free-stream dynamic pressure, newtons/meter^2 (lb/ft^2)
S	wing area, meters^2 (ft^2)
T	thrust, newtons (lb)
$(T/W)_{INSTALLED}$	ratio of installed thrust to take-off gross weight
V	approach velocity, knots
W	total aircraft weight, newtons (lb)

W/S wing loading, kilonewtons/meter^2 (lb/ft^2)

α angle of attack, degrees

B.L.C. boundary-layer control

L.E. leading edge

STOL short take-off and landing

DISCUSSION

The essential elements of a two-dimensional high-lift-flap computer program that has recently been obtained by contract from the Lockheed-Georgia Company (ref. 1) are shown in figure 1. Depicted in the sketch in the figure is a single-slotted flap having flow around the two airfoil elements and through the slot forward of the flap. The first step in computing the single-slotted-flap characteristics is to obtain the potential-flow solution for the flow around each of the airfoil elements in proximity of the other. From the data obtained, the basic boundary layer for each airfoil element is computed, and these results are combined with results from the slot-flow analysis program to obtain the flow interaction between the basic boundary layer and the flow through the slot in what is called the "confluent boundary layers." A combined solution is computed and then iterated to obtain the pressure coefficients over the main airfoil and the flap. The two-dimensional forces and moments are obtained by integrating the pressure data. It has been found that the initial results from this two-dimensional program agree well with available two-dimensional data on flapped airfoils.

An application of the Lockheed-Georgia program is shown in figure 2 and reported in reference 2. The experimental data were obtained from reference 3 from a gap optimization investigation using a single-slotted flap having a 10° drooped nose and a flap deflection of 30°. The flap was moved with respect to the airfoil in order to vary the gap as is indicated in the sketch. The two-dimensional lift coefficient is presented as a function of the flap gap in percent of the basic wing chord. The open symbols indicate the experimental data at an angle of attack of 0° and the solid symbols, at the angle of attack corresponding to $C_{L,max}$. The predicted analytical results are shown by two curves; the inviscid prediction from potential flow is shown as the dashed curve and the viscous prediction already described, as the solid curve. An angle of attack of 0° was selected for the comparison because the analytical prediction cannot account for separation effects and is only valid up to that point where separation occurs on the wing. The inviscid prediction does not give an optimum gap setting, but it indicates that the maximum lift for this flap configuration at 0° angle of attack would occur for zero gap, which is not in agreement with previous experience. The viscous prediction, although the variation with flap gap does not agree completely with the experimental data, indicates that the optimum gap is about 2 percent c, which is in agreement with the experimental data. One other important result shown in figure 2 is that the optimum gap prediction for $\alpha = 0^{\circ}$ is also the optimum gap for the maximum lift coefficient as shown by the experimental data. These predictions show promise that the analytical procedures will provide valuable guidance in optimizing flap configurations. An attempt will be made to expand this two-dimensional theoretical work into three-dimensional applications in the near future. Some recent preliminary work at the Langley Research Center has indicated that analytical techniques such as those employed on unflapped airfoils can be used to predict the lift on three-dimensional airfoils with high-lift devices. Future work to improve this technology will be undertaken at Langley.

Some information on the landing performance of high-lift systems is presented in figure 3. The variation of lift coefficient with angle of attack for two types of high-lift systems is shown in the left-hand plot of figure 3. The bottom curve represents the present state of the art for mechanical flaps, and the top curve is for these same mechanical flaps with discrete leading-edge blowing. Mechanical-flap systems can attain maximum lift coefficients as high as 3.5. With this level of maximum lift coefficient, the maximum usable lift coefficient of approximately 2.0 (indicated by the circle symbol) can be maintained in the landing approach. The calculated usable lift was based on the present-day rules for civil transports which use a 1.3 speed margin between the stall speed (maximum lift capability) and the approach speed. In the right-hand plot of figure 3 is presented the variation of lift coefficient with velocity for wing loadings of 3.83 kN/m^2 (80 lb/ft^2) and 5.75 kN/m^2 (120 lb/ft^2). Also given in this plot is the field length that goes with the velocities when they are considered to be approach velocities. It can be seen from the data that, for a lift coefficient of 2.0 in the approach, as indicated for the mechanical flap, the level-flight approach speed is in the neighborhood of 138 knots with a corresponding field length of about 1.5 km (5000 ft) for a wing loading of 5.75 kN/m^2 (120 lb/ft^2). However, if this same lift capability is used on an airplane with a low wing loading, such as 3.83 kN/m^2 (80 lb/ft^2), the approach speed can be reduced to about 115 knots and field length can be reduced to about 1.06 km (3500 ft). If leading-edge blowing or discrete boundary-layer control is used at the leading edge of a mechanical flap system in such a way that it is not appreciably affected by the engine power level and if the present ground rules for speed margins apply, an approach lift coefficient of around 2.5 can be attained. With an approach lift coefficient of 2.5 and the higher wing loading, field lengths of 1.2 km (4000 ft) can be attained. For the lower wing loading, the speed can be reduced to about 90 knots with a corresponding reduction in field length to 0.76 km (2500 ft). If performance levels much higher than that shown for the flaps plus leading-edge blowing are desired, use of what is called powered lift will be required. Powered-lift concepts develop circulation lift that is proportional to the installed thrust-weight ratio of the aircraft and depend on the efflux from the power source to develop a large portion of this additional lift. Some high-lift flap concepts that might be considered for STOL airplanes and make use of powered lift are shown in figure 4.

The airplane configuration in figure 4(a) has an internally blown flap system in which air is taken from the engines and ducted through the wing out to slots across the wing span. The engines have to have a relatively high pressure ratio to pump air efficiently through these ducts. The sketch at the top of figure 4(a) shows a conventional jet flap which could be installed on such an airplane for which this ducted air is blown out of slots over a simple trailing-edge flap system. A considerable increase in lift capability can be obtained with the jet flap. An augmentor-wing high-lift system which also might be used on this airplane is illustrated at the bottom of figure 4(a). This augmentor wing not only uses the air that is blown out of the slot but the flap itself is essentially an ejector which entrains some of the free-stream air into the flap system to augment the jet thrust. With this augmentation, the efficiency of the flap is improved; however, the mechanical complexity is increased.

The externally blown flap systems or those systems that depend on the external flow of the efflux from the engine are shown in figure 4(b). This type of system takes advantage of very high-bypass-ratio engines that have low noise levels and represent a simple application of the jet-flap principle. The sketch at the top of figure 4(b) shows an externally blown flap system with a well-designed, double-slotted flap which is deflected behind the engine so that the air flows through the slots and induces circulation lift over the wing. Another externally blown concept shown at the bottom of figure 4(b) routes the efflux from the engine through a duct and utilizes a nozzle at the exit of the duct to deflect the flow. The duct can be extended to locate the exit in the vicinity of the flap system as shown in the sketch; there is thus obtained direct turning of the jet internally within the duct and at the same time additional induced circulation lift over the wing.

The variation of lift coefficient with angle of attack for the powered-lift STOL concepts just discussed is presented in figure 5 (open symbols). The externally blown flap data were obtained from unpublished results of tests made in the Langley V/STOL tunnel; the data for the other concepts were obtained from references 4 to 6. The level of lift shown in this figure was adjusted so that at 0° angle of attack (solid symbol) all the powered-lift concepts had identical lift capabilities. The variation with angle of attack of the lift that would be obtained for a constant thrust coefficient (throttle setting) throughout the angle-of-attack range is illustrated. It is interesting to note that all the powered-lift concepts show the same variation with angle of attack and the same performance level. However, in order to attain this performance level, a different level of thrust coefficient is used and, thus, a different amount of installed engine power is required in the airplane. The conventional jet flap and the augmentor wing have thrust-weight ratios between 0.32 and 0.39 installed in the airplane to obtain the performance capability shown. The externally blown flap and the deflected-thrust flap systems require higher installed thrust-weight ratios of about 0.47 to 0.48. These installed thrust-weight levels have not been finalized at this time, and they depend considerably on the amount of margin between the usable lift and the maximum lift attainable because of considerations with regard to operational constraints that are discussed later. Some other important factors need to be considered first. The conventional jet flap and the augmentor wing both require internal ducting which has an advantage in regard to lateral trim for the engine-out problem, but relatively high-pressure-ratio engines are required to pump the air through the wing ducting efficiently. On the other hand, the externally blown flap and the deflected-thrust concepts can take advantage of the quieter high-bypass-ratio, low-pressure-ratio fan engines. It turns out that the gas generators or core engines required to drive the high-bypass-ratio fan and to pump either the augmentor wing or the conventional jet flap are about the same size. The static thrust rating for the high-bypass-ratio engines used with externally blown flaps is higher than that for the high-pressure-ratio engines used with internally blown jet flaps. Therefore, the installed thrust-weight ratio is really not as far out of line as it first appears. With regard to the performance of these configurations, it can be seen in figure 5 that an approach lift coefficient in the neighborhood of 4 can be utilized. With this level of approach lift performance, an airplane with a wing loading of 5.75 kN/m^2 (120 lb/ft^2) would have a field length of around 0.76 km (2500 ft) or slightly higher and with a lower wing loading of 3.83 kN/m^2 (80 lb/ft^2) would have a field length of 0.61 km (2000 ft) which is the goal for STOL vehicles.

The variation of lift coefficient with installed thrust-weight ratio for externally blown flap configurations is shown in figure 6. The lower curve shows the approach lift capabilities and the upper curve shows the maximum lift capabilities. There is a considerable gap between the approach lift and the maximum lift attainable as shown by the data. The large difference is related to the margins that must be allowed to account for some of the operational and safety aspects of STOL flight. (See refs. 7 to 9.) Some of the considerations that go into the setting of these margins are shown in figure 7 where lift coefficient is plotted as a function of increasing lift capability. For the mechanical flap systems, present regulations require a speed margin of about 1.3 between stall speed and approach speed and that margin is indicated in the figure. If just the speed margin or the operational margin is considered, as indicated by the difference between the upper curve and the next curve below, there is still a margin of only 1.2 to 1.3 stall speed even for the powered-lift concepts. However, other considerations need attention as the lift is increased and power is used to create that lift. Considerations of engine out and ground effect need to be accounted for, so that the total lift margin for powered-lift systems with the higher lifting capabilities represents a stall speed margin of about 1.5. The operational margins that are provided for present-day mechanical flap systems are well understood and include the margins needed to provide a flare capability at touchdown and a margin in angle of attack between the approach lift and the stall so that gusts do not cause an inadvertent stall on the wing. The angle of attack that can be used in the landing flare must be limited to avoid tail contact with the ground as the airplane touches down. Also, it must be recognized that for mechanical flap systems, the margin provided for engine out is inherent in establishing the maximum lift capability but the maximum lift capability is not dependent on engine power. Very little if any account is taken of the engine-out condition but, as the lifting capability is increased, the engine-out condition becomes more of a problem. There are, depending on the configuration under consideration, engine-out conditions that require strict accounting for the lateral-directional trim as well as keeping the wings level to account for the loss of lift on the side of the dead engine. One other characteristic that all these STOL lift systems have in common is that, as the powered lift is increased above a level of about 3.0, there comes into being an adverse ground effect due to this lift capability as the airplane approaches the ground (ref. 7). This adverse effect becomes worse with increases in lift, as indicated in figure 7 by the margin shown for the ground effect. It is not known at this time whether these margins can be substantially reduced. However, it is believed that further insight into these problem areas will result from the forthcoming NASA Flight Research Program to use the STOL aircraft to investigate the problem areas. However, a look into the future to predict what would happen to these margins or this lifting capability indicates that it might be possible to attain performance characteristics as shown in figure 8. A smaller difference is seen between the maximum lift available and the usable lift, when compared with the information in figure 7. The usable lift may get closer to the maximum lift available by utilizing an active control system discussed by A. Gerald Rainey in paper no. 13 or by using other airplane control concepts that will help to account for the engine out and the ground effects better than they can be handled manually. Those working in this area are hopeful that the engine-out and ground-effect problems will have less important roles in the required margins and, as better control systems are put into the airplane and as more experience is gained in flying these powered-lift systems, more of the maximum lift available can be utilized. Furthermore, the complexity of the airplane configuration may be reduced for the same lifting capability or an increase in the wing loading and, thereby, the productivity of the STOL airplane can be realized.

CONCLUDING REMARKS

From results of research on high-lift devices to improve the take-off and landing performance of civil transport aircraft, it is evident that future aircraft will utilize powered lift. The degree to which this powered lift is utilized will depend on whether the usable lift that can be gained from powered-lift concepts can be increased. It is believed that future research should effect an increase in the amount of usable lift.

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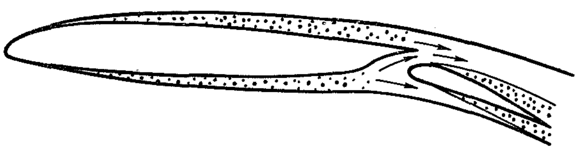


- POTENTIAL-FLOW SOLUTION
- BASIC BOUNDARY LAYER
- SLOT-FLOW ANALYSIS
- CONFLUENT BOUNDARY LAYERS
- COMBINED SOLUTION

Figure 1.- High-lift-flap computer program.

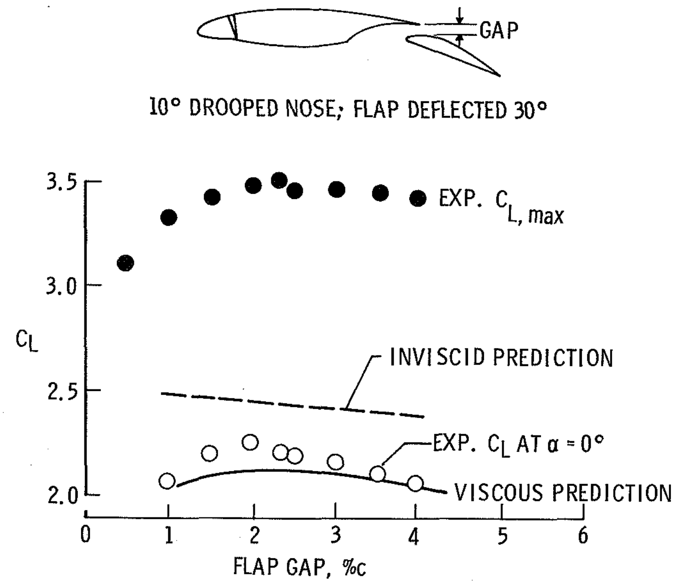


Figure 2.- Gap optimization.

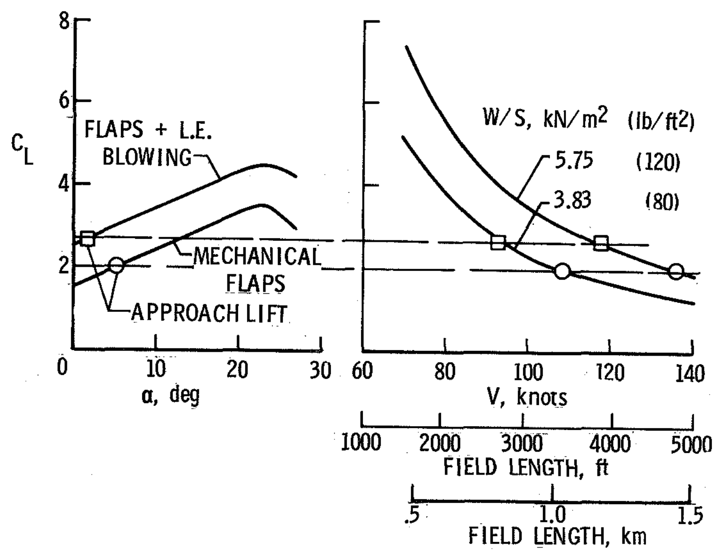
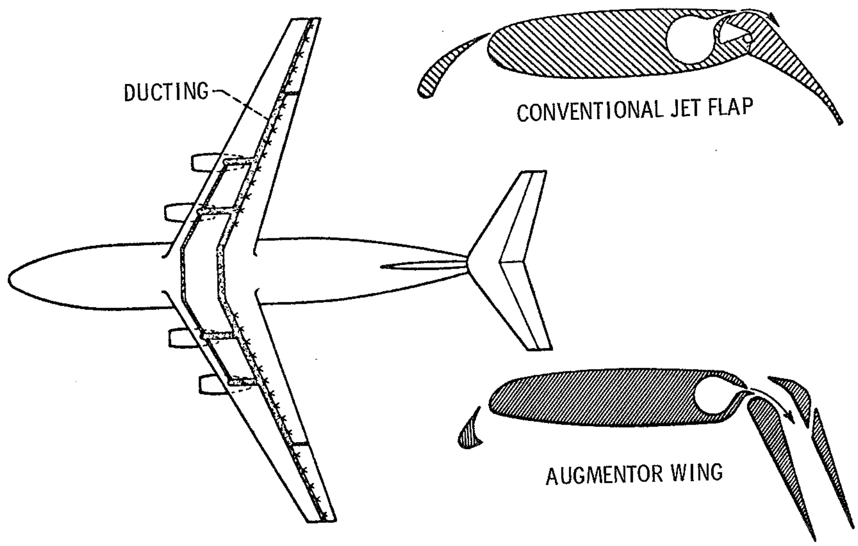
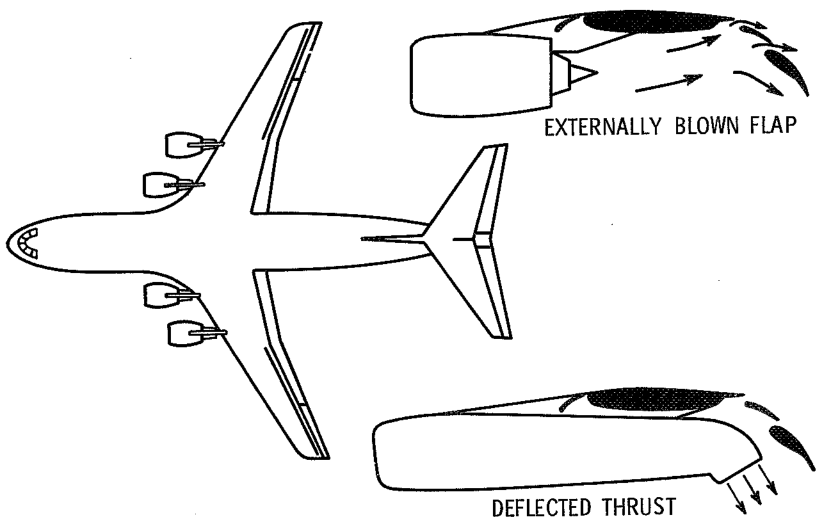


Figure 3.- Landing performance of unpowered high-lift systems.



(a)



(b)

Figure 4.- STOL high-lift systems.

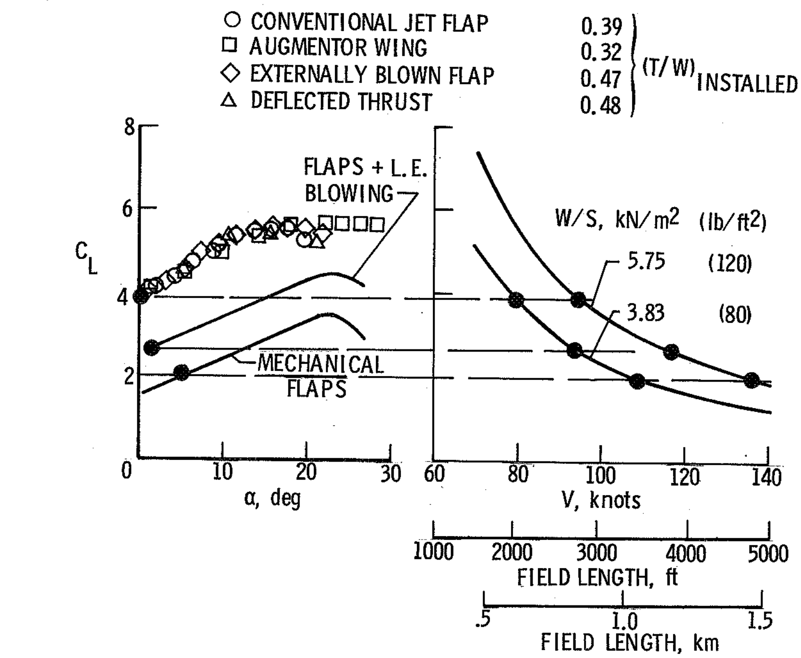


Figure 5.- Landing performance of high-lift systems.

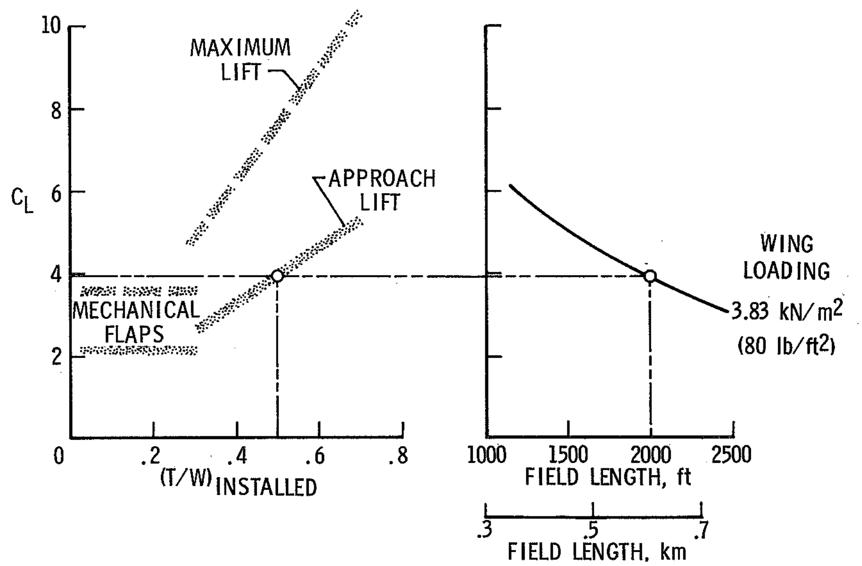


Figure 6.- Landing performance of externally blown flap configurations.

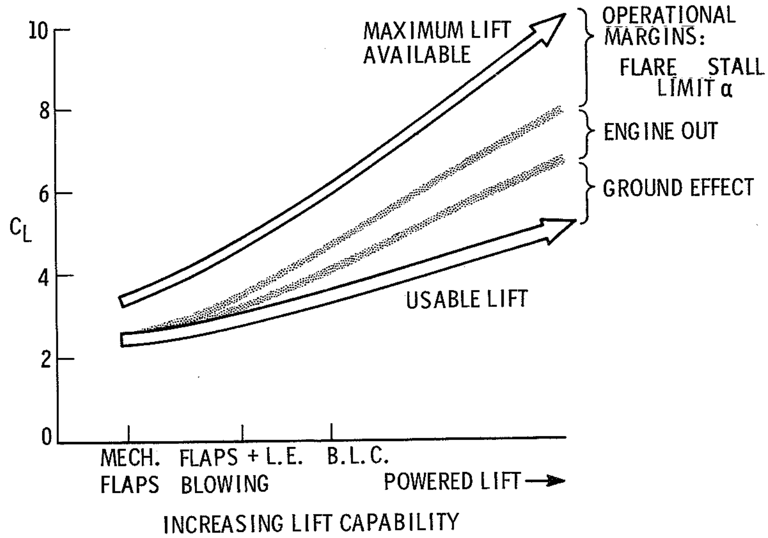


Figure 7.- Limitations imposed on usable lift.

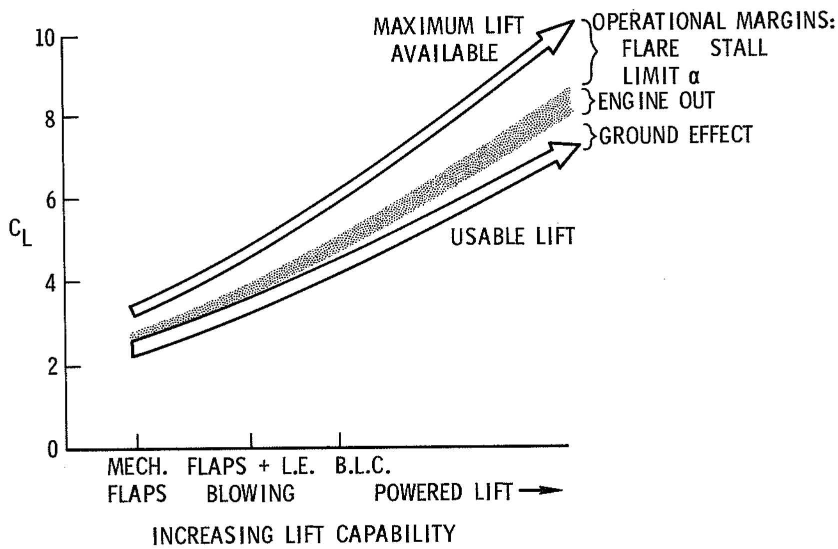


Figure 8.- Research goals for increased usable lift.

SUBSONIC AND TRANSONIC AERODYNAMIC RESEARCH

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SUMMARY

Results of some subsonic and transonic aerodynamic research applicable to the design of civil aircraft in the 1970's and beyond are presented, and some of the needs for future research are outlined. Advances in technology afforded by the supercritical aerodynamic concepts and the combination of control-configured vehicle and variable-wing-sweep concepts are discussed, and both progress and needs in aerodynamic theory in the area of component interference, mixed flow, and leading-edge vortex flow are reviewed.

3

INTRODUCTION

The purpose of this paper is to discuss some results of subsonic and transonic aerodynamic research that are expected to have an impact on the design of civil aircraft in the 1970's and beyond and to indicate some of the directions in which future research is expected to be profitable. Aerodynamic research in this speed range obviously cannot be limited to those areas applicable to subsonic and transonic aircraft but must be related to a wide range of aircraft from subsonic designs through hypersonic designs. From the wide matrix of aerodynamic considerations three main topics have been selected for discussion. The first topic, supercritical aerodynamics, is primarily applicable to aircraft designed to cruise at subsonic or near-sonic speeds, and the concepts and some potential aerodynamic improvements to be derived will be discussed. With regard to the subsonic aerodynamics of aircraft designed to cruise at supersonic or hypersonic speeds, the variable-sweep wing concept has been selected, and application of the control-configured vehicle concept to broaden the variable-sweep aircraft design options will be discussed. The final area deals with the improved aerodynamic theories needed for the design and analysis of advanced aircraft.

SYMBOLS

A	aspect ratio
C_D	drag coefficient
C_L	lift coefficient

C_p	pressure coefficient
ΔC_D	drag-due-to-lift coefficient
c	chord
D	drag
L	lift
l	fan-pod length
M	Mach number
$\left(M \frac{L}{D}\right)_{max}$	aerodynamic efficiency parameter
t	airfoil thickness
V	free-stream velocity
x	longitudinal coordinate
α	angle of attack

Subscript:

max	maximum

SUPERCRITICAL AERODYNAMICS

A very promising opportunity for improving aerodynamic performance of civil aircraft in the 1970's and beyond is that afforded by the supercritical aerodynamic concepts developed by Dr. Richard T. Whitcomb and his colleagues at the NASA Langley Research Center. The basic element of this technology is the two-dimensional supercritical airfoil concept, which provided the major breakthrough by materially increasing the drag-rise Mach number.

The basic principle of the supercritical airfoil is illustrated in figure 1 by comparison with a conventional airfoil. For the conventional airfoil, approximately 12 percent thick and operating at a Mach number of 0.7, the flow accelerates over the upper surface, and a rather extensive supersonic flow region develops that is terminated by a strong shock wave. Of course, some drag increase is associated with energy losses in this decelerating wave; however, the greatest portion of the drag increase results from boundary-layer separation induced by the pressure rise through the strong shock. For the supercritical airfoil, the upper surface curvature is markedly reduced except near the trailing edge. This reduced curvature decreases the velocities and delays the formation of the supersonic flow region to a higher flight velocity. The example shown is for a Mach number of 0.8, and even for this relatively large increase in Mach number over that for the conventional airfoil, the velocities in the supersonic region are diminished and the vertical extent of the region is reduced. Therefore, the energy loss in the wave is reduced, but more importantly, the shock-induced boundary-layer separation is eliminated. The lift lost by the reduced velocities is then recovered by the high trailing-edge camber in the subcritical flow aft of the shock. The concept can also be utilized to provide increased airfoil thickness while the same drag-rise Mach number as a thinner conventional airfoil is maintained.

Application to Straight Wings

The application of the two-dimensional supercritical concept to improve the characteristics of a straight-wing aircraft is illustrated in figure 2. Here the cruise Mach number is presented as a function of wing thickness-chord ratio for both conventional and supercritical airfoils as applied to the T-2C aircraft. Two sources of data are presented. The shaded lines represent estimates based on wind-tunnel data and illustrate both the effect of airfoil section and the well-known decrease in cruise Mach number with increasing thickness. The circular symbols represent full-scale flight data obtained from a joint program by NASA, the U.S. Navy, and North American Rockwell, which utilizes the T-2C. For the case in which the thickness is held constant at 12 percent, wind-tunnel studies have indicated that a well-designed supercritical airfoil could provide a 15-percent increase in cruise Mach number relative to the conventional T-2C airfoil. It should be kept in mind that the particular percentage improvement quoted applies to the 12-percent-thick airfoil.

If the supercritical airfoil is used to allow an increase in thickness, both the flight results and the wind-tunnel results indicate that a 42-percent increase in thickness ratio can be provided while a constant cruise Mach number is maintained. Because of the rather extreme thickness, a somewhat conservative approach was taken in the design of this particular supercritical airfoil. It is believed that by additional research on thick airfoils, the cruise Mach number curve could be raised as indicated by the "projected" curve, and a total increase in thickness of 58 percent could be achieved. These large increases in wing thickness can be used either to reduce wing weight or, for the same weight, to allow an increase in wing aspect ratio. In either case this increase would provide greater wing volume for housing such items as powered-lift systems or increased fuel tankage. There should be many opportunities to apply this concept to future civil aircraft. Discussions relative to general-aviation and STOL aircraft are presented in paper no. 18 by M. R. Barber and Jack Fischel and in paper no. 21 by Woodrow L. Cook, respectively. An important area for future research is the development of additional supercritical airfoils covering a range of thicknesses and design lift coefficients.

Application to Swept Wings

With regard to swept wings, the supercritical airfoil, the area rule, and the wing sweep concepts can be combined so that the drag rise of a subsonic transport can be delayed to near-sonic speed. This concept is illustrated in figure 3. Here the upper bound for drag-rise Mach number of a lifting wing with a supercritical airfoil is presented as a function of wing half-chord sweep according to the well-known cosine rule. Also shown is the drag-rise Mach number of a good equivalent body of revolution developed at the Langley Research Center with the aid of the supercritical technology, which represents the drag-rise Mach number associated with volume and sets the upper limit for the complete aircraft configuration. It will be noted that the drag-rise Mach number of this body is nearly sonic. The optimum wing sweep and the good equivalent body are matched at the intersection of the two boundaries. The match is accomplished by providing the complete configuration with the good equivalent-body area progression by means of fuselage indentation and the wing glove and by providing the full benefit of sweep by use of the wing glove to reduce the unfavorable wing-fuselage interference. The symbol represents the experimental results of such a design approach and demonstrates the near-sonic capability.

Application of these concepts to increase both the aerodynamic cruise efficiency and the speed of a swept-wing transport aircraft is shown in figure 4, where the variation of the aerodynamic efficiency term in the range equation is presented as a function of Mach number for a conventional design and a supercritical design based on wind-tunnel development tests. The design incorporates the supercritical airfoil, improved planform, and good area progression, and it will be noted that the optimum cruise is very near sonic speed and that sizable increases in both cruise speed and efficiency are indicated relative to the conventional configuration. Also shown is an ideal upper bound and a wave-drag cutoff, and it can be seen that the supercritical concepts are pushing close to the upper right corner. The upper bound is based on the span and wetted area of the supercritical configuration, the assumption of full leading-edge suction, an elliptical span loading, a skin friction 10 percent above the flat-plate turbulent value, a compressibility-drag increment of 0.0025, and no trim drag.

It should be pointed out that for certain classes of aircraft the aft engines cause a serious balance problem. In this regard the Langley Research Center is currently supporting both in-house and contract research directed toward optimizing wing-mounted engine configurations. The probable impact of the supercritical technology on future subsonic and near-sonic transport aircraft is discussed in paper no. 22 by William J. Alford, Jr. Beyond a Mach number of 1.0, increases in wave drag and sonic boom are encountered, which will require additional configuration approaches. An approach to a possible transonic or low-supersonic transport is presented in paper no. 23 by Robert T. Jones.

Some directions in which additional research would be profitable include aerodynamic data required for design trade-off studies involving wing thickness, planform, and twist distribution and configuration arrangement with regard to both performance and stability. Langley Research Center is currently engaged in the initial phases of a program of this type.

VARIABLE SWEEP

With regard to improving the subsonic aerodynamic characteristics of supersonic and hypersonic cruise vehicles, the variable-sweep wing concept is still an attractive approach and is worthy of continued aerodynamic research, particularly in view of the broadened design options that might be expected from future advances in other areas such as materials and stability augmentation.

Before the possibility of increased options is discussed, a brief review of one of the potential subsonic aerodynamic benefits of variable sweep will be made. An illustration of the potential benefit of the variable-sweep wing with regard to the subsonic aerodynamic performance of a supersonic commercial transport is presented in figure 5, where the subsonic maximum lift-drag ratio is presented as a function of wing span for a constant aircraft wetted area. For a constant wetted area, the maximum lift-drag ratio tends to increase linearly with increasing wing span, and it is, of course, the ability of the variable-sweep wing to increase its span that provides its improved subsonic performance. To illustrate the general degree of improvement in subsonic cruise performance afforded by variable sweep, comparison is made between the SCAT 15-F type of fixed-sweep supersonic transport and a variable-sweep configuration. The typical range of span increase results in an improvement in subsonic L/D of about 20 percent. Further subsonic gains through the application of advanced materials and the supercritical airfoil are possible but require research related to the supersonic design point. The combination of large span and low sweep angle affords the opportunity for a more efficient high-lift system and reduced noise levels.

One of the problems encountered in previous studies of the application of variable sweep to a commercial supersonic transport was the limitation of configuration options associated with longitudinal stability requirements. However, the control-configured vehicle concept, which appears to be generating considerable interest, may offer the possibility of increasing the configuration options for variable-sweep aircraft. Basically, this concept utilizes a highly reliable stability augmentation system and relaxes the unaugmented stability requirements to allow more freedom to optimize the aircraft, and some general applications are discussed in paper no. 13 by A. Gerald Rainey.

The specific application of the control-configured concept to increase the design options of a variable-sweep aircraft is illustrated in figure 6. The upper sketch illustrates the usual situation where the aircraft is required to be longitudinally stable with the augmentation system off. With the center of gravity chosen to minimize the cruise trim drag, an outboard wing pivot location and large wing glove are usually required to avoid longitudinal instability with the wings in the low sweep position at subsonic speeds. The outboard pivot concept is very effective in reducing the shift in aerodynamic center that accompanies wing sweep changes (refs. 1 and 2) and has played an important part in the successful application of variable sweep to military aircraft. However, the associated large glove, while assisting in control of the aerodynamic center, does tend to contribute to stability and control problems at moderate and high lift, which can further reduce the design options by requiring "fixes" such as wing flow control devices and a low horizontal-tail location. The requirement for a low horizontal-tail location tends to create a jet-exhaust—tail interference problem for engine locations usually considered for a supersonic transport. These stability and interference problems contributed to the switch from variable sweep to fixed sweep for the now-canceled U.S. supersonic transport.

The lower sketch of figure 6 illustrates the additional freedom in configuration that may be possible by use of the control-configured vehicle concept. With this concept the wing pivot location is no longer dictated by a requirement for unaugmented static stability and could be moved inboard and forward while the desired open wing span is maintained. This particular application of the control-configured concept allows more freedom in pivot location, with the inboard position providing more depth for the pivot structure. Probably more important is the additional freedom in glove size, with the reduced size associated with the new pivot location alleviating the adverse high-angle-of-attack stability and control problems encountered with the large glove. This in turn reduces the need for wing flow control devices and allows more freedom in horizontal-tail location. The increased freedom with regard to horizontal-tail location is of particular importance with regard to jet-exhaust—tail interference. With the increase in configuration options and with a well-focused aerodynamic research program, the large potential gains in subsonic performance could become a reality in a second- or third-generation supersonic transport.

particularly if operational requirements or utilization considerations result in the need for appreciable subsonic flight.

IMPROVED THEORIES FOR DESIGN AND ANALYSIS

With the many advances in aircraft technology, the increases in design approaches, the importance of off-design conditions, and the highly competitive situation in the commercial aircraft field, the need for improved aerodynamic theories is increasing. With improved theoretical methods, a greater advantage could be taken of the computerized design and analysis techniques made possible by high-speed digital computers. The three general types of aerodynamic flows that have been selected for discussion relative to theoretical methods are illustrated in figure 7: (1) component interference, (2) mixed flow encountered at transonic speeds, and (3) leading-edge vortex flow. Some recent advances in these areas and some directions in which future analytical research should be directed will be discussed.

Component Interference

With regard to component-interference theory, high-speed digital computers have provided the opportunity for more realistic mathematical modeling of the complex nonplanar and multicomponent features of aircraft, and methods of varying degrees of complexity have been developed. Much of the early development of numerical techniques for arbitrary nonlifting bodies in incompressible potential flow utilizing distribution of singularities over the body surface was accomplished in the late 1950's and early 1960's at the Douglas Aircraft Corporation. An excellent summary of this important work and an extensive bibliography are presented in reference 3. At the present, most major aircraft manufacturers and research organizations have programs based on inviscid flow which are capable of representing a complete aircraft in the lifting condition at subcritical speeds. The degree of detail simulated is limited primarily by computer capacity or cost. Research is now underway both within industry and within NASA to determine the range of applicability of the various mathematical modeling techniques used with regard to aircraft design and analysis. An example of such a joint experimental and analytical program that is underway at the Ames Research Center is illustrated in figure 8. The model under study represents the wing, fan pod, pylon, and cruise-engine nacelle of a V/STOL configuration in the cruise mode. The analytical method being evaluated was developed by North American Rockwell under contract to the Ames Research Center, and the details are presented in reference 4. In this program the shapes of the fan pod, wing, and pylon are arbitrary and the cruise engine nacelle is axisymmetric. Vortex and source singularities are placed on the surfaces of the fan pod and nacelle, respectively. The pylon and wing are divided into two regions, each treated differently.

The major portions have the singularities placed on the chordal plane, whereas near component intersections they are located on the mean surface to provide a more realistic interference flow. Use of this program to reduce high-speed component interference and to avoid premature drag rise for complicated V/STOL configurations is currently being evaluated at Ames Research Center by suitable experimental tests, and some very recent and preliminary results of this study are shown. In figure 8 the interference pressures induced on the fan pod by the nacelle-pylon combination, as measured in the Ames 12-foot pressure wind tunnel at a wing-chord Reynolds number of 2.3×10^6, are compared with the theoretical predictions. In general, the results are encouraging, and the differences in the aft region may be due to boundary-layer effects. This investigation is continuing in the Ames 11-foot transonic wind tunnel over a range of Mach and Reynolds numbers and includes a very complete pressure-distribution coverage which should provide an important contribution in establishing the range of applicability of the numerical methods.

There are several areas in which future research should be directed. Methods of predicting the static and dynamic stability derivatives of complete aircraft configurations including thickness, nonplanar, and wake effects need to be developed and evaluated. In this regard the Langley Research Center has a research program underway which includes a contract with North American Rockwell for the development of an analytical program for the static and dynamic aerodynamic derivatives of complete aircraft, which includes improved calculations of leading-edge-suction distribution, particularly near leading-edge discontinuities, as well as the thickness, nonplanar and wake effects. The resulting method will be evaluated by suitable in-house experimental and analytical studies. This research program is enhanced by Langley's unique capability in the area of measurement of dynamic-stability derivatives. Some other areas of required analytical research are inclusion of boundary-layer effects, accurate techniques for calculating the three-dimensional lift-dependent profile drag, and improved methods of accounting for aero-elastic effects on stability and performance.

Mixed Flow

The possibility of cruising efficiently at near-sonic speeds has emphasized the need for the development of accurate mixed-flow theories for use in the design and analysis of advanced aircraft. The need for transonic-flow theories has, of course, been recognized for many years, and in the late 1940's and early 1950's, various approximate methods were used to study transonic flows. These methods, however, were unsuccessful in predicting mixed subsonic-supersonic regimes with embedded shock waves. In recent years, with the application of finite-difference methods for digital computation, more useful solutions have been obtained, primarily by the relaxation techniques. Reference 5 presents a very useful bibliography of transonic-flow theory.

Two examples of the recent progress in the area of two-dimensional mixed-flow theory are illustrated in figure 9. Presented on the left are sample results from a two-dimensional airfoil design method developed by Garabedian and Korn of New York University under a Langley grant. The method is based on an inverse, hodograph, complex-characteristics method and includes a turbulent-boundary-layer correction. In the example shown, the method was used to design a 12-percent-thick airfoil that provides smooth, shock-free flow at a supercritical Mach number of 0.75 and a lift coefficient of 0.63. The resulting airfoil shape is shown and the resemblance to the Langley developed supercritical airfoil is readily apparent. Experimental pressure distributions (obtained by Kaprycynski at the National Aeronautical Establishment, Ottawa) are in excellent agreement with the theory except for a very weak shock near the mid chord. As indicated on the right, an analysis method has also been developed, based on a relaxation method, and applied successfully to this same airfoil at off-design conditions where a relatively strong shock wave was present. The analysis method, however, does not account for boundary-layer effects, and it is believed that the slight difference indicated in shock location between the theory and experiment is due to the boundary-layer displacement effect.

At present the research on shockless-supercritical-flow theory is limited to two-dimensional flow, and extension of the shockless design method to three-dimensional wings does not appear to be feasible at the present time. However, researchers at both the Ames and Langley Research Centers are also actively engaged in the development of transonic analysis theories for both two- and three-dimensional inviscid mixed flows with shocks. An example of progress at the Ames Research Center in the difficult three-dimensional mixed-flow wing problem is illustrated in figure 10. The pressure distribution over the surface of a 6-percent-thick, nonlifting rectangular wing of aspect ratio 4 has been calculated by a relaxation method for a Mach number of 0.91 at an angle of attack of zero. The three-dimensional nature of the flow near the wing tip is clearly evident, and the pressures at the root are in excellent agreement with two-dimensional experimental results. This method is currently being extended to more general planforms at angles of attack. Similar analytical research is also underway at Langley Research Center.

Although significant progress has been made in mixed-flow theories, much work remains to be accomplished. For example, methods of handling arbitrary three-dimensional lifting wings with shocks are needed and ultimately should be extended to complete aircraft configurations. There also exists a need for estimating shock—boundary-layer interaction effects for both two- and three-dimensional flows. There is, of course, related experimental research that will be required, and in this regard there appears to be a need for a transonic tunnel with a higher Reynolds number capability than those presently in existence. An indication of the probable requirements for such a facility is presented in reference 6. For Mach numbers very near 1.0, tunnel boundary effects are still a problem and free-flight techniques may be required.

Leading-Edge Vortex Flow

The leading-edge separation-induced vortex flow associated with slender, sharp-edge wings is of particular interest for supersonic and hypersonic cruise aircraft with regard to take-off and landing characteristics and operation in the subsonic and transonic speed regimes although it may also be important at the supersonic design point. Regarding the apparent benefits of the vortex flow at the supersonic design point, the reader is referred to reference 7. The large increases in lift and the stable type of flow afforded by the leading-edge vortex have been known for many years and the supersonic transports flying today, the Anglo-French Concorde and the Soviet Tupolev TU-144, utilize this powerful vortex lift to eliminate the need for high-lift devices and horizontal tails although some drag-due-to-lift penalty is encountered, relative to potential flow, because of the loss of wing leading-edge suction.

Slender wings can be expected to be of continued interest for supersonic civil aircraft in the 1970's and beyond, and slender hypersonic cruise aircraft may also utilize vortex flows in the lower speed ranges since the prospects for actively cooled wing leading edges may allow the relatively sharp leading edges also desirable with regard to hypersonic drag reduction. It is, therefore, very important that improved analytical methods of accounting for the vortex type of flow be developed.

Because of the difficulty in developing analytical models of the vortex flow, slender-body approximations have, in general, been resorted to and solutions have been limited to flat wings. Credit for much of the early work belongs to the French researchers at O.N.E.R.A. (ref. 8, for example), and the most elaborate analytical model based on slender-wing theory is that developed at the British R.A.E. (ref. 9). These methods, however, are not completely appropriate for subsonic speeds and do not account for Mach number effects at supersonic speeds. Because of these shortcomings, recent analytical research at Langley has been based primarily on a leading-edge-suction analogy concept. This concept, which is described in reference 10, allows the use of potential-flow lifting-surface theories to calculate the complicated nonpotential leading-edge-separation vortex flow characteristics and avoids the limitations of slender-wing theories. The analytical methods that have resulted from this analogy have been very successful in predicting the lift and drag characteristics of flat wings over a wide range of planforms and for Mach numbers well into the supersonic range (refs. 10 and 11).

The important need at the present time is the development of analytical methods capable of both the detailed design and analysis of arbitrarily cambered and twisted wings, which include the vortex-flow effects. Both this need and the present capability to predict drag due to lift for flat wings with vortex flow can be illustrated with figure 11 where the drag due to lift is presented as a function of leading-edge sweep angle for slender, flat delta wings at a Mach number of 0.8 and a lift coefficient of 0.3. For reference, the theoretical lower bound for potential flow with planar wakes is shown, as well as a theoretical upper bound for the leading-edge separation case which accounts for the zero leading-edge-suction condition but neglects the vortex lift. The symbols represent experimental data for flat wings with vortex flow, and the large errors associated with the neglect of the vortex lift effect are very evident. The solid curve represents the theory based on the leading-edge-suction analogy, and excellent agreement is obtained. One phase of the current research is directed toward the problem of predicting the drag level for cambered and twisted wings at off-design conditions relative to the flat wings and the lower bound. However, the fact that the drag of the very slender flat wings with separated flow is below the attached-flow lower bound raises important questions with regard to the real location of the lower bound for warped wings having vortex flow. These and other questions require expanded analytical research on vortex flows. In this regard Langley has recently sponsored a preliminary study of the surface load problem (ref. 12).

CONCLUDING REMARKS

Although the discussion in this paper is limited to three general areas, it is apparent that there are many promising avenues for future research in subsonic and transonic aerodynamics applicable to civil aviation. Research on the supercritical aerodynamic concepts needs to be extended to provide the trade data necessary for design optimization. The combination of variable wing sweep and the control-configured vehicle concept may offer large improvements in the subsonic and transonic characteristics of supersonic and hypersonic cruise vehicles, and research in this area should be continued. Although much progress has been made in subsonic and transonic aerodynamic theory, an expansion of research in this area is needed to provide the improvements in theoretical methods required to take full advantage of computerized aircraft design and analysis methods. This is particularly true in the areas of three-dimensional mixed flows, shock—boundary-layer interaction, and leading-edge vortex flows.

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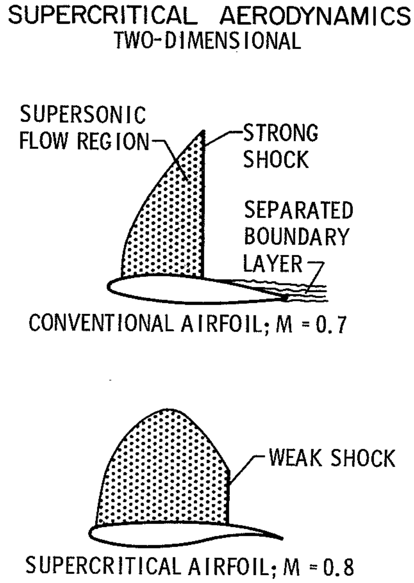


Figure 1

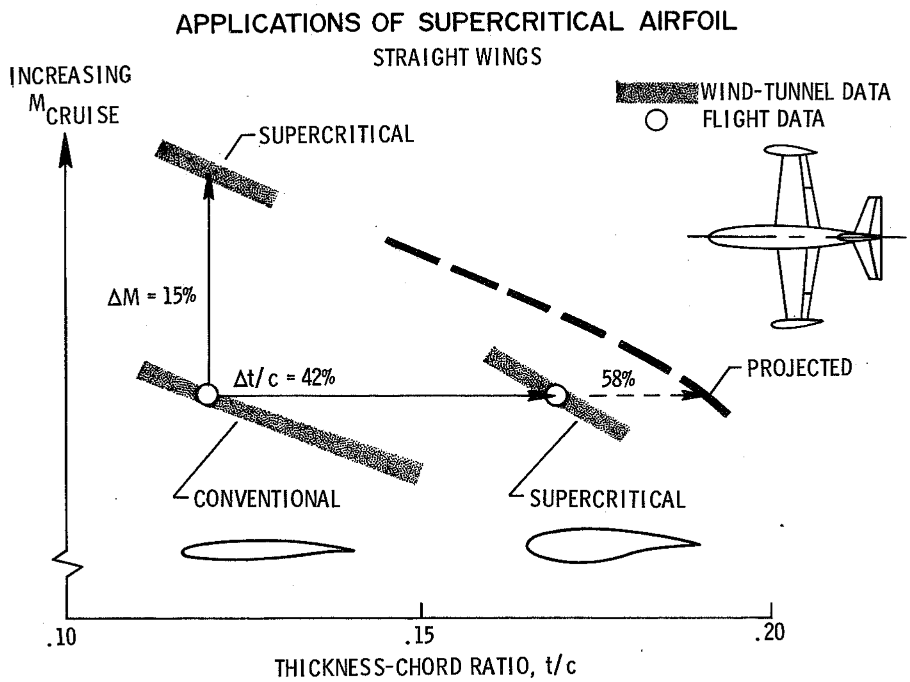


Figure 2

SUPERCRITICAL AERODYNAMICS
THREE-DIMENSIONAL

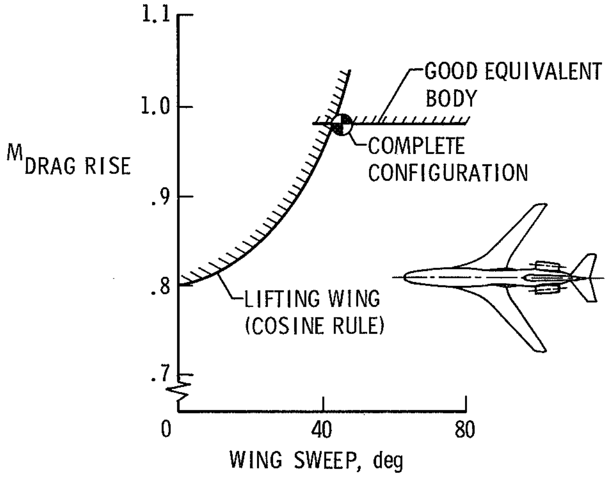


Figure 3

INCREASED CRUISE EFFICIENCY AND SPEED

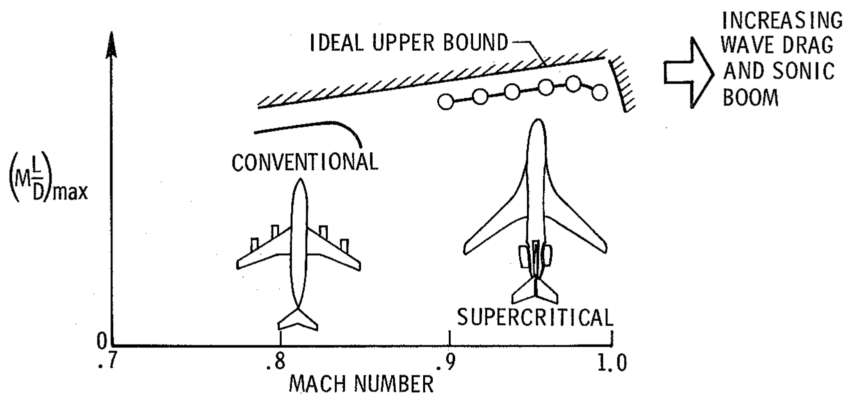


Figure 4

VARIABLE-SWEEP SUPERSONIC AIRCRAFT
SUBSONIC PERFORMANCE

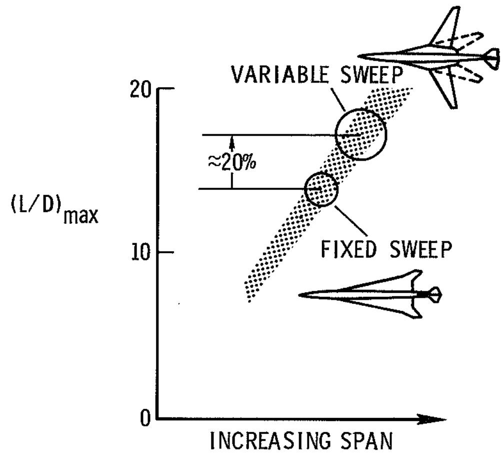
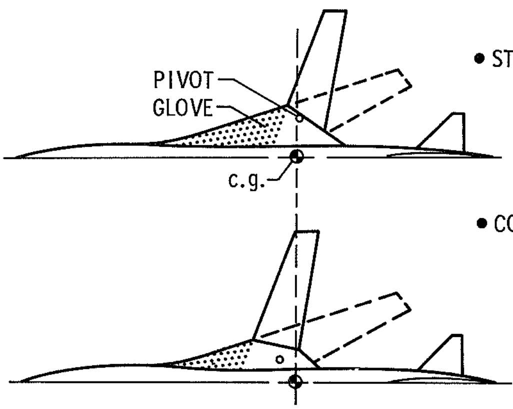


Figure 5

VARIABLE-SWEEP SUPERSONIC AIRCRAFT
SUBSONIC LONGITUDINAL STABILITY

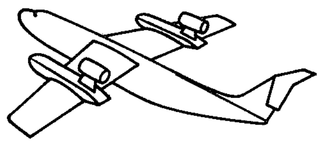


- STABLE WITH AUGMENTATION SYSTEM OFF
- CONTROL CONFIGURED CONCEPT OFFERS MORE FREEDOM IN:
  - PIVOT LOCATION
  - GLOVE SIZE
  - FLOW CONTROL DEVICES
  - HORIZONTAL-TAIL LOCATION

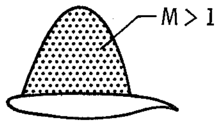
Figure 6

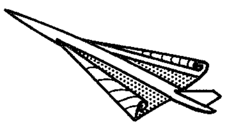
IMPROVED THEORIES FOR DESIGN AND ANALYSIS

SUBSONIC-TRANSONIC FLOW



- COMPONENT INTERFERENCE (SUBCRITICAL)
- MIXED FLOW





- LEADING-EDGE VORTEX FLOW

Figure 7

COMPONENT INTERFERENCE THEORY

SUBCRITICAL FLOW

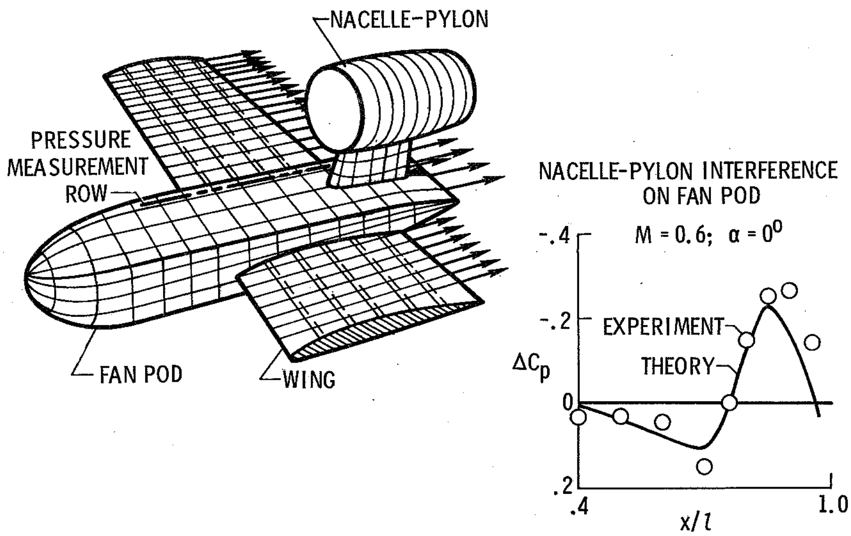


Figure 8

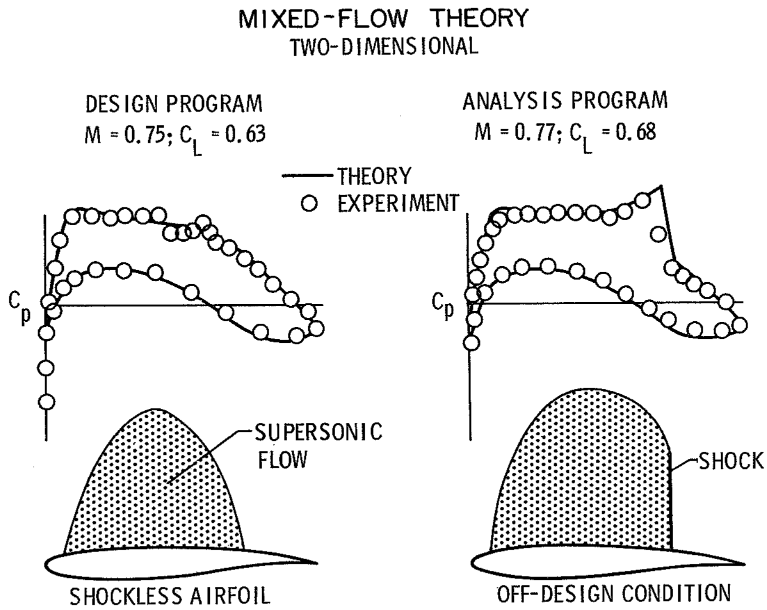


Figure 9

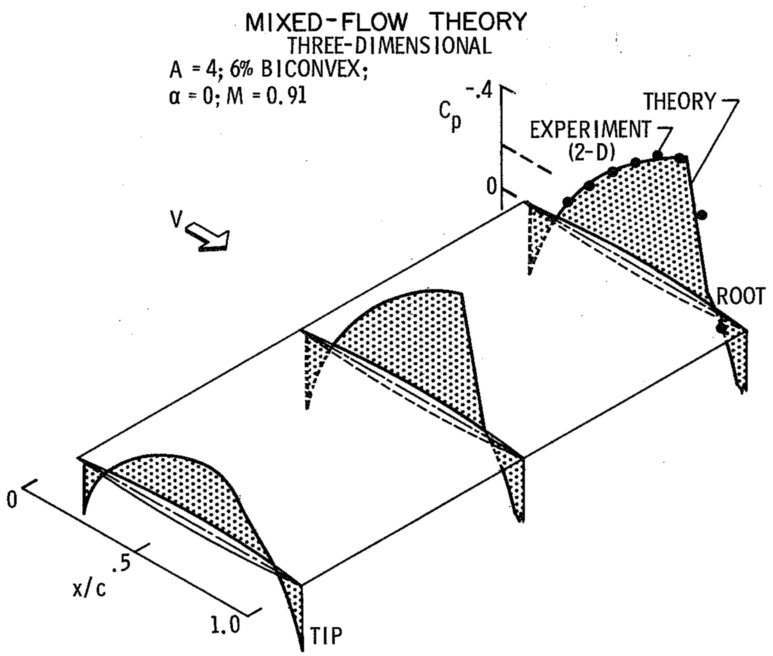
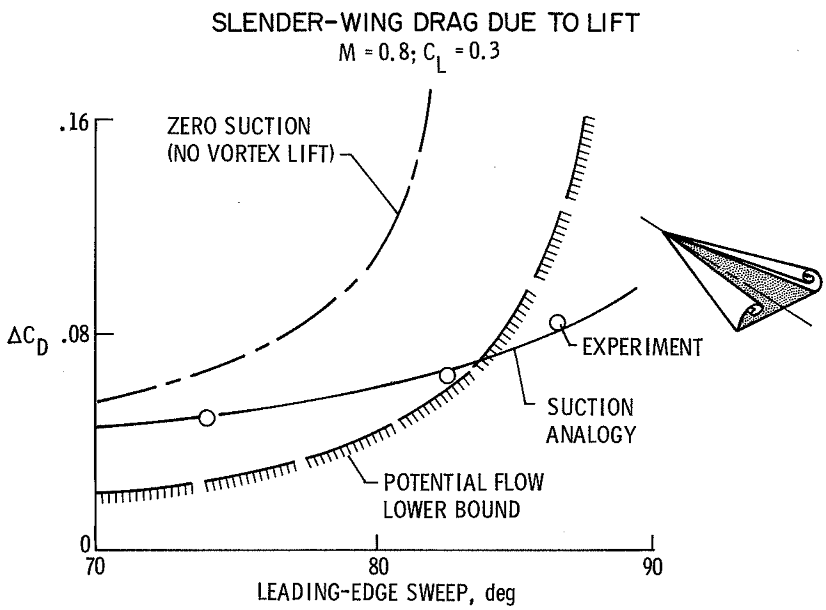


Figure 10



SLENDER-WING DRAG DUE TO LIFT
M = 0.8; C_L = 0.3
.16
.08
0
ΔC_D
ZERO SUCTION
(NO VORTEX LIFT)
EXPERIMENT
SUCTION
ANALOGY
POTENTIAL FLOW
LOWER BOUND
70
80
90
LEADING-EDGE SWEEP, deg

Figure 11

SUPERSONIC AERODYNAMIC TECHNOLOGY

By Harry W. Carlson
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SUMMARY

Supersonic aerodynamic design technology, which has advanced dramatically within the past decade, is expected to undergo continual refinement. Maximum lift-drag ratios in excess of 10 can be anticipated for large SST aerodynamic designs which have a reasonable promise for development into practical aircraft. Much progress has been made in exploration of the sonic-boom-minimization problem; however, there is not yet any development that promises sonic-boom reduction to such an extent as to permit unrestricted overland SST operations.

4

INTRODUCTION

This paper is concerned with the present status and the further improvement of supersonic aerodynamic technology applicable to civil aircraft in the Mach number range of 1.5 to 3.5. The first topic to be discussed will be advances in aerodynamic design technology for cruise-efficiency improvement and for sonic-boom minimization. Then attention will be given to improved methods for aerodynamic trades analysis which help to insure that aerodynamic factors exert their proper influence on the many compromises that must be made in the development of practical aircraft designs. The discussion will include present computer methods, their applicability and limitations, prospects for future development, and problem areas.

SYMBOLS

Values are given in both SI and U.S. Customary Units. The measurements and calculations were made in U.S. Customary Units.

C_D drag coefficient

C_L lift coefficient

C_m pitching-moment coefficient

C_n yawing-moment coefficient

C_p	pressure coefficient
h	altitude, meters (feet)
L/D	lift-drag ratio
$(L/D)_{max}$	maximum lift-drag ratio
l	airplane length, meters (feet)
M	Mach number
Δp	sonic-boom overpressure, N/m^2 (lb/ft^2)
W	weight, newtons (pounds)
α	angle of attack, degrees
β	angle of sideslip, degrees
δ_t	tail deflection, degrees

AERODYNAMIC DESIGN TECHNOLOGY

Cruise Efficiency Improvement

Dramatic aerodynamic design advances have taken place since the search for efficient supersonic transport designs was first seriously undertaken in the late 1950's and early 1960's. (See, for example, fig. 1.) The bomber configuration shown as representative of 1960 technology was considered at the time to employ some rather advanced aerodynamic design features. Foremost among these was the use of favorable interference on the wing undersurface from a large engine package installation. The reduced drag level for the present technology design stems from the selection of the highly swept arrow-wing configuration and from employment of analytic optimization methods for fuselage shaping, component arrangement, and wing camber and twist. References 1 and 2 summarize these design methods. The improvement has been made despite a greater usable volume requirement for passenger accommodation and for fuel storage. The estimated cruise lift-drag ratio for the present design is about 9.5 compared with 7.2 for the 1960 design – about a 30-percent gain.

It may be instructive to examine the typical drag contributions at the cruise design point of a high-efficiency supersonic transport (SST); these contributions are shown in figure 2. Wave drag, drag due to lift, and skin-friction drag are shown in bar-graph form. Current levels and theoretical limiting values are given. Wave drag at zero lift has already reached an almost irreducible level through application of supersonic area-rule minimization techniques. If the total configuration volume were contained in a miminum-drag body of revolution of the same length as the configuration, only a modest drag reduction could be achieved. The current values of drag due to lift have been achieved by employment of a twisted and cambered wing design based on theoretical methods and modified by wind-tunnel test experiences. A planform modification as shown in the figure would, according to theory, permit about a 30-percent reduction from current levels of drag due to lift. However, attainment of an appreciable part of this potential gain is critically dependent on compromises that must be made with structural weight and stability and control considerations. Skin friction comprises about 45 percent of the total drag. The most practical way of minimizing the turbulent skin-friction level is by reducing the wetted area by wing-fuselage blending; this blending might permit about a 5-percent reduction in skin friction. Wind-tunnel studies have indicated a possibility for a reduction of similar magnitude by injection into the boundary layer of available engine bleed air. The engineering feasibility of this approach has not yet been demonstrated. Revolutionary improvements in aerodynamic performance would result from achievement of laminar-flow drag levels. However, attempts to implement promising laboratory laminar-flow-control methods in flight-test programs have not yet proven successful even at subsonic speeds. The great potential of laminar-flow control remains a tantalizing challenge.

The national SST program has concentrated on a cruise Mach number of 2.7, and it is at this Mach number that attention to aerodynamic efficiency has been focused. The same or at least similar optimization techniques could be applied over a broad supersonic speed range, as illustrated in figure 3. Aerodynamic efficiency potential, as represented by maximum lift-drag ratio, is shown as a function of design Mach number. The maximum L/D of the $M = 2.7$ design previously discussed is shown as a point of reference. Aerodynamic considerations clearly show the superiority of an arrow wing with a leading edge swept behind the Mach line. With such a design, the zero-lift wave drag may be kept low and wing camber and twist may be employed to avoid the otherwise high drag due to lift of slender wings. However, low-speed landing and take-off considerations and noise restrictions introduce minimum span constraints, and this design philosophy cannot be employed at high supersonic speeds. This consideration is responsible for the inflection in the curve. In contrast to their cruise aerodynamic benefits, arrow wings are relatively inefficient from the standpoint of structural design and wing weight and also present problems of low-speed stability and control. Thus, compromises are required and the final design often tends to approach the modified delta-wing arrangements corresponding to the lower part of the $(L/D)_{max}$ band. A discussion of the many factors affecting the choice of wing planform and of cruise Mach number is given in paper no. 24 by Nichols, Keith, and Foss.

Sonic-Boom Minimization

Sonic boom continues to be one of the most severe problems in the development of a completely acceptable supersonic transport. As illustrated in figure 4, current design practices, which are not influenced to any large degree by boom considerations, would produce N-waves with nominal overpressures of about 96 N/m^2 (2 lb/ft^2) for large SST's with cruise weights in the 2.67 MN (600 000 lb) range. Theoretical studies, verified in wind-tunnel experimental programs, have indicated a potential for attainment of flat-top signatures with overpressures less than one-half the current design level by means of airplane shaping (primarily a redistribution of fuselage cross-sectional area). When tempered with practical design considerations, more realistic goals, which are still difficult to achieve, might be as designated by the practical minimum curve. As can be seen, weight reduction is a direct means of sonic-boom reduction.

As shown in figure 5, airplane length also can have a powerful effect on the sonic-boom characteristics. The upper curve shows the variation of overpressure with airplane length for a configuration shaped to yield flat-top signatures. The solid symbol represents the same point as shown on the lower curve of figure 4. A different shaping could be employed to reduce the shock strength at the bow and tail of the signature; peak overpressures in this case would be somewhat higher. The real significance is that as indicated for lengths in excess of 122 meters (400 ft), the shock strength could be reduced to zero, and at least for outdoor exposures, the noise would be expected to be imperceptible. These theoretical data which are based on the most advanced optimization techniques and take into consideration the most recent knowledge of atmospheric propagation factors were presented in a paper on sonic-boom minimization given by R. Seebass at The Second Sonic Boom Symposium held during the 80th Meeting of the Acoustical Society of America in Houston, Texas, November 1970. In assessing the implications of the data of figure 5, it must be stressed that the curves are based on idealized theoretical concepts and that no account has been taken of the inevitable weight increases that would accompany increased length. Thus, the length requirement shown in figure 5 would be inadequate for practical application of the concept.

Consideration has been given to means of increasing the effective airplane length by use of retractable fins at the nose and tail. (See fig. 6.) Because of the inclination of supersonic-flow shock systems, the tip of the fin produces an effect similar to that of an extended nose. To avoid excessive weight penalties, the thermal fin concept would make use of a heat field created by burning part of the onboard fuel to affect the surrounding air in much the same way as would properly shaped but quite large fins. The feasibility of such a concept is being explored in a NASA contract with the Aerospace Corporation.

A review of the status of sonic-boom research as of October 1970 is presented in reference 3.

Technology Projection

As indicated in figure 7, the aerodynamic design technology which has advanced so dramatically within the past decade is expected to undergo continual refinement. The estimated cruise lift-drag ratios shown are based on extrapolation to full-scale conditions of validated wind-tunnel data for configurations that show a reasonable promise for development into practical airplane designs. The projection to ratios of lift to drag in excess of 10 is based on a further growth in airplane size to nearly all wing designs with wing areas larger than 930 m^2 (10 000 ft^2). It is also dependent on utilization of the best available theoretical aerodynamic optimization techniques and a fine tuning of the design through carefully planned experimental programs. No account has been made possible of improvements through use of laminar-flow control. Estimated sonic-boom levels for advanced SST's with the configurations shown are also given. It is difficult to make a projection of boom levels for the advanced SST, because of the unknown extent to which boom-minimization techniques discussed herein can be applied in practical airplane design. Although further progress in sonic-boom research can be anticipated, there is not yet any development that promises sonic-boom reduction to such an extent as to permit unrestricted overland SST operations.

AERODYNAMIC TRADES ANALYSIS

Present Methods

As previously mentioned and as illustrated in figure 8, there must of necessity be a large degree of compromise in supersonic cruise vehicle design. Although requirements for cruise aerodynamic efficiency can be generally well established, a host of other important design requirements including low-speed performance, structural efficiency, and payload capacity drive the resultant configuration well away from an aerodynamic optimum. Thus, the aerodynamicist must be concerned not only with the improvement of the aerodynamic potential but also with provision of information necessary in the conduct of valid trade studies. Computer-implemented evaluation techniques serve in this task by providing initial assessment of aerodynamic characteristics for many candidate configurations and by permitting a day-to-day evaluation of the aerodynamic effects of configuration changes made in response to other requirements.

A simplified schematic drawing of a system of Langley computer programs which are also representative of those in use throughout the industry is shown in figure 9. In usage, a numerical model is prepared from the aircraft description, and geometry programs provide the particular numerical representation required for the aerodynamic programs. Skin-friction drag is evaluated by use of a surface-strip technique, drag due to lift by a linearized-theory lifting-surface approach, and zero-lift wave drag by application of supersonic area-rule equivalent-body methods. Effective-area developments from the lift and wave-drag programs may be used in a sonic-boom program for the calculation of theoretical signatures. The combined program results provide theoretical aerodynamic characteristics. Although not shown, the lift and the wave-drag programs have options for direct optimization of the lifting surface and the fuselage shape, respectively. Computer-generated tapes can be used in the programing of machine tools for construction of model components for wind-tunnel tests. Similar tapes are used to program automatic plotting machines to provide a graphic representation of the configuration being studied.

The computer graphics capability is illustrated in figure 10. These drawings are useful in assessing the adequacy of the airplane representation and in checking for obvious errors in input data. As seen in the drawing, a considerable degree of airplane complexity may be taken into account; nevertheless, the thin-wing, slender-body assumptions on which the underlying theories are based tend to limit applicability of the computer methods to reasonably efficient cruise vehicles. A more complete description of these computer methods is given in reference 4.

Applicability and Limitations

The applicability of computer-program techniques to the evaluation of aerodynamic performance has been demonstrated in numerous correlations with wind-tunnel data such as that shown in figure 11. In figure 11(a), lift-drag curves are shown for an SST configuration at its cruise Mach number of 2.7. Generally for vehicles of this class, which must of necessity have thin wings and slender fuselages for high aerodynamic efficiency, the methods provide quite accurate predictions. The data shown in figure 11(b) are for a fighter configuration. It is anticipated that the degree of correlation shown here would be representative of that attainable for supersonic business jets.

Progress in adapting linearized-theory lifting-surface methods to stability and control analysis is illustrated in figures 12 and 13. In figure 12 a comparison on the experimental and theoretical effect of horizontal-tail deflection on model pitching moment is shown. At present, application of the analytic techniques has not been consistently successful as witnessed by the good correlation for the low-tail configuration and the poor correlation for the high-tail configuration. Application of similar techniques to directional characteristics is shown in figure 13. Yawing moment is shown as a function of sideslip angle. For the simple research model with the single vertical fin, the correlation is good. For the more complex model with twin vertical fins, the correlation is poor. Further work is required to refine the methods and to establish the extent of their applicability. Some factors which might affect the applicability of the methods will be discussed subsequently.

Future Development

Results from an advanced linearized-theory computational technique (ref. 5) which provides more complete loads information is shown in figure 14. The method has the same theoretical basis as the previously discussed techniques, but surface panels rather than mean surfaces are employed and account is taken of wing-fuselage interference. A good correlation of the experimental and theoretical surface pressures is shown not only on the wing but also on the fuselage and in the wing-fuselage juncture region. It is anticipated that further development and the introduction of new computers will permit removal of the present restriction to a cylindrical fuselage and will permit consideration of more than 100 fuselage and 100 wing panels for a better definition of complex airplane configurations.

Much work is being directed toward the development of more exact aerodynamic analyses than can be provided by linearized-theory methods. Many of the efforts are concerned more directly with the hypersonic-speed regime and with blunt high-volume configurations. However, techniques which solve those more difficult problems could also be expected to offer an improved capability for supersonic speeds. One higher order theory program (ref. 6) now under development is illustrated in figure 15. In this method, a fine-mesh-grid system surrounds the shape to be analyzed and finite-difference numerical techniques are employed to solve sets of nonlinearized fluid-motion equations. The method provides a definition not only of the surface pressures and forces but also of the entire flow field. A "capturing" technique is used to locate and define the strength of any shocks that may be formed. At present, the method has been applied only to rather simple shapes, with primary attention being given to the sonic-boom problem. A number of advanced theoretical techniques are discussed in reference 7.

Problem Areas

Problem areas for which adequate analytic techniques have not yet been developed are illustrated in figure 16. These factors have little influence on large SST cruise point design but could be of importance under other circumstances. A detached vortex system on a delta wing and separated-flow regions in the vicinity of a nacelle installation are illustrated in a somewhat exaggerated fashion. As shown in this figure, an unstable boundary layer may separate and cause shock formation or an impinging shock may trigger separation. Shock—boundary-layer interaction and flow separation are important factors for less slender vehicles, such as fighters and business jets, and could affect SST's in off-design conditions. Although certain specific problems may be treated, no generally applicable methods of handling these complex flow interactions have yet been developed. A proper handling of the detached-vortex system is believed to be essential in the development of broadly applicable stability and control analysis methods. Some progress has been made in the prediction of the effects on lift and drag of detached vortices by means of the leading-edge suction analogy, which is discussed in paper no. 3 by Edward C. Polhamus, but much work remains to be done in detailed vortex description.

CONCLUDING REMARKS

Supersonic aerodynamic design technology, which has advanced dramatically within the past decade, is expected to undergo continual refinement. Maximum lift-drag ratios in excess of 10 can be anticipated for large SST aerodynamic designs which have a reasonable promise for development into practical aircraft. Much progress has been made in exploration of the sonic-boom-minimization problem; however, there is not yet any development that promises sonic-boom reduction to such an extent as to permit unrestricted overland SST operations.

A critical factor in the development of efficient aircraft designs is providing aerodynamic trades data to insure that aerodynamic factors exert their proper influence on the many compromises that must be made. Current computer methods provide a reasonable handling of performance aerodynamic factors, but further development is required for stability and control analysis. Wing vortex patterns and separated-flow regions remain as problem areas which resist analytic treatment and prevent adequate analysis of vehicles less slender than SST's and of certain off-design conditions even for SST's. Generally, though, supersonic aerodynamic technology applicable to civil aircraft is in a relatively advanced state of development, and the greatest concern is the lack in this country of a development program in which it can be put to use.

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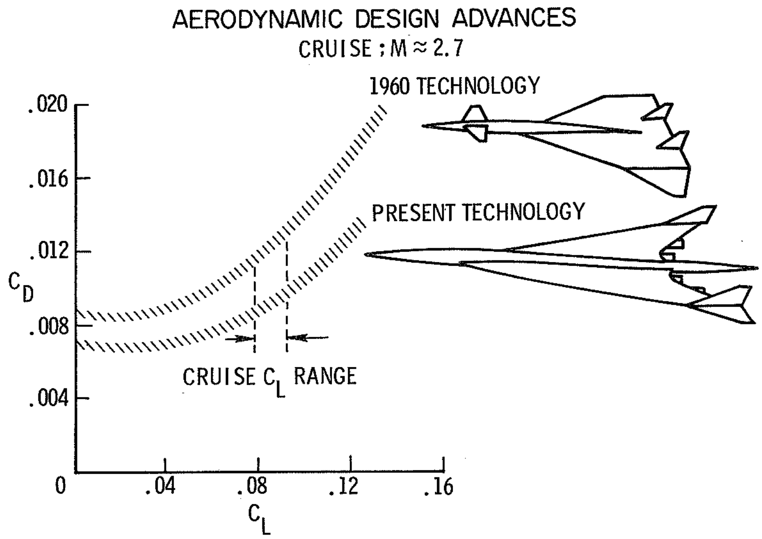

Figure 1

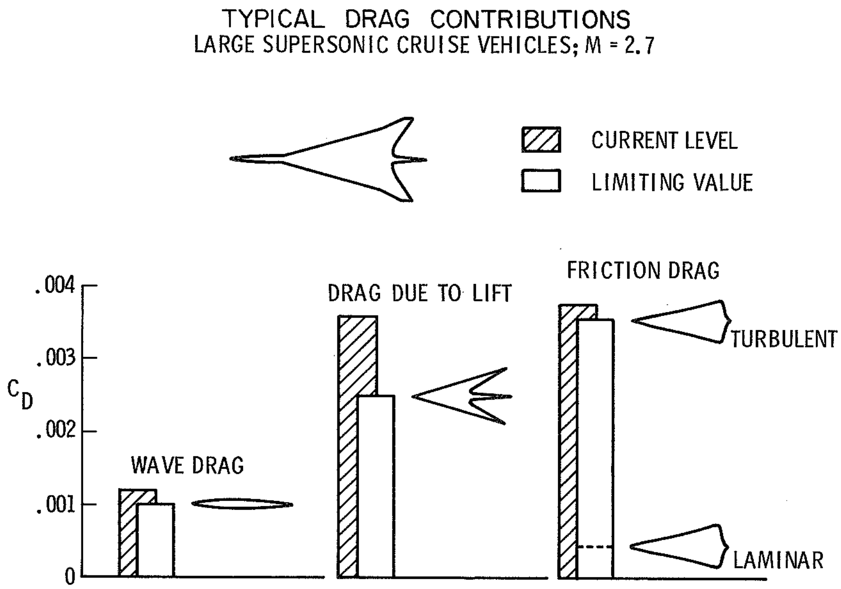

Figure 2

AERODYNAMIC EFFICIENCY POTENTIAL

LARGE SUPERSONIC CRUISE VEHICLES

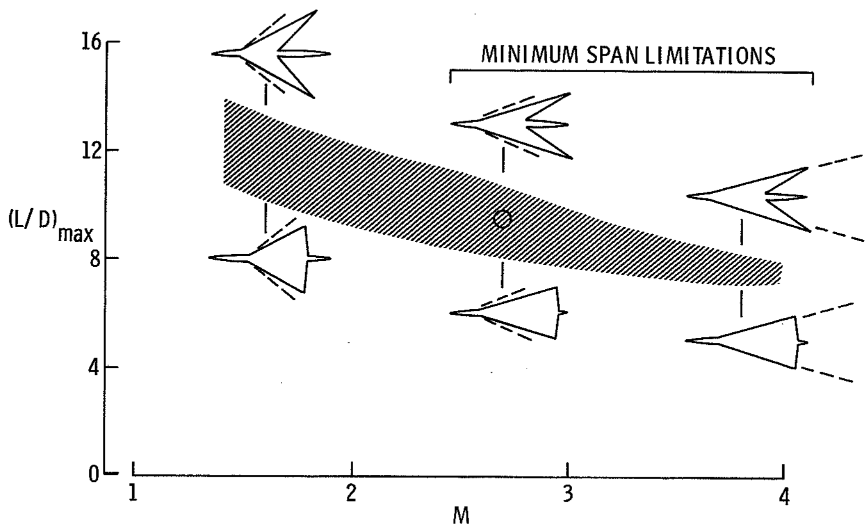

Figure 3

AIRPLANE SHAPING FOR SONIC-BOOM REDUCTION

M = 2.7; h = 18 km (600 000 ft); l = 91 m (300 ft)

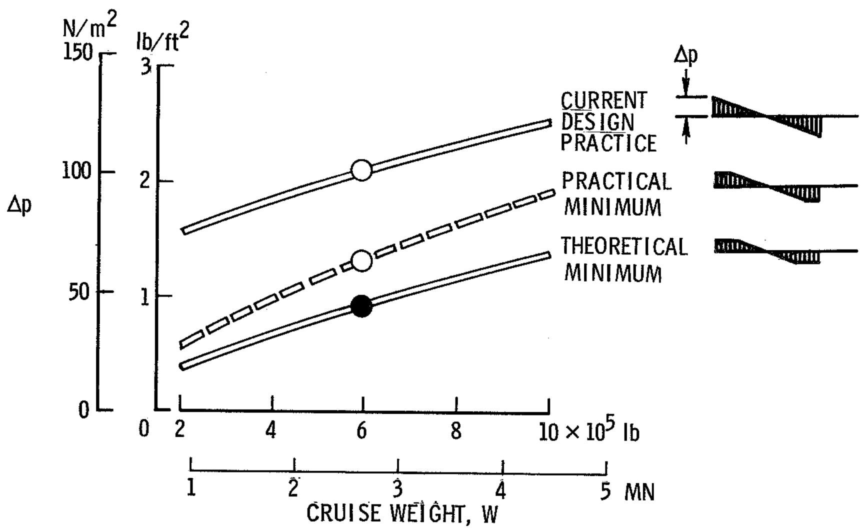

Figure 4

AIRPLANE LENGTHENING FOR SONIC-BOOM REDUCTION

M = 2.7; h = 18 km (600 000 ft); W = 2.67 MN (600 000 lb)

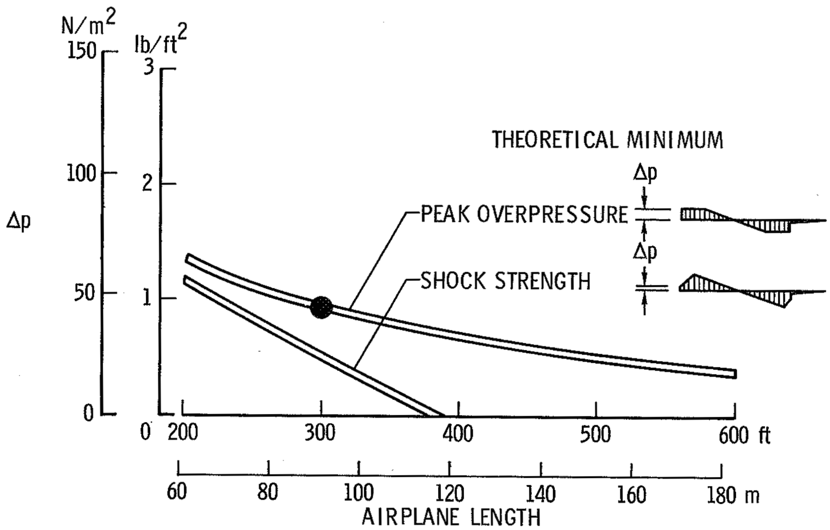

Figure 5

THERMAL FIN CONCEPT

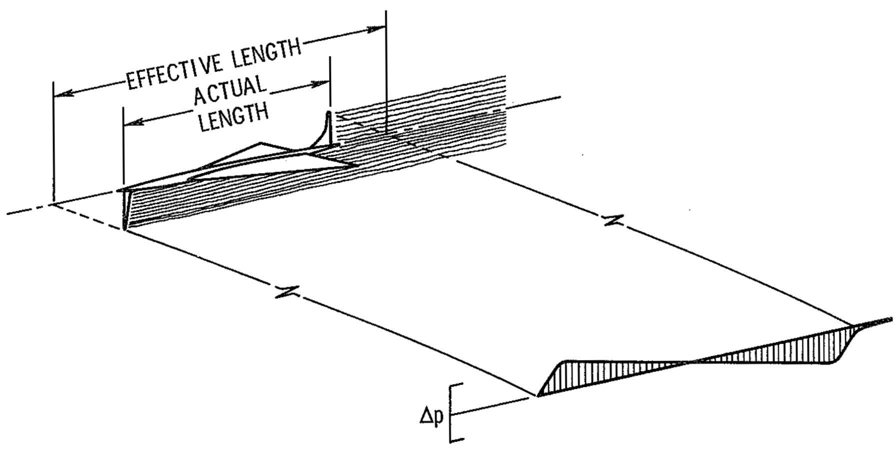

Figure 6

PROJECTION OF AERODYNAMIC EFFICIENCY IMPROVEMENTS

LARGE SUPERSONIC CRUISE VEHICLES; M = 2.7

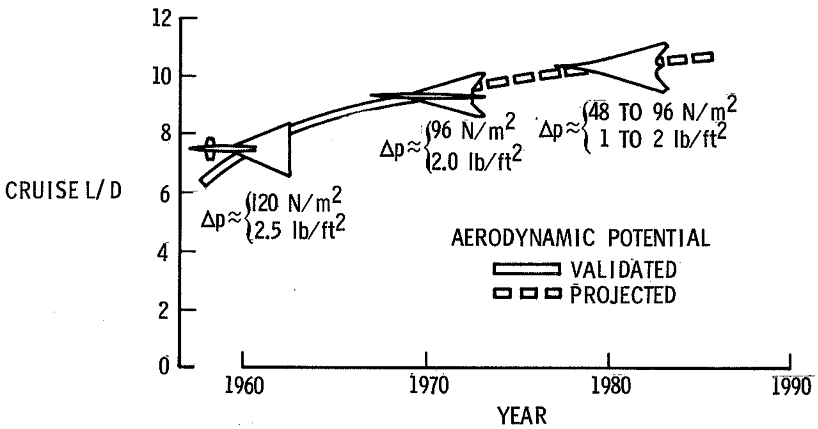

Figure 7

COMPROMISE IN DESIGN OF SUPERSONIC CRUISE VEHICLES

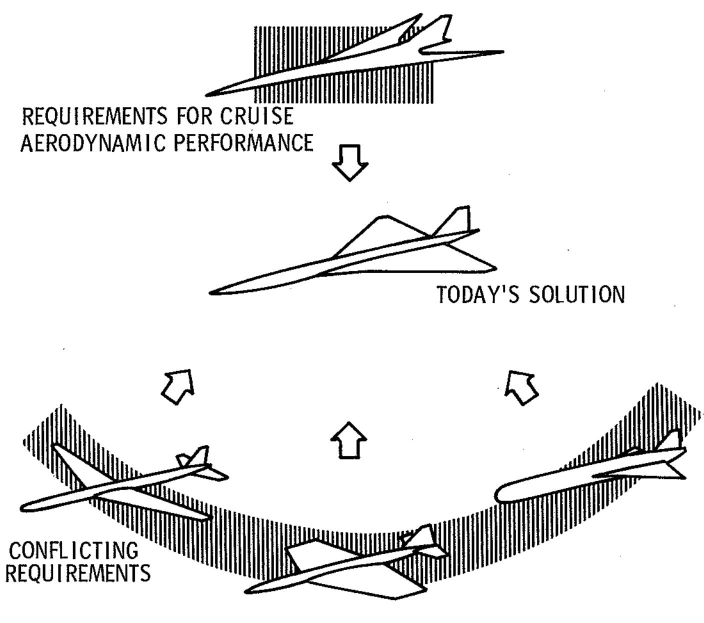

Figure 8

COMPUTER-IMPLEMENTED ANALYTIC TECHNIQUES

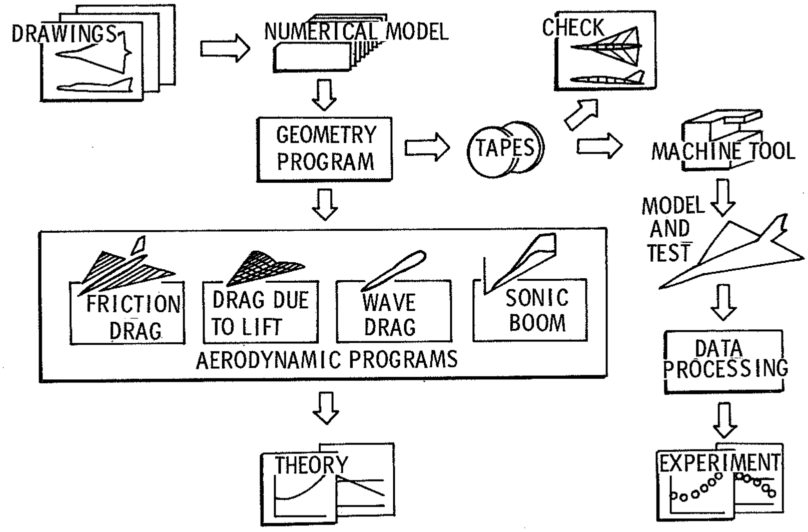

Figure 9

COMPUTER GRAPHICS

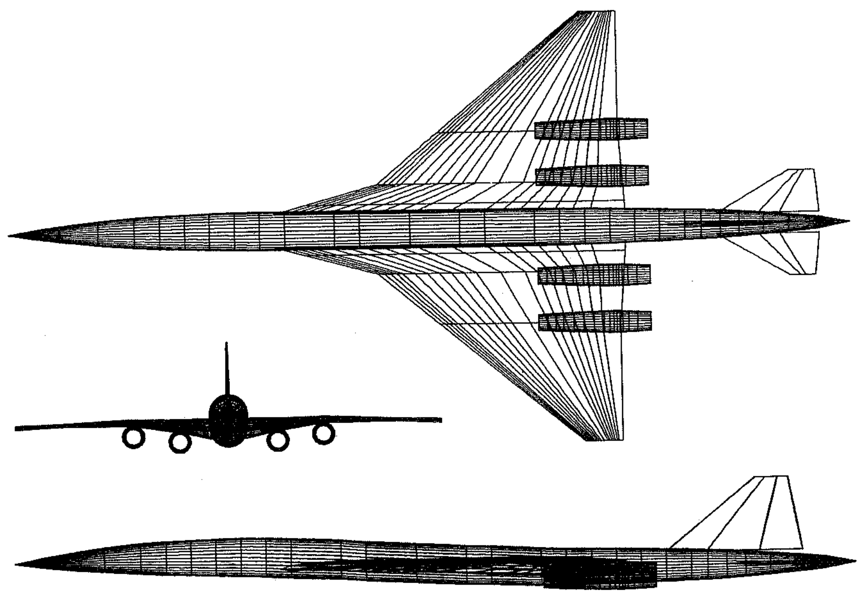

Figure 10

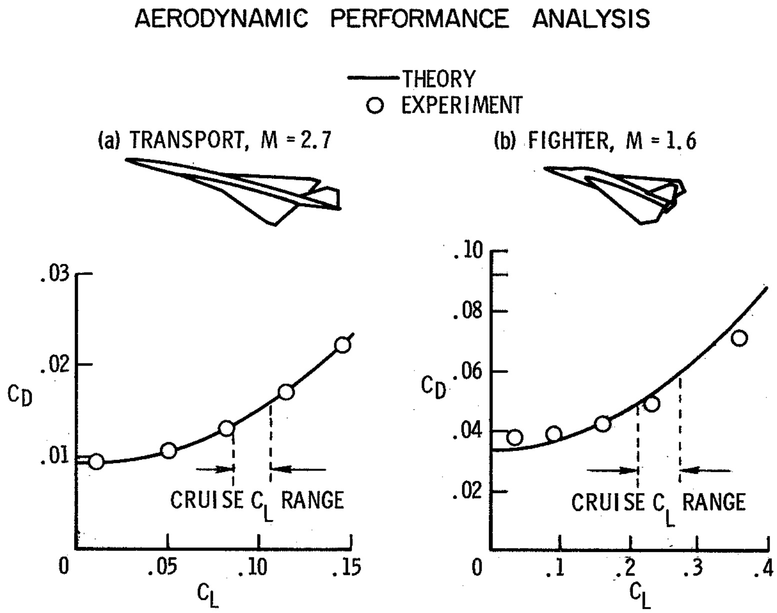

Figure 11

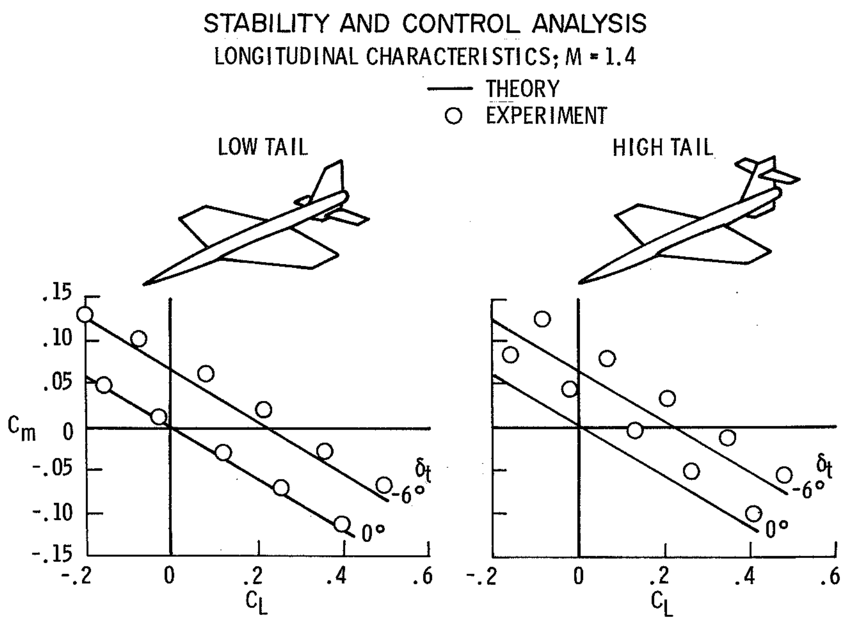

Figure 12

STABILITY AND CONTROL ANALYSIS

DIRECTIONAL CHARACTERISTICS; M = 2.8

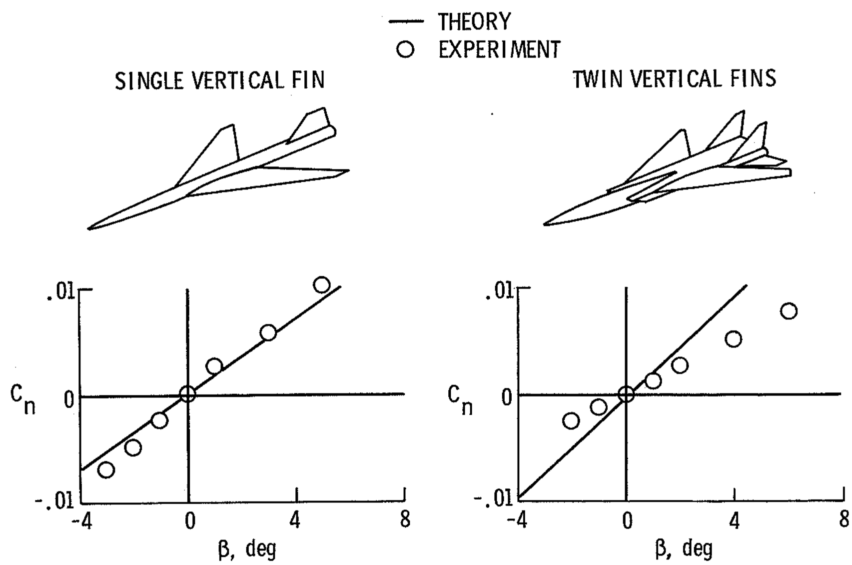

Figure 13

LINEARIZED-THEORY SURFACE-PANEL METHOD FOR MORE COMPLETE LOADING INFORMATION

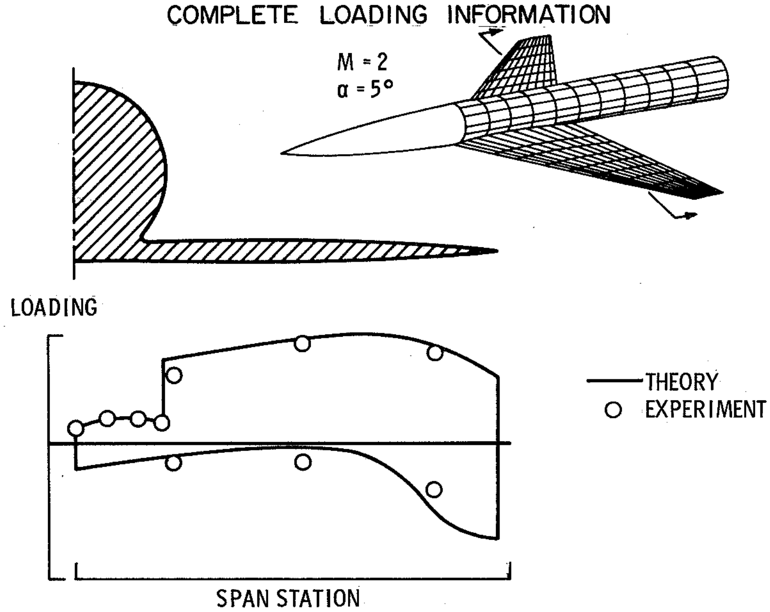

Figure 14

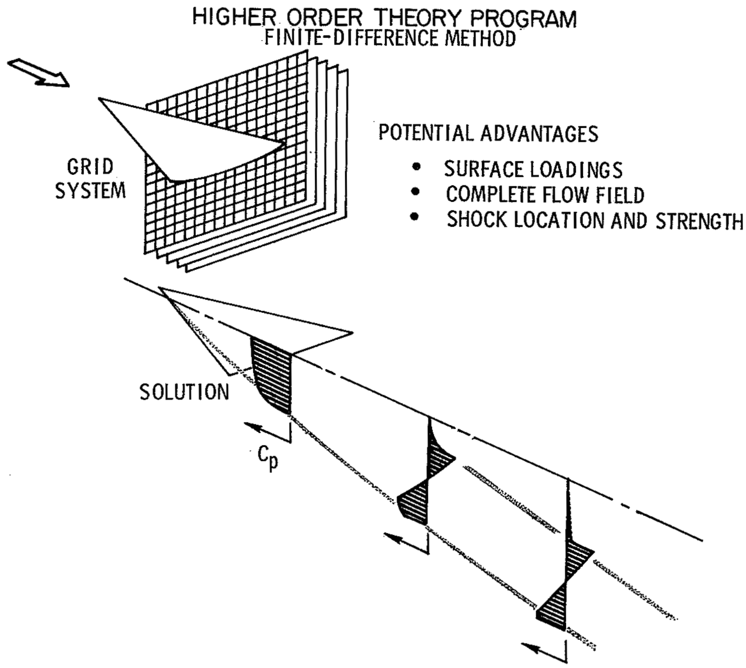

Figure 15

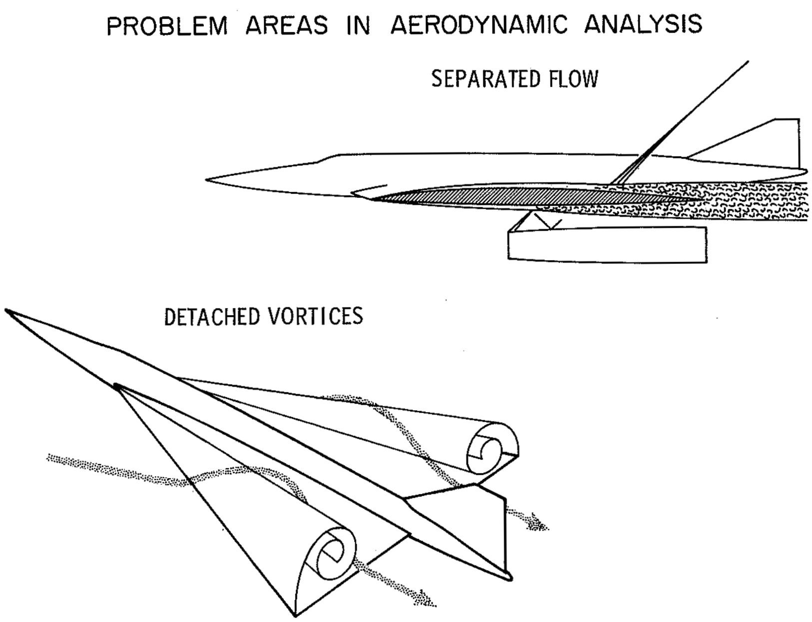

Figure 16

HYPERSONIC AIRPLANE AERODYNAMIC TECHNOLOGY

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SUMMARY

This paper reviews the status of hypersonic aircraft aerodynamic technology and pinpoints critical research areas for the 1970's. The outlook for numerical flow-field calculations and recent research in the problem areas of engine-airframe integration, slot injection as an active airframe cooling technique, lee-side vortex heating, and hypersonic boundary-layer transition are considered.

It is concluded that with realistic progress in aerodynamic technology, an efficient hypersonic cruise aircraft can be developed. The most critical area for research is the integration of the air-breathing propulsion system with the lower surface of the aircraft, since the forebody compression surface acts as an inlet spike and the lower part of the vehicle afterbody acts as an exhaust nozzle for the propulsion modules. Also a significant gain in payload can accrue from the further development and application of active cooling technology for both external and internal cooling concepts.

The development, within the next 3 years, of numerical solutions of the complete inviscid and viscous flow equations over real wing-body configurations should allow optimization of the aerodynamic configuration and should result in much less design uncertainty for thermal-protection systems. Because of the influence of real-gas effects upon the expansion of the engine exhaust over the aircraft afterbody, accurate numerical flow-field solution techniques are essential to the development of an efficient hypersonic cruise aircraft.

INTRODUCTION

Hypersonic airplane aerodynamic technology dates from the late 1940's, and therefore the research field approaches the quarter-century mark. Research conducted in the first 10 years involved problem definition and exploratory studies usually associated with a new field of endeavor. The advent of the X-15, Dyna Soar, and intercontinental ballistic missile projects, with the resultant need for design methodology at Mach 6 and above, forced the research into a more applied vein in the middle to late 1950's. During this time some of the research effort aimed at slender, aircraft-type configurations was redirected into support for technology involved in the manned space missions (Mercury, Gemini, and Apollo) which employed very blunt

5

configurations. Also, the applied research was generally directed at "hot structures" (such as ablators or heat sinks).

The lack of a follow-on technology demonstrator program to replace the X-15 program forced the hypersonic aircraft aerodynamic technology field into a period of more fundamental, rather than applied, research during the middle 1960's. Therefore, in contrast to the subsonic and supersonic technologies in which considerable effort is currently involved in improving existing systems, hypersonic airplane aerodynamics is in the basic-technology and concept-definition stages.

There has, however, been a change in emphasis in the basic work. Previous to the early 1960's, the theories available consisted primarily of solutions for highly idealized conditions, a situation necessitated by the inability of the available mathematical methods or digital computers to solve any except the simplest flow equations. Therefore, the ground facilities were used to duplicate, as closely as possible, the expected flight conditions. The experimental results, in most cases, were directly extrapolated to flight. The development of rapid digital computers with large-core storage changed this situation. With the present generation of computers, it is possible to solve general two-dimensional viscous flow problems, even where "elliptic" or separated-flow regions are involved. As the digital machines develop further, this capability can be extended to three-dimensional-flow situations about actual vehicles. The experimental facilities will therefore be used more and more to generate critical and detailed flow-field data to check and develop the numerical solutions. These calculation methods would then be used for flight prediction rather than the extrapolation of experimental results.

Shown in figure 1 is a sketch of a typical vehicle currently under consideration for the hypersonic cruise mission. To increase the aerodynamic efficiency for cruise (thermal efficiency rather than aerodynamic efficiency dictated the design of the blunt manned reentry shapes used in the 1960's), these vehicles are slender and highly blended with swept surfaces.

General features of the hypersonic cruise aircraft are

(1) Mach numbers from 5 to 12
   (a) High heat transfer
   (b) Low angle of attack ($L/D \rightarrow 4$ to 6)
   (c) Large engine airflow

(2) Hydrogen fuel
   (a) Large vehicle
   (b) Turbulent flows (large Reynolds number)

(3) Closely integrated vehicle
  (a) Aerodynamics
  (b) Propulsion
  (c) Structures

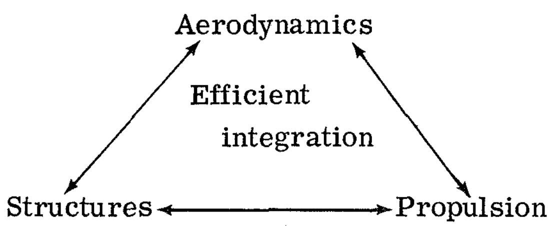
Since prospective flight Mach numbers range from 5 to 12, high aerodynamic heating is present. Coping with this heat transfer is as important a consideration as the more usual problem of attaining a high lift-drag ratio.

To gain back the performance loss associated with the lower $(L/D)_{max}$ at hypersonic speeds (compared with low-speed and supersonic values), the vehicles use hydrogen, a higher energy fuel. The use of hydrogen allows operating ranges on the order of 8000 kilometers (5000 miles). (See paper no. 25 of the present compilation for further details concerning the operational characteristics of hypersonic aircraft.) Because of the necessity of storing the hydrogen (as a cryogenic liquid), the vehicles to maintain high L/D must have large volume and are therefore quite long (>91 meters or 300 feet). Hence, full-scale Reynolds numbers are as high as 300×10^6 and, consequently, the viscous flows over the vehicle are primarily turbulent. Major gains have been made in recent years in understanding and predictive capability in the area of hypersonic turbulent boundary layers. This capability has significantly increased the aerodynamic data base from which feasibility of a hypersonic transport is ascertained.

Much more so than at lower speeds, a hypersonic cruise vehicle must be designed with intimate integration of the propulsion, aerodynamic, and structural aspects. Figure 2 depicts the propulsion-airframe integration problem. Rather than using separate engine pods with attendant large structural weight and cooling requirements, the engine is tucked into the underside of the vehicle. The vehicle nose acts as an inlet spike, the combustor being the only part of the engine contained within a housing. The under-expanded supersonic combustor flow then expands over the rear of the vehicle, as indicated in figure 2, and produces thrust on the aft end. Advantages of this engine-airframe integration include lower structural weight and cooling requirements and the possibility of vectoring the vehicle thrust to increase the lift coefficient. Further discussion of hypersonic cruise mission requirements and trade-offs is available, for example, in reference 1 and the references contained therein.

The major aerodynamic problem areas for hypersonic vehicles at cruise conditions to which the supporting aerodynamic technology must direct itself are

(a)



(b) Advanced numerical computational methods

(c) Active cooling techniques

(d) Interference and upper surface vortex flows

Obviously, a hypersonic vehicle must also pass through the low and supersonic speed ranges as well as land and take-off. Therefore, there are additional important problem areas (not shown) which occur at lower speeds. These areas are concerned primarily with resultant aerodynamic forces associated with flying at relatively low speeds with a configuration designed for most efficient flight in the hypersonic corridor. The resolution of these additional problems will probably involve the use of some geometry change, such as movable canards. In the present paper only the aerodynamic problems associated with hypersonic cruise conditions are considered.

A major technological problem involves the efficient integration of the propulsive, structural, and aerodynamic aspects. The aerodynamic and propulsion integration problem previously discussed indicated that the vehicle takes on the aspect of a flying engine centerbody. (See ref. 2.) Structurally, sufficient information is not yet available to determine the structural weight fraction with any degree of certainty for application to flight in a hypersonic cruise environment.

Because of large real-gas effects upon the afterbody expansion of the combustor exhaust flow, solutions to the engine-airframe integration problem will probably have to come from advancements in numerical flow-field computation methods. Also, once that more complete computational methods are available, automated vehicle design methods can be employed (such as discussed by Harry W. Carlson in paper no. 4).

Recent systems studies (refs. 3 and 4) indicate that actively cooled hypersonic vehicles may be more structurally efficient than those designed by use of the usual uncooled or "hot" structural approach. Necessary questions then arise as to choice of type of cooling system, internal or external, and the practical aspects of cooling system design. The design of such a cooling system is another problem area where the advanced numerical computational methods provide a faster and more efficient method of attack than the conventional combination of "cut and try" empiricism with approximate "theory." Significant progress has been made in recent years in this area of active cooling techniques.

The detailed design of a vehicle cooling system is complicated by the presence of interference heating associated with shock impingement (ref. 5) and interaction with the surface of longitudinal vortex flows over the lee side (refs. 6 and 7). The magnitude and extent of this interference heating must be known, the condition eliminated, or the condition alleviated before an efficient cooling system can evolve.

The present paper discusses significant recent research developments and approaches in the problem areas shown. Additional subjects considered include the status of hypersonic boundary-layer-transition research and the comparative aerothermal environment between the hypersonic cruise and space shuttle missions.

SYMBOLS

L/D	aerodynamic lift-drag ratio
M_∞	free-stream Mach number
M_e	local Mach number
p'	root-mean-square static-pressure fluctuation
$\bar{p}$	mean static pressure
s	slot height
T_w	adiabatic wall temperature
T_t	isentropic stagnation temperature
U	velocity
U_e	external velocity
x	surface distance in main flow direction
Λ	sweep angle
α	angle of attack

Subscript:

max	maximum value

OUTLOOK FOR PREDICTION METHODS

The basic equations which describe the inviscid and viscous flow about hypersonic configurations are a coupled set of nonlinear (primarily due to the advection terms) partial differential equations (ref. 8). No general analytical solutions exist for these equations and in the past the usual approach has been to simplify the equations by suitable linearization and approximation until a set of equations was obtained which were amenable to an analytical solution. Hopefully, one obtained a set of equations which were not only solvable, but which described a meaningful physical situation, that is, resulted in a reasonable approximation to the actual flow situation. It has often been difficult to assess accurately the consequences of the various linearization and approximation techniques employed.

Within the last decade, another approach, that of numerical solutions to the complete equations, has become increasingly useful. This general research field, termed "computational fluid mechanics," has enjoyed an almost explosive growth since the advent in the middle 1960's of the large-scale digital computers (for example, refs. 9, 10, and 11). These machines allowed the solution of fairly general two-dimensional problems and a limited but useful number of three-dimensional problems. The next generation of digital computers, with greatly increased storage capacity, should be available by 1973 and should provide an order-of-magnitude reduction in computational cost. Also, the increase in core memory of these machines will allow solution of three-dimensional flow problems for general configurations. Published computational fluid-mechanics research papers particularly applicable to hypersonic aircraft configurations include references 12, 13, 14, and 15. In reference 12 the three-dimensional supersonic and hypersonic small-disturbance inviscid equations (which are valid at small angles of attack) are solved for general vehicle geometry. In reference 13 the complete inviscid equations are solved for bodies with regular geometry. The major assumptions in reference 13 are (1) spherical nose geometry and (2) supersonic or higher speed afterbody flow. The first of these assumptions could be relaxed by using reference 16, which is a solution to a general three-dimensional nose region. The second assumption is basic to the solution technique, a marching procedure for the steady supersonic inviscid-flow equations. This simplification allows a three-dimensional solution to be obtained with reasonable simultaneous storage requirements. For mixed supersonic and subsonic afterbody flow, the flow field becomes elliptic and simultaneous storage is required for all body stations. The solution to three-dimensional elliptic problems may become possible by using the next generation of machines. However, there are certain approximations (such as the method of lines, ref. 17) which could be used for high angles of attack where subsonic regions are encountered. For hypersonic cruise configurations, fortunately, the angle of attack is generally small so that mixed flow regions are not often encountered.

For the viscous flow problem, solutions (such as refs. 14 and 15) are available for the complete two-dimensional boundary-layer equations. This work is being extended to the general three-dimensional case.

A prognosis for the development of the numerical predictive capability is as follows:

Last 5 years:

(a) General two-dimensional boundary layers

(b) Combinations of approximate methods } Real configurations

(c) Small disturbance – inviscid flow } Real configurations

Next 1 to 3 years:

(a) Inviscid flow field – real configurations with supersonic afterbody flow

(b) Lee-side vortex and engine expansion flows

(c) Complete three-dimensional boundary layers

Far down the road – next generation and beyond computer:

(a) General three-dimensional viscous flow field – empirical turbulence models

(b) Time integration of complete equations to compute turbulent flow

With the existing high-speed machines, it will be possible to solve, within the next 2 years or so, the complete three-dimensional boundary-layer equations and inviscid flow about general configurations with predominantly supersonic afterbody flow. As the large storage machines become available, these solutions will be more economical (by almost an order of magnitude in dollars).

Another area which will aid the hypersonic cruise mission designer is a solution for the interaction of the lee-side vortex flow with the vehicle surface. See section entitled "Lee-Side Vortex Problem." As recognized by Spalding (ref. 18), as long as there is no recirculation against the main flow, it is possible to neglect the diffusion terms in the streamwise direction (the diffusion terms being kept in the cross-sectional plane). Therefore, one can solve by a marching procedure (with low storage requirements) a corner-flow or an afterbody-flow problem where longitudinal vortices are present and the flow is primarily supersonic or higher speed.

With the next generation of computers, the complete three-dimensional Navier-Stokes equations can be applied to limited areas of the flow (such as shock—boundary-layer interaction regions). However, a complete three-dimensional viscous flow field about general configurations is probably beyond the capacity of even the next generation machine.

For all the computational procedures discussed thus far, the application to turbulent flows involves some modeling of the second-order turbulence correlation (or "Reynolds stress") terms appearing in the mean-flow equations. Presently, there is no fundamental theory of turbulence suitable to supply models for these correlations and, therefore, experimental data must be used and "universality" of these models beyond the experimental conditions from which they were obtained must be assumed.

There is hope that eventually turbulence can be computed by leaving in the time-dependent terms and using the instantaneous equations. This method is already being attempted for simple flows. (See ref. 19.)

This review, current status, and projection of the flow-field computation capability indicates that many of the uncertainties currently present in the design of hypersonic vehicles will be reduced in the near future as general flow-field programs become available. These codes should allow overall configuration optimization studies to proceed much more rapidly, accurately, and efficiently.

RESULTS OF RECENT RESEARCH

Engine-Airframe Integration

As discussed in references 1 and 2 and in the "Introduction" section of the present paper, significant reductions in structural weight and engine cooling requirements can accrue from efficient integration of the engine into the airframe at hypersonic speeds. As indicated in figure 3, the vehicle forebody acts as an inlet spike and the afterbody acts as an exhaust nozzle. Therefore, in order to obtain an efficient propulsive system, the underside of the basic vehicle must be designed as much from propulsion requirements as from the usual aerodynamic consideration of attaining high L/D.

The flow-field programs discussed in the previous section can be used to design the forebody section. The exhaust nozzle region, or vehicle afterbody, however, has several additional problem areas as indicated in figure 4. The combustor exhaust flow is a nonuniform, underexpanded, reacting (H_2-air system) gas. Major problems in the design of the vehicle afterbody (exhaust nozzle) region include calculating a three-dimensional flow with viscous and real-gas effects. Also, flow separation is possible at off-design (primarily lower Mach number) conditions.

Analytically, a three-dimensional method of characteristics program which includes real-gas effects will be used for the basic inviscid calculation. A numerical marching technique including the viscous terms will be developed to account for the effects of the mixing regions which surround the engine exhaust flow (where it mixes with the local external flow). Because of the real-gas effects upon the expansion process, ground simulation of the flow would be extremely difficult. Therefore, the approach of obtaining test data with partial simulation is taken and the data are used to calibrate the calculation procedures. These calibrated computational methods are then used to design the flight system.

Another benefit of engine-airframe integration is the possibility of vectoring the thrust to increase the lift coefficient. (See fig. 2.) At hypersonic speeds the gross thrust-weight ratio approaches 1. Also, the required inlet areas become very large and approach the body cross-sectional area. Therefore, as Mach number increases, the net thrust becomes the small difference between two large numbers, ram drag and gross thrust. Therefore, extreme care must be taken to place the thrust vector through the center of gravity; otherwise, a large trim penalty can occur which would negate the favorable effects of thrust vectoring. (See ref. 2.)

Active Cooling Techniques – Slot Injection

As pointed out in the "Introduction," recent studies (refs. 3 and 4) indicate that actively cooled hypersonic vehicles offer an attractive alternative to the "hot structures" concepts previously considered. (See ref. 1.) One possible active cooling technique involves the use of film cooling or tangential slot injection from rearward-facing slots. Early studies, using low-speed data for the cooling lengths involved, indicated that slot injection would not be competitive with internal cooling systems. However, recent data obtained at New York University (ref. 20) and Langley Research Center (ref. 21) indicate that slot injection is much more effective at hypersonic speeds, perhaps because of the thicker viscous sublayers associated with hypersonic turbulent flows.

A schematic of a slot injection flow is shown in figure 5. The injected flow is taken on board, cooled by the cryogenic fuel by using a heat exchanger, and then injected into the local boundary layer. This cold flow near the wall mixes with the hot external flow and therefore the heating increases from near zero at the slot exit to the usual large values downstream. When the heat transfer (or adiabatic wall temperature) becomes too large for the surface material used, another slot is necessary. In contrast to the STOL or high-lift applications where the boundary layer must be energized by slot injection, the slot velocity is less than the external value and the boundary-layer flow is deenergized. This slot injection scheme reduces the local skin friction as well as the heat transfer and is applied externally; thereby, the requirement of high wall thermal conductivity usually associated with internal cooling is relieved. Systems studies indicate a small (≈5 percent) net drag penalty associated with the cooling system. Some of the ram drag penalty associated with taking the air on board is offset by the skin-friction reduction and thrust from the slot flow. (See refs. 3 and 4 and paper no. 25 of this compilation.)

A calculation method has been developed to handle the slot-injection problem for the nearly optimum situation of matched static pressure between the slot exit and external flows. This method results in minimum total pressure for a stable slot flow at a slot Mach number near 1.0. This calculation method (ref. 22) is an extension of an available finite-difference solution technique to the complete two-dimensional compressible boundary-layer equations (ref. 14). A comparison between the calculation method (ref. 22) and the experimental data of reference 21 is shown in figure 6. In this situation the calculation method was developed first and then the data were obtained. The agreement between data and theory is excellent when the complex viscous flow which is relaxing downstream of injection is considered. The adiabatic wall temperature reduction which occurs immediately downstream of injection $\left(\text{as compared with the "undisturbed" value of } \frac{T_w}{T_t} \approx 0.9\right)$ is easily seen in figure 6 as is the relaxation back toward the undisturbed value which begins 10 to 15 slot heights downstream of the slot. Several mechanisms can be employed to increase the slot effectiveness and thus reduce the net drag penalty. Examples of these mechanisms or modifications include the use of multiple slots and inclusion of radiation cooling as well as the use of a thin lip above the slot (to reduce mixing). All these effects are included in the calculation method of reference 22 and results are indicated on the right of figure 6 for the effects of multiple slots on film-cooling effectiveness.

The flow situation for the multiple-slot case in figure 6 corresponds to a simulated flight case for a wing with edge Mach number 5 and initial boundary-layer thickness of 2.5 cm (1 inch). The x-distance shown is measured from an initial station where the first slot is placed. The mass flow is the same for each slot and injection occurs at matched static pressure with a slot average exit Mach number of 1.0.

Downstream of the first slot the wall temperature rises in a manner similar to that shown on the left of figure 6. At the 37.5 cm (15-inch) station downstream, the wall temperature reaches 700° K (1260° R). At this time, if a titanium surface were used, another slot must be inserted. As seen in the figure, the second slot is more effective since it operates in the cooled wake of the first slot, and thus more than twice the distance (83.8 cm or 33 inches) can be attained before a third slot is necessary. This same beneficial effect of multiple slots continues to occur, but at a diminishing rate, for each succeeding slot.

The calculation procedure of reference 22 has been extended to the three-dimensional case of an infinite swept slot and data are being obtained for verification. This computational method provides a fairly realistic tool to use in systems studies of slot-injection cooling for actual configurations at flight conditions and indicates the power of the numerical calculation techniques in providing accurate design information.

Lee-Side Vortex Problem

The basic problem associated with lee-side (or upper surface) vortices at hypersonic speeds is illustrated in figure 7. Oil-flow data are shown for the lee side of delta-wing and delta-half-cone models at $M_\infty = 6.8$ and $\alpha = 7^\circ$. These results are typical of observations for research models of this type (which simulate somewhat the forward section of hypersonic aircraft) and data obtained by using actual candidate configurations.

Near the center line on the lee surface, the oil flow indicates a region of high shear for the lower model in figure 7, a flat-topped delta wing. With volume added on the top surface (half-cone-delta model), similar high-shear regions are seen along the sides of the conical surface. Surface heat-transfer data indicate localized heating peaks along this high-shear region which are roughly a factor of 5 greater than the undisturbed lee-surface levels. These data show the fallacy of the common assumption that lower heating occurs on the lee surface than on the windward surface. Strong experimental evidence (flow-field visualization and surveys) indicates that these heating peaks are caused by the interaction between a set of longitudinal vortices and the vehicle surface. Such lee-side vortices are commonly observed at lower speeds (see ref. 23) but a fundamental difference exists between the hypersonic behavior of these vortices and their effects at lower speeds. For speeds up to supersonic, the lee-surface vortices affect the local pressure field to first order and are responsible for a significant increase in lift coefficient; therefore, their presence is primarily beneficial. (See paper no. 3 by Edward C. Polhamus.) At hypersonic speeds the lee-surface vortices do not seem to appreciably affect the local pressure field for practical values of sweep angle and their main effect is to cause large local increases in heating.

A possible alleviation technique for this problem is illustrated in figure 8. (See ref. 24.) Again surface-oil-flow results are shown for the lee side of delta-wing models with sharp leading edges. Here the models are similar to that employed in the lower part of figure 7, but with various types of leading-edge planform contouring. The nose contouring used for the model on the left is a conventional circular section. The oil flow for this model indicates that the vortices are still present and now form at the circular-arc afterbody juncture where a discontinuity in second derivative (or planform curvature) occurs. Again, high heating is measured in the high-shear areas.

The results for the models discussed thus far indicated that a configuration employing a planform with continuous curvature might not have a vortex heating problem. Such a model was constructed and tested; the results are shown on the right-hand side of figure 8. As expected, no vortices were observed over the lee surface of this model and the data showed no regions of increased heating. These results indicate the possibility that hypersonic vehicles could be designed to eliminate the adverse heating effects of longitudinal vortices generated at the nose and wing-root regions.

Boundary-Layer Transition

A fundamental design parameter needed for hypersonic vehicles is the location of boundary-layer transition. Regions upstream of the transition location are generally subjected to lower heating and skin friction compared with regions downstream of transition. Also, a local peak in heating occurs at the end of the nominal transition region (the beginning of turbulent flow) and the transitional flow region is subjected to large surface-pressure fluctuations. At lower speeds the laminar boundary layer is very sensitive to vehicle surface roughness and regions of adverse pressure gradient; therefore, transition usually occurs very near the nose or leading edge. Thus, no appreciable runs of laminar flow have been achieved for flight vehicles in the subsonic and supersonic flight regimes without extreme effort.

At hypersonic conditions several mitigating circumstances are observed. The laminar boundary layers seem to exhibit higher transition Reynolds numbers and much less sensitivity to roughness. Combined with these conditions is the increase in transition Reynolds number due to wall cooling (if smooth, actively cooled surfaces are employed). Therefore, some likelihood exists that a measurable fraction of the vehicle, perhaps 20 percent, could be subjected to laminar-transitional flow. Also, as indicated on the left of figure 9, a knowledge of the location of transition is important for the design of wing-root and propulsion-inlet regions, where the initial flow lengths are comparatively short. Accurate theoretical solutions for transition location, where the important nonlinear effects are included, are not yet available. Therefore, the transition information available is primarily obtained in ground facilities. At low speeds there was early recognition that free-stream disturbances in terms of vorticity or velocity fluctuations affected, to first order, transition data obtained in wind tunnels (as shown, for example, in ref. 25).

At hypersonic speeds the free-stream vorticity is extremely low and therefore the hypersonic transition data were at first thought to be relatively free of facility disturbance effects. However, during the past several years, considerable evidence (for example, ref. 26) indicated that another source of free-stream disturbance was present at hypersonic speeds, that is, radiated sound from the facility turbulent nozzle wall boundary layer.

Moving turbulent sources in the nozzle wall flow produce this sound or pressure fluctuation which impinges upon models tested in the free stream (shown schematically in fig. 9). Recent investigations at Langley Research Center (by Calvin Stainback) resulted in data, shown on the right-hand side of figure 9, which shows quantitatively that at hypersonic speeds transition results have been affected to first order by the facility free-stream noise levels. Therefore, the transition Reynolds number for flight (at small $p'/\bar{p}$) is probably considerably above the values obtained thus far in wind tunnels. This is borne out by the available flight data, which, although scarce, do indicate a transition Reynolds number of 20×10^6 at $M_e = 5$. This value is considerably above the 7×10^6 or less shown for $M_e = 5$ in figure 9 (obtained in a wind tunnel).

To conduct meaningful transition research for hypersonic cruise application, a facility with extremely low free-stream noise level is necessary. Design concepts and requirements for such a facility are under study at Langley Research Center. The basic problem is one of keeping the facility sidewall boundary layer laminar. Therefore, this work should yield not only far more realistic values for transition Reynolds number, but also contribute significantly to the knowledge concerning laminar flow control at hypersonic speeds.

COMPARATIVE AEROTHERMAL ENVIRONMENT OF HYPERSONIC TRANSPORT AND SPACE SHUTTLE

There is current interest in the possible use of space shuttle technology for a ground-to-ground continental and semiglobal transport, probably of the boost-glide type. In considering the design of a hypersonic cruise transport vehicle, the question naturally arises as to the applicability of shuttle technology. Figure 10 shows the comparative aerothermal environment between the shuttle and hypersonic cruise missions. Because of the blunter vehicles and much higher angles of attack and altitudes associated with the shuttle mission (see, for example, fig. 11), the local Mach numbers and Reynolds numbers for the hypersonic cruise case are much higher than those for the shuttle. Therefore, based on fluid-mechanic grounds, the local environment of the hypersonic cruise and shuttle missions differ markedly. However, much of the technology involved in materials, thermal protection, and other systems will be very useful in making design trade-offs for a hypersonic cruise vehicle.

Recent data concerning the aerodynamics of candidate hypersonic cruise configurations are given in references 27 to 29. These publications include effects of Reynolds number and Mach number on $(L/D)_{max}$ and show that the hypersonic cruise L/D is in the range from 4 to 6. Conventional values for the shuttle vehicles are of the order of 1.5 to 2.5.

CONCLUDING REMARKS

The paper reviews the status of hypersonic aircraft aerodynamic technology and pinpoints critical research areas for the 1970's. The outlook for numerical flow-field calculations and recent research in the problem areas of engine-airframe integration, slot injection as an active airframe cooling technique, lee-side vortex heating, and hypersonic boundary-layer transition are considered.

With realistic progress in aerodynamic technology, an efficient hypersonic cruise aircraft can be developed. The most critical area for research is the integration of the air-breathing propulsion system with the lower surface of the aircraft, since the forebody compression surface acts as an inlet spike and the lower part of the vehicle afterbody acts as an exhaust nozzle for the propulsion modules. A significant gain in payload can accrue from the further development and application of active cooling technology for both external and internal cooling concepts.

The development, within the next 3 years, of numerical solutions of the complete inviscid and viscous flow equations over real wing-body configurations should allow optimization of the aerodynamic configuration and should result in much less design uncertainty for thermal-protection systems. Because of the influence of real-gas effects upon the expansion of the engine exhaust over the aircraft afterbody, accurate numerical flow-field solution techniques are essential to the development of an efficient hypersonic cruise aircraft.

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TYPICAL CANDIDATE HYPERSONIC CRUISE CONFIGURATION

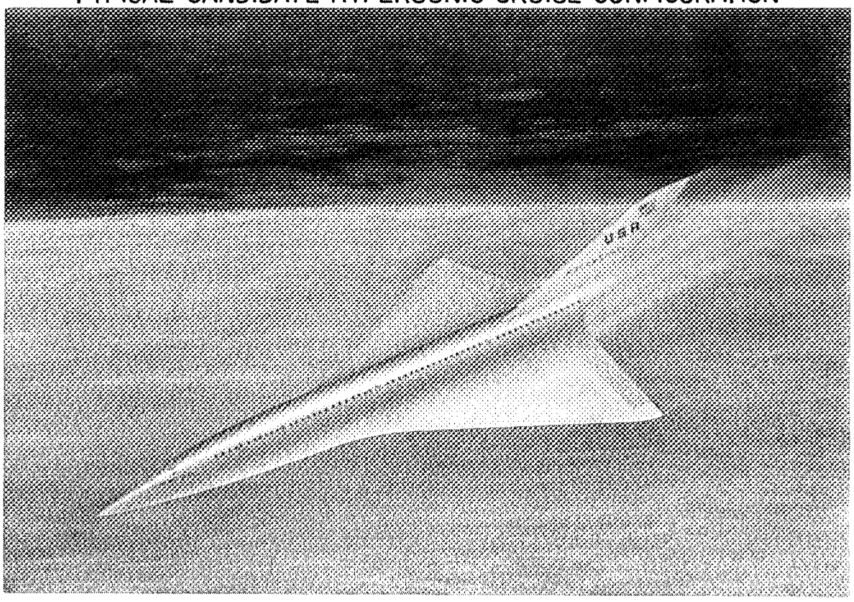


Figure 1

SCRAMJET-VEHICLE INTEGRATION

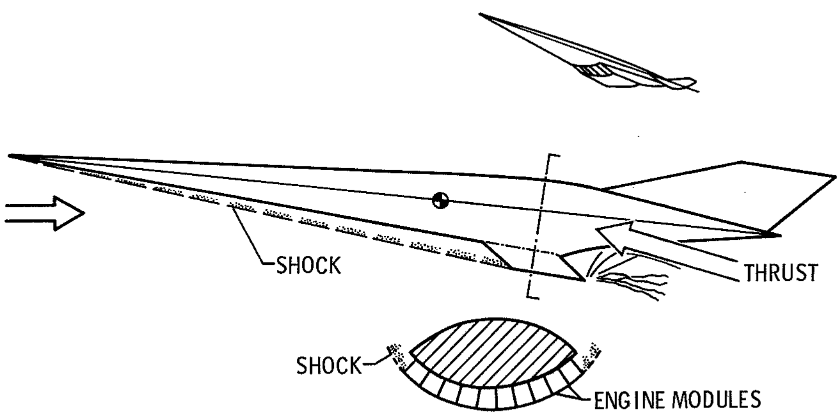


Figure 2

AIRFRAME CONTRIBUTION TO ENGINE PERFORMANCE

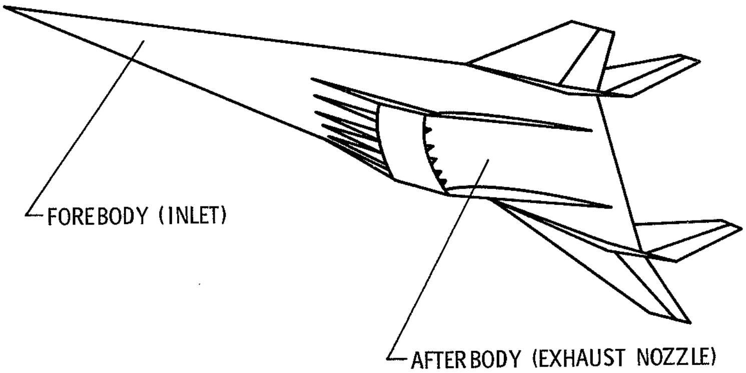
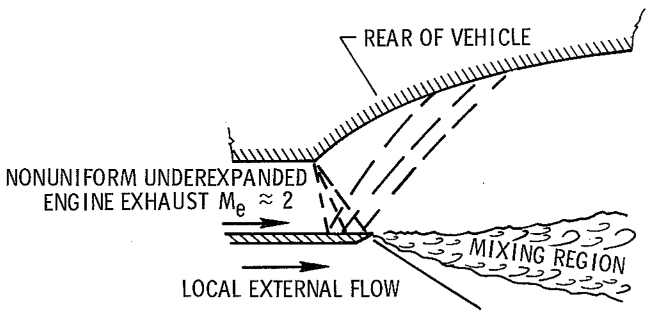


Figure 3

AERODYNAMICS OF HYPERSONIC ENGINE—AIRFRAME INTEGRATION

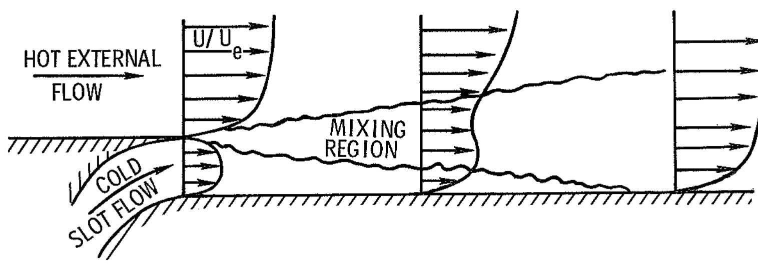


PROBLEMS

- BASIC THREE-DIMENSIONAL EXHAUST REGION DESIGN/OPTIMIZATION
- OFF-DESIGN OPERATION
- VISCOUS MIXING/REAL-GAS EFFECTS
- INCREASED HEATING
- AERODYNAMIC STABILITY AND CONTROL $\left(\frac{\text{GROSS THRUST}}{\text{GROSS WEIGHT}} \approx 1\right)$

Figure 4

TANGENTIAL SLOT INJECTION
PROSPECTIVE ACTIVE COOLING TECHNIQUE



ADVANTAGES OF TANGENTIAL SLOT INJECTION

- HEAT-TRANSFER REDUCTION
- SKIN FRICTION REDUCED
- COOLING METHOD APPLIED EXTERNALLY
- SMALL NET DRAG PENALTY

Figure 5

SLOT COOLING - ADIABATIC WALL DATA AND THEORY

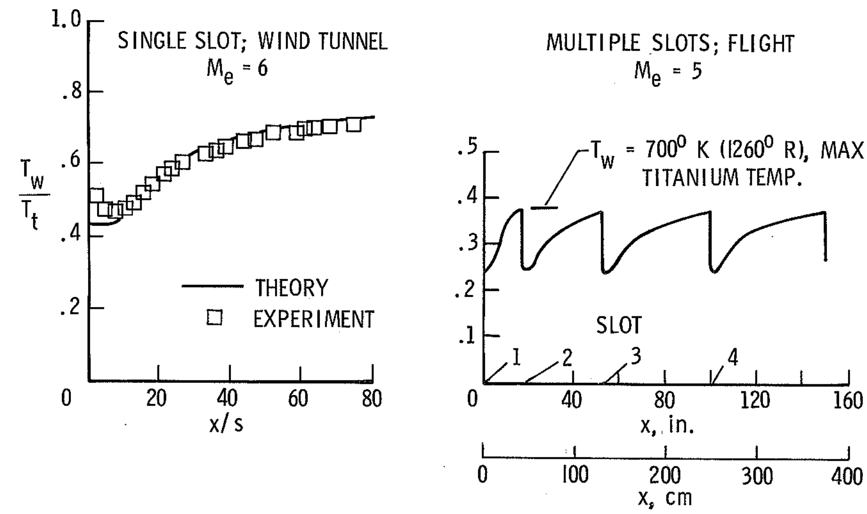


Figure 6

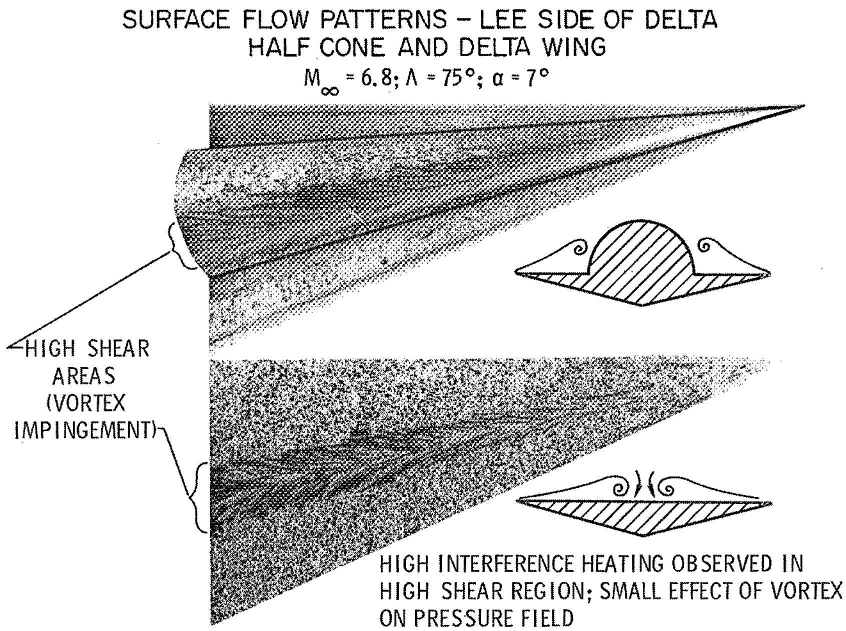


Figure 7

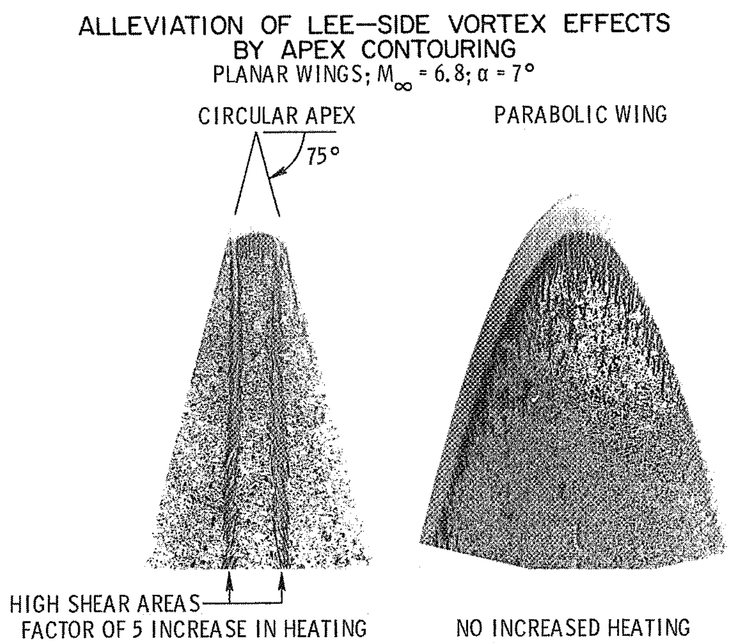


Figure 8

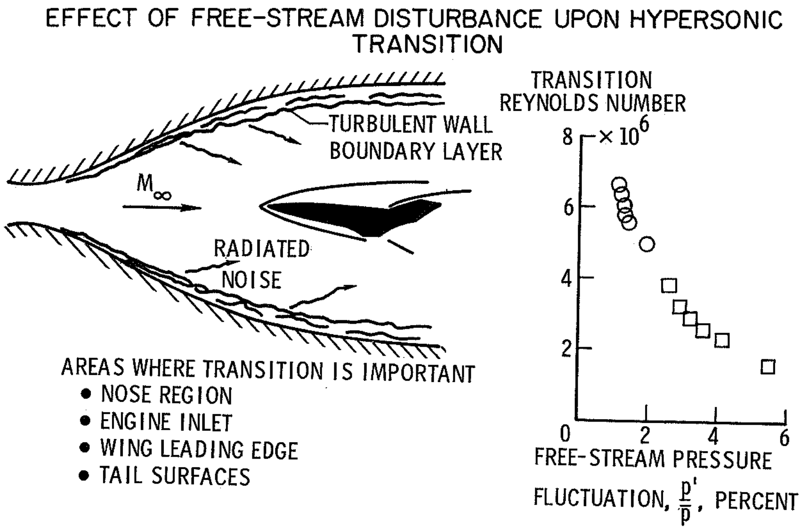


Figure 9

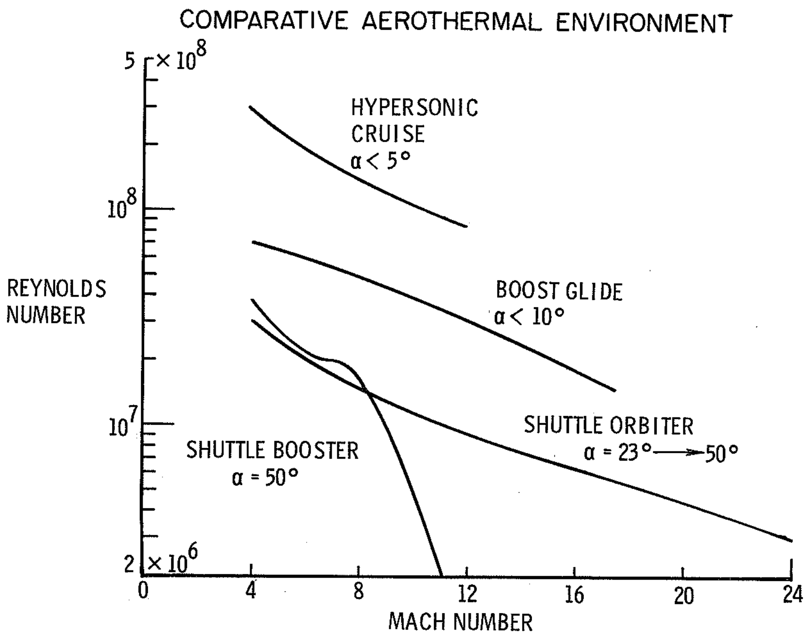


Figure 10

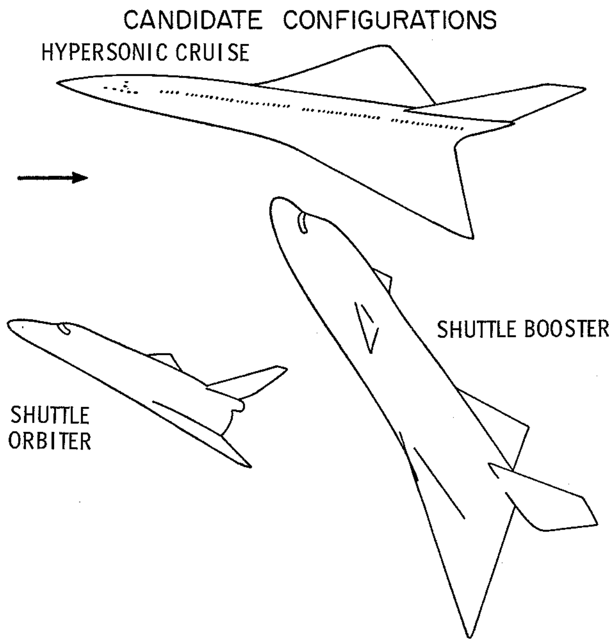

CANDIDATE CONFIGURATIONS
HYPERSONIC CRUISE
SHUTTLE BOOSTER
SHUTTLE
ORBITER

Figure 11

MATERIALS FOR JET ENGINES

By John C. Freche
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SUMMARY

The key to improved turbojet engine performance lies in the development of suitable materials and the engineering techniques required to use these materials effectively. Discussed herein are the improvements that can be expected during the ensuing decade in the materials used for key engine components, such as the fan blades and early-stage compressor blades, the latter-stage compressor blades, the turbine disks, and the turbine stator vanes and blades. Some of the materials problems posed by the engine environment, particularly corrosion and thermal fatigue, are also discussed as are some of the more advanced protection systems designed to solve these problems.

INTRODUCTION

Substantial advances in jet engine performance can confidently be expected in the ensuing decade. However, the key to the realization of these advances lies in the development of suitable materials as well as the engineering techniques which will be required in order to use such materials in an effective and efficient manner. Since a jet engine is a heat engine, the higher the working temperature, the higher the efficiency and specific output. Thus, if materials with higher use-temperature cabability can be provided, better engine performance will result. Also, if materials of greater strength and/or lower density can be substituted for currently used materials, significant savings in engine weight can be realized. These weight savings would in turn be reflected in improved engine performance.

This paper indicates what can reasonably be expected in the way of extending the temperature and strength capabilities and reducing the density of materials used for key engine components (fig. 1). It deals with low-temperature components (the fan blades and early-stage compressor blades), intermediate-temperature components (the latter-stage compressor blades and turbine disks), and high-temperature components (the turbine stator vanes and turbine blades). Also, some of the major materials problems posed by the engine environment, in particular the problems of corrosion and thermal fatigue associated with the high-temperature components, and the progress being made in solving these problems are discussed.

FAN AND EARLY-STAGE COMPRESSOR BLADES

The materials currently being used for fan blades and early-stage compressor blades are principally titanium alloys. Their maximum temperature capability is currently about 427° to 482° C (800° to 900° F). A new class of materials affords potential substitutes – namely, composites (particularly polymer matrix composites, although certain types of metal matrix composites should not be excluded). The processing steps involved in making composite blades are illustrated schematically in figure 2. In the case of both polymer-matrix-composite and metal-matrix-composite blades, the first step involves the production of monolayer composite sheets consisting of fibers and matrix material. There are a number of ways by which such monolayers can be made. One way is shown in this figure. Once made, the monolayers are cut to the desired dimensions, stacked, and fabricated by a pressurization process to the final blade shape. The blade fabrication process is similar for both polymer and metal matrix materials except that the temperature, pressure, and atmosphere requirements vary depending on whether the product is a polymer or metal matrix material.

To review the steps in the process very briefly, the polymer-composite monolayer can be made by unwinding the reinforcing fiber from a spool, coating it with the polymer by pulling it through a polymer bath, and winding it onto a rotating drum. Partial curing is achieved by heating in an oven. Monolayer sheets are removed from the drum by cutting.

The metal matrix monolayer can be produced by winding a single layer of fiber on a drum and spraying with an organic binder to hold the fiber array in position. A section is cut and placed between metal matrix foil sheets and is diffusion bonded into a monolayer by application of heat and pressure.

The various monolayers that make up the blade are cut from the monolayer sheets and stacked with the fiber orientation of successive monolayers arranged so as to provide the proper strength characteristics, as dictated by the blade design.

What are the advantages that can be expected from composite blades? Some are apparent from figure 3, which shows two fan blades for the Allison TF-41 engine (ref. 1). The one on the right is a conventional forged Ti-6Al-4V alloy blade. The one on the left is a composite containing 50-volume-percent SiC coated boron in a Ti-6Al-4V alloy matrix. This particular composite blade was made at TRW Inc. under U.S. Air Force sponsorship. The major differences apparent to the eye are the absence of the midspan stiffener and a reduction in the blade section thickness. Because the composite has a much higher modulus than the bulk titanium, its resistance to vibration induced by engine pulses is greater. This feature makes possible the removal of the midspan stiffener. The greater strength of the composite permits a reduction in section thickness. Substitution of the titanium-alloy—boron composite for the conventional forged titanium-alloy blade results in an airfoil-weight savings of 30 percent. Also, additional engine-weight savings should result from the reduced amount of armor needed for the fan casing. In this particular example, a 3-percent increase in airflow results from removal of the midspan stiffener. This increase, of course, can improve engine performance. Programs are currently underway at the National Aeronautics and Space Administration to investigate the use of polymer matrix composites for fan blades in V/STOL applications.

Figure 4 (obtained from ref. 2) indicates the advantages of fibers of various materials over bulk metals. Tensile strength divided by density is plotted against modulus divided by density for fibers of various materials and for bulk metals. The data are normalized by dividing by density to permit a valid comparison of many different materials having widely differing densities. Some fibers such as S and E glass are stronger than the bulk metals but not more rigid. Others such as boron and carbon are substantially superior in both properties by factors of about 5 and 7, respectively. Although not shown in this figure, the very strongest and stiffest materials are whiskers of silicon carbide and carbon (0.5 μm (0.00002 in.) in diameter and 500 μm (0.02 in.) long). On a normalized basis, these materials have strength values about four times as large as the values for the strongest carbon fiber shown and twice its modulus. Because of handling difficulties, however, the use of whiskers as reinforcements in composites is considerably more remote than the use of fibers. In any case, by placing such reinforcements as fibers in appropriate resin or metal matrix materials, it is possible in a composite to take advantage of a large part of the property improvements fibers exhibit over bulk metals.

Figure 5 shows current and anticipated use-temperature limits for various matrix composites (refs. 3 to 7). The polymer matrix composites have already demonstrated use-temperature capabilities ranging from approximately 93° C (200° F) for the epoxies to about 260° C (500° F) for the polyimides and pyrrones. In fact, recent work at the NASA Lewis Research Center with a graphite-fiber-reinforced polyimide shows that the initial strength of the composite can be retained at least for 1000 hours at 260° C (500° F). The expected growth in use-temperature potential is indicated by the cross-hatched regions. A reasonable anticipated upper limit for the polymer matrix composites is about 316° C (600° F) and will probably be achieved with the pyrrones and polyimides. The metals which are sufficiently light and have the desired combination of properties needed for early-stage compressor blades are aluminum and titanium. Composites which employ these metals as matrix materials have demonstrated use temperatures of 316° to 482° C (600° to 900° F) in short-time tensile tests. It is reasonable to expect that the potential use temperatures of aluminum and titanium matrix composites may eventually be raised to as high as 482° and 649° C (900° and 1200° F), respectively. If indeed such use temperatures are achieved with these metal matrix composites, their usefulness could be extended to engine components other than the fan and early-stage compressor blades. Thus, they could become candidates for the latter-stage compressor blades.

Finally, what are the major problems that still must be solved before composites can be used in engine service in early-stage compressor blading? Research has identified several major problem areas (ref. 3). One major area of concern is the ingestion of sand, dirt, and rain which can occur in varying degrees depending on the manner in which the aircraft is used. The ingestion of material of this type causes an erosion of the blade to occur with time and leads to eventual loss of substantial portions of the blade. The other major problem area is the ingestion of larger objects such as stones, tire tread, ice, birds, bolts, rivets, and mechanics' tools and rags. Objects of this type can cause immediate loss of portions of the airfoil or even failure at the blade root with loss of the complete airfoil.

The more conventional titanium blades presently in use in engines are more resistant to erosion and impact than are the fiber-resin composite blades but are still vulnerable to the degree that erosion has been a very severe problem in helicopter engines. In addition, the severity of the impact problem with titanium blades is illustrated by the fact that the Federal Aviation Administration requires that the engine be capable of being shut down without danger to the aircraft if objects such as mechanics' rags, tire tread, and large birds are ingested. In other words, impact is a severe problem regardless of which material is used for the blades, but it is most severe in the case of the fiber-resin composites. For some metal matrix composites, blade leading-edge protection against foreign-object damage would also be required. This protection would be necessary because of the relative weakness of the matrix material as in the case of aluminum.

Research is underway to provide answers to these problems for composite blades. Leading-edge protection devices have been developed that appear to have solved the erosion problem to a large degree. These devices consist of special metallic shields and screens that are embedded in the composite in the leading-edge area where most of the erosion takes place. In addition, these devices have substantially improved the resistance of composite blades to impact by larger objects. For example, ice no longer appears to be a problem in the limited tests run to date. However, strikes by large birds in the 0.9- to 1.8-kg (2- to 4-lb) category still pose a severe problem.

In summary, considerable testing needs to be done with composite blades to establish the practicality and long-term effectiveness of the protection devices from an erosion point of view, and considerable development work needs to be done to bring the level of protection against damage from the larger objects up to a par with that provided by bulk titanium-alloy blading.

LATTER-STAGE COMPRESSOR BLADES

The latter-stage compressor blades in advanced engines will be expected to operate at temperatures from 482° to 649° C (900° to 1200° F). Current usage in advanced engines at these temperatures is limited to nickel-base alloys. If higher strength titanium alloys were developed and the problems associated with their use at temperatures above 482° C (900° F) solved, substantial weight savings would be realized since the density of titanium alloys is only 50 to 60 percent that of nickel-base alloys.

Titanium technology has for the most part been developed since 1950 and this material still offers considerable promise for further advances in properties. Figure 6 illustrates the progress that has been made in the development of titanium alloys since then and projects the advances that may be expected. Much of the information presented in figure 6 was obtained from Harold D. Kessler of Reactive Metals, Inc., Niles, Ohio. Alloy use temperature is plotted against the year various titanium alloys were introduced to service. If a linear projection is assumed, alloys with use-temperature capability of the order of 593° to 649° C (1100° to 1200° F) at the stresses required for compressor-blade application should be available by the mid-to-late seventies. Also shown in this figure are the evolutionary steps, some already known and others projected, in the development of titanium alloys with the desired capability. Titanium undergoes an allotropic transformation from a body-centered cubic (β phase) structure to the hexagonal close-packed (α phase) crystal structure at about 900° C (1650° F). A gain in use-temperature capability from 371° to 482° C (700° to 900° F) has been achieved by continued development of α plus β phase alloys. These alloys consist primarily of α and up to 10-percent β phase, and the addition of solid-solution-strengthening and α-phase-stabilizing elements such as Al, Si, and Zr resulted in the gains that have been made. An additional gain of about 55° C (100° F) in use temperature has been realized by the addition of 0.2 to 0.5 percent Si to α plus β phase alloys. This gain resulted in strength increases due to the formation of microscopic, complex silicide particles which precipitate and are dispersed in the α plus β matrix. The strongest Ti alloys of this type are 5521S and 5621S (ref. 8). Above 482° C (900° F), protective coatings are required to overcome problems of hot-salt stress corrosion (ref. 9) and oxygen contamination (ref. 10) with titanium alloys. Research programs to provide suitable protection against these environmental hazards are underway at NASA and elsewhere.

In order to achieve the 649° C (1200° F) use-temperature design goal, a number of alloy development approaches are being considered. These approaches include powder metallurgy techniques. They are intended to provide more uniform dispersion of strengthening elements such as silicon, carbon, and boron than can be obtained by conventional melting. Still another approach is to utilize all β phase alloys (ref. 11). Preliminary results of research at the Lewis Research Center indicate that significant strength advantages can be obtained with β phase alloys over the most advanced α plus β phase alloys. Of course, much alloy development work still needs to be done before a titanium alloy can actually be applied at 649° C (1200° F) in an engine.

DISKS

Substantial improvements in materials used for the latter-stage compressor disks and the turbine disks can also be expected. These components will be required to operate under high stress at temperatures of 649° to 760° C (1200° to 1400° F). The materials currently being used for disks are conventionally wrought nickel-base alloys. Their strength is inherently limited by the amount of alloying content that they can contain without encountering massive segregation in the cast ingot. The amount of alloying content also largely determines their resistance to deformation and whether they can be worked by conventional forging procedures. Recent innovations in processing show that substantial strength advantages can be achieved in the intermediate temperature range with nickel-base prealloyed powder products (ref. 12). These products avoid the segregation problem, and they can be superplastically shaped by isothermal forging regardless of the amount of alloy content.

Figure 7 illustrates segregation in a highly alloyed nickel-base alloy casting. An X-ray of the tensile bar containing a nominal alloy content shows the structure to be homogeneous. The X-ray of the highly alloyed bar shows light and dark areas indicating a nonhomogeneous structure. This nonhomogeneous structure is the result of segregation of some of the alloying constituents which occurred as the melt solidified.

Prealloyed powder processing (ref. 13) is illustrated schematically in figure 8. An alloy of any given composition is vacuum melted. When the melt is poured, inert gas jets are directed at the molten metal and cause it to atomize. The atomized metal droplets solidify rapidly into powders that are collected and sized. The powders are then compacted by extrusion or hot pressing, and the resultant compact can be subjected to any desired heat treatment. The rapid solidification of the atomized metal droplets results in an extremely fine grain size and a homogeneous structure without segregation.

Figure 9 illustrates the results of some recent work with the prealloyed powder technique at the Lewis Research Center (ref. 14). It shows the substantial strength gains possible when the prealloyed powder process is applied to the highly alloyed, cast material NASA-TRW VI-A. This material could not be forged into a disk by conventional forging practice. The tensile properties of the alloy are compared in the prealloyed powder product form and cast form as a function of temperature. The prealloyed powder product has an ultimate tensile strength of 1890 MN/m^2 (274.5 ksi), about twice that of the cast version. Such high tensile strength in a disk alloy of course means that the disk burst strength will be high. At 649° C (1200° F), the strength value is 1630 MN/m^2 as compared with 1140 MN/m^2 for the cast version (236 ksi as compared with 165 ksi). Above 816° C (1500° F), the curves cross and the prealloyed powder product is weaker. This weakness is also reflected in abnormally high elongations of 300 percent at 1093° C (2000° F), which are indicative of superplasticity resulting from the fine grain size. In nickel alloys, grain boundaries are stronger than the matrix at low temperatures and weaker than the matrix at high temperatures. One can take advantage of the superplastic nature of the powder product at high temperatures to deform the material isothermally into any desired shape, such as a disk. Thus the powder product can be used at the lower temperatures of interest for disk applications.

High longtime strength properties are also a requirement of a disk alloy. At 649° C (1200° F) and 1035 MN/m^2 (150 ksi), the powder product demonstrated a rupture life six times as long as that of the strongest conventional wrought alloy in use today (618 hours as compared with 100 hours) and more than an order of magnitude longer than that of cast VI-A, which has a 14-hour life (ref. 14). By utilizing an appropriate heat treatment, life was further increased to 2013 hours. These results represent a significant breakthrough in showing the potential of using highly alloyed, normally unworkable superalloys for achieving superior properties at temperatures encountered in turbine disks. They also indicate a major new direction that superalloy development will take for meeting advanced-engine disk design requirements in the ensuing decade.

TURBINE STATOR VANES AND BLADES

Currently, turbine stator vanes and turbine blades are made from cast nickel-base and cobalt-base alloys, particularly nickel-base alloys. These are the workhorse materials of the "hot" engine components and will remain so for some time to come. Their current use-temperature limitations are being overcome to some extent by various cooling schemes. Inevitably however, the need for increased engine performance will require higher cyclic operating temperatures, and substantial advances will have to be made in this area as well. Two of the most promising approaches for extending the use-temperature capability of these components are dispersion-strengthened materials and metal matrix composites.

In figure 10 the cross-hatched region illustrates the strength capability of cast nickel-base and cobalt-base alloys as a function of temperature (ref. 15). Superimposed on this plot is the curve for WAZ-20, an advanced-temperature cast nickel-base alloy resulting from recent work done at the Lewis Research Center (ref. 16). Its strength advantage over the strongest currently available cast nickel-base and cobalt-base alloys lies in the temperature range from approximately 1150° C (2100° F) to slightly above 1204° C (2200° F). Results for WAZ-20 are shown simply to indicate that some gains in use temperature are still possible by conventional alloy development techniques for nickel-base alloys. However, the strength of cast alloys clearly drops off sharply at high temperatures, and in the general vicinity of 1204° C (2200° F) they have very little strength. Also superimposed on this figure is the curve for a promising dispersion-strengthened material, TD NiCr (ref. 17 and unpublished data obtained at the Lewis Research Center). In materials such as this one, high-temperature strength is achieved by distributing nonreactive, fine particles such as stable refractory oxides (in this case thoria) throughout a base metal matrix. This procedure results in a relatively flat strength curve, in which useful strength is maintained virtually to the melting point of the base metal. Dispersion-strengthened materials such as TD NiCr are being subjected to intense investigation for use as stator vanes and combustion-chamber liners, which are relatively low stressed components but which require the ability to withstand the highest gas temperatures in the engine operating cycle. Because of the relatively low strength of dispersion-strengthened materials at temperatures below 1093° C (2000° F), they are not as yet suitable for application to the more highly stressed hot engine components such as the turbine blades. One objective of research with dispersion-strengthened materials is to apply successfully the dispersion-strengthening process to alloys which already have good intermediate-temperature strength. In this way, it may be possible to meet the high-stress design criteria for turbine blades as well as the ultrahigh-temperature requirements of stator vanes with dispersion-strengthened products.

One of the most promising approaches for increasing the high-temperature strength capability of materials for potential turbine-blade applications is by means of metal matrix composites (ref. 2). It has been pointed out previously (in connection with fan blades and compressor blades) how composites permit the superior properties of materials in fiber form to be utilized. For high-temperature, highly stressed components such as the turbine blades, refractory-metal-fiber—superalloy composites hold great promise. The superalloys have adequate ductility but lack strength above 1093° C (2000° F). Refractory metals such as tungsten and molybdenum, on the other hand, have the strength required for turbine blades at temperatures well above 1093° C (2000° F), but they are generally brittle and oxidize catastrophically at these temperatures. Refractory-metal-fiber—superalloy composites afford the potential of combining the best properties of both components. Figure 11 provides a comparison of the strength-density ratio for a 1000-hour life as a function of use temperature for two nickel-alloy matrix composites recently developed at the Lewis Research Center (refs. 2 and 18) and one of the strongest conventional cast superalloys. The horizontal band on the figure represents a range of strength-density values required for turbine blades for an advanced turbojet engine. The composite reinforced with tungsten – 2-volume-percent-thoria

($W-2ThO_2$) fibers shows an increase of 110° C (200° F) above the use temperature of the strongest cast superalloy. The composite reinforced with tungsten-hafnium-carbon (W-Hf-C) fibers shows an increase in use temperature of 220° C (400° F).

The use of such composites for turbine blades poses a number of problems. One problem is fiber-matrix interaction during fabrication. In the actual engine application, resistance to foreign-object damage and of course oxidation resistance must be provided. Adequate fabrication processes to control reaction during fabrication of composite test specimens have already been developed. Impact resistance comparable to that obtained with cast superalloys has been achieved by heat treating and the proper selection of volume-percent fiber content. Work must now be directed toward the development of metal-matrix blade hardware for engine test and evaluation.

Mention should also be made of nonmetallic refractory compounds as a class of materials with potential for stator-vane and turbine-blade applications for ultrahigh temperature use, on the order of 1371° C (2500° F) and above. Their high melting temperatures, low densities, and resistance to oxidation, corrosion, and abrasion make them of interest. Silicon nitride and silicon carbide appear to hold considerable promise. However, the brittleness of refractory compounds renders them subject to catastrophic failure from stress concentrations or mechanical impact. There is hope that these liabilities may be overcome or circumvented by the application of composite-materials technology and proper component design. Continued work with these materials from basic deformation mechanistic studies to hardware testing is desirable to fully establish their usefulness.

This section of the paper has shown how substantial increases in strength and use temperature can be achieved for turbine-stator-vane and turbine-blade applications for advanced engines. There are, however, two major problems imposed by the engine environment. Their severity increases with increasing temperature and they must be overcome in order to apply superalloys successfully to turbine-engine stator vanes and blades. These problems are corrosion and thermal fatigue.

CORROSION RESISTANCE

Figure 12 indicates the interrelation of the corrosion and strength limitations in nickel-base materials (ref. 19). Use temperature for a 1000-hour stress-rupture life at a stress of 138 MN/m^2 (20 ksi), a stress typical of that encountered in turbine blades, is plotted against the year of introduction of a number of nickel-base alloys. The corrosion limit is plotted as a band, since the corrosion capabilities of any one alloy are dependent on test conditions. Up to 1960, the strength capability of these alloys precluded a corrosion problem in engines by limiting their upper use temperature. In the years that followed, higher use temperatures were achieved by compositional changes, but at the expense of corrosion resistance. As a result, the use temperature of the bare alloy in the case of some of the strongest, most recent alloys such as B-1900 and VI-A is set by corrosion limitations rather than strength limitations. A similar evolutionary process occurred in the development of cobalt-base alloys. Since nickel or cobalt will be the base metals for dispersion-strengthened alloys as well as the matrix materials for composites, this corrosion barrier is equally applicable to these advanced materials. The problem of high-temperature corrosion of superalloys in gas turbine engines is further complicated by cyclic engine operation, high-velocity gas flows, and the presence of abrasive particles in the gas.

Figure 13 illustrates the effectiveness of several of the most advanced protection systems developed to date (ref. 20 and unpublished data obtained at the Lewis Research Center). In this figure, the time to measurable weight loss of a representative nickel-base alloy, IN-100, is shown after exposure to Mach 1 jet-fuel combustion gas in a burner facility. To simulate engine conditions, the test samples were subjected to cyclic operation. The maximum metal temperature in each cycle was 1093° C (2000° F). Bare IN-100 began to lose weight almost immediately, an indication that it is inadequate even for short-time service under these rather severe cyclic operating conditions. Application of one of the best metal claddings developed to date, a FeCrAlY foil, substantially improved performance but still gave only a limited life, on the order of 200 hours, before material degradation occurred. The application of complex metal aluminide coatings, in which the aluminum and metal substrate react and interdiffuse, is another method of providing corrosion protection. One of the best of these coatings is seen to extend life to about 300 hours before measurable weight loss occurred. A combination of a NiCrAlSi cladding and an aluminide coating provided a protection system which extended life to at least 800 hours in this rather severe test. These results show that protection systems are being developed which hold promise of overcoming the corrosion barrier associated with the high gas temperatures anticipated with advanced engines.

THERMAL-FATIGUE RESISTANCE

Of equal concern as a barrier to high-temperature operation is the problem of thermal fatigue. This problem is manifested by the initiation and propagation of cracks in a material caused primarily by alternate heating and cooling, a situation encountered many times by turbine blades and stator vanes during normal engine operation (refs. 21 and 22). Such heating and cooling gives rise to a nonuniform temperature distribution which in turn causes thermal stresses. Figure 14 shows what thermal fatigue can do to a gas turbine blade. After 55 hours of cyclic operation in a stationary research engine, small cracks appeared perpendicular to the sharp leading edge. These cracks grew with time and finally, after 306 hours, they caused the loss of the upper portion of the blade.

Figure 15 shows one of the approaches being used to overcome this problem. Shown (courtesy of Pratt & Whitney Aircraft) are two turbine blades of the same nickel-base alloy, each cast differently. The one on the left (PWA 659) was cast in a conventional manner and exhibits a random polycrystalline grain structure. Thermal-fatigue cracks such as are shown in figure 14 typically initiate at the intersection of grain boundaries with the leading edge and follow an intergranular path. By using an advanced casting technique known as directional solidification, grains may be oriented parallel to the blade span as shown in the blade in the right-hand illustration (PWA 664). In this way, potential crack sites are eliminated and the time to cracking is significantly increased. Another approach, of course, is to provide coatings which cover potential crack sites.

The dramatic improvement that can be obtained in thermal-fatigue resistance by these techniques is illustrated in figure 16, which shows the comparative thermal-fatigue resistance of a typical nickel-base alloy, IN-100 (ref. 23). The specimens were subjected to cyclic tests in a fluidized bed. In the uncoated, conventionally cast condition, the first crack was observed after 30 cycles. In the coated, conventionally cast condition, life to the first observed crack was extended to over 500 cycles. In the directionally solidified condition, life was extended by another order of magnitude to over 5000 cycles.

Of course, the techniques described represent the tangible steps being taken to overcome the problem of thermal fatigue. Hand in hand with these techniques must go the ability to predict, in advance of service, the life of components subjected to complex thermal-fatigue conditions imposed by the engine environment. Extensive efforts are being directed toward that end at the Lewis Research Center, and methods which take into account the interrelation of low-cycle fatigue and creep damage in predicting component life are being developed (refs. 24 and 25).

CONCLUDING REMARKS

In conclusion, substantial improvements in the properties of materials used for key engine components can be anticipated during the ensuing decade. These improvements are summarized in table I. Very briefly, the major advances in materials research for the low-temperature engine components will come in the area of polymer matrix composites. For the intermediate-temperature components, prealloyed nickel powder products and possibly advanced titanium alloys hold the greatest promise. For the high-temperature components, dispersion-strengthened superalloys and refractory-metal-fiber—superalloy composites will provide major advances. The problems of corrosion and thermal fatigue will be contained by various protection systems and directional solidification.

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TABLE I.- PROJECTED IMPROVEMENTS IN ENGINE MATERIALS

COMPONENT	APPROACH	ADVANCES
FAN & EARLY-STAGE COMPRESSOR BLADES	POLYMER MATRIX COMPOSITES	REDUCE COMPONENT WEIGHT AT TEMPERATURES TO 316° C (600° F)
LATTER-STAGE COMPRESSOR BLADES	TITANIUM ALLOYS	REDUCE COMPONENT WEIGHT BY HALF BY SUBSTITUTING FOR NICKEL ALLOYS TO TEMPERATURES OF 649° C (1200° F)
TURBINE & COMPRESSOR DISKS	NICKEL-BASE ALLOYS BY PREALLOYED POWDER PROCESSING	DOUBLE STRENGTH IN 649° TO 704° C (1200° TO 1300° F) RANGE OVER CAST ALLOYS PERMITS REDUCTION IN DISK SECTION THICKNESS AND ENGINE WEIGHT
TURBINE STATOR VANES	DISPERSION-STRENGTHENED SUPERALLOYS	RAISE USE TEMPERATURE TO 1314° C (2400° F) TO LOWER COOLING REQUIREMENTS AND INCREASE ENGINE PERFORMANCE
TURBINE BLADES	SUPERALLOY MATRIX COMPOSITES	RAISE USE TEMPERATURE TO 1204° C (2200° F) TO LOWER COOLING REQUIREMENTS AND INCREASE ENGINE PERFORMANCE
CORROSION RESISTANCE	SUPERALLOYS	ADVANCED COATINGS & CLADDINGS TO IMPROVE COMPONENT LIFE AND RELIABILITY
THERMAL-FATIGUE RESISTANCE	SUPERALLOYS	DIRECTIONAL SOLIDIFICATION & COATINGS TO IMPROVE COMPONENT LIFE AND RELIABILITY

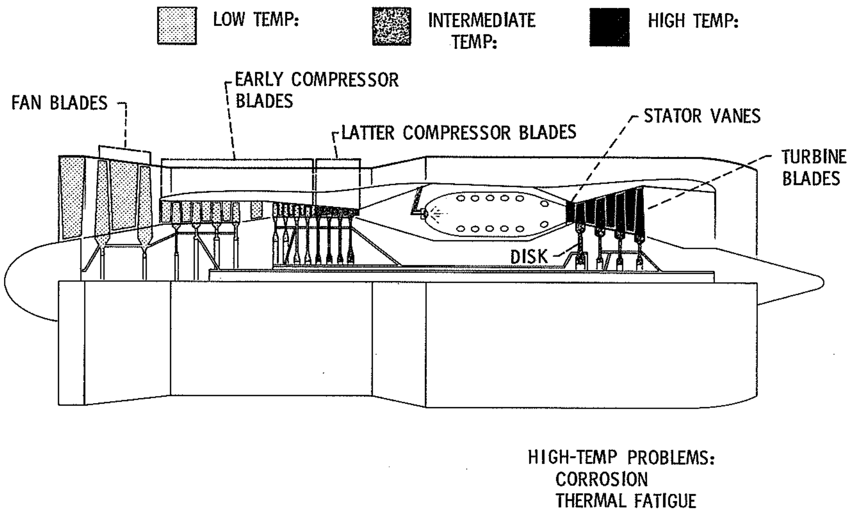


Figure 1.- Schematic diagram of jet engine.

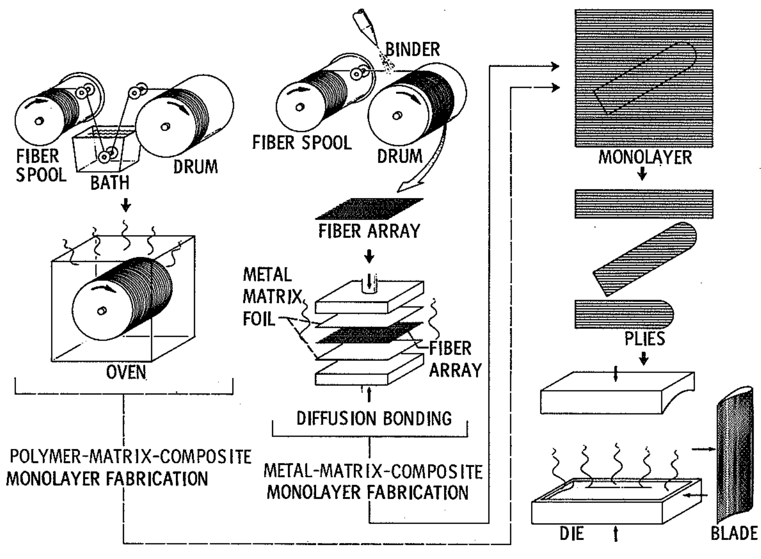


Figure 2.- Composite-blade fabrication.

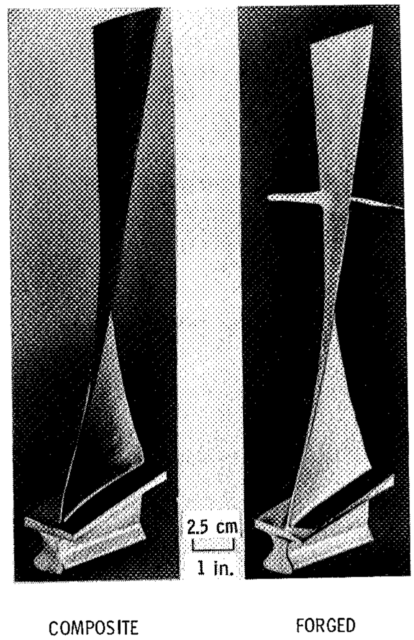


Figure 3.- Composite and forged titanium alloy blades.

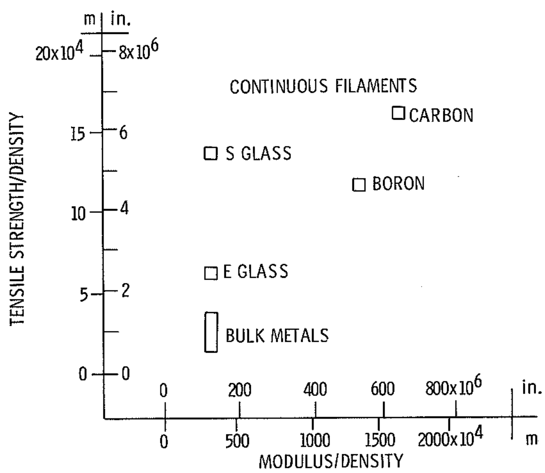


Figure 4.- Opportunities for improvement of strength and rigidity with fibers.

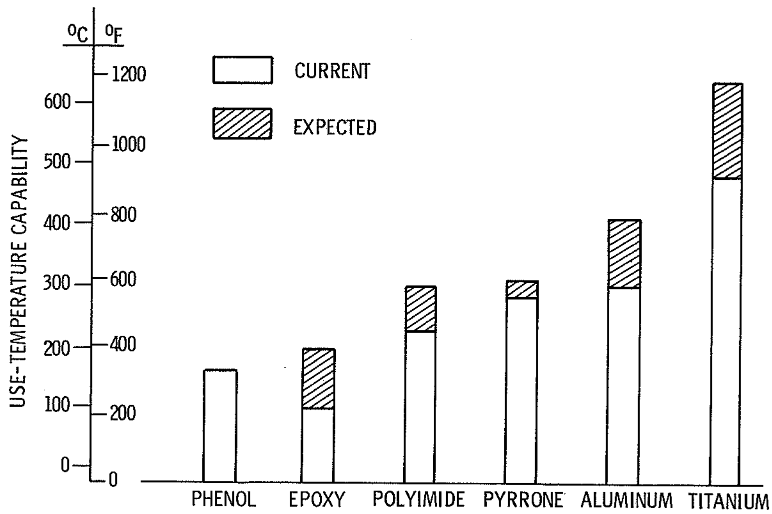


Figure 5.- Composite use-temperature limits.

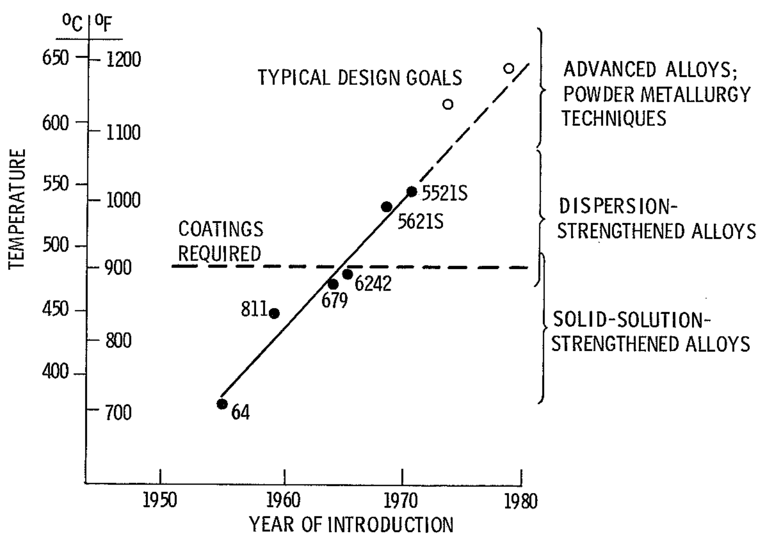


Figure 6.- Progress in development of titanium alloys (0.1-percent creep in 1000 hours at 40 ksi).

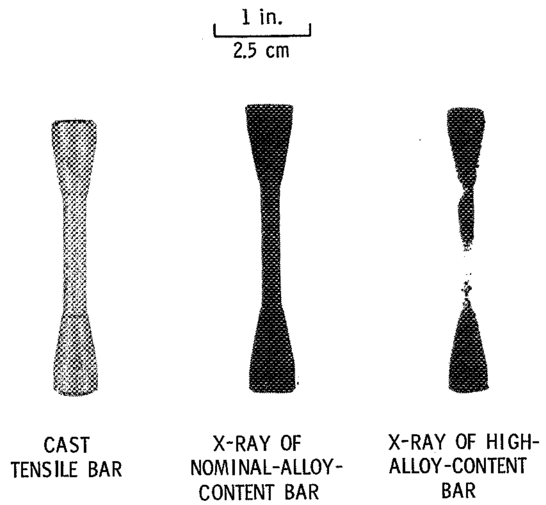


Figure 7.- Segregation in highly alloyed nickel-base casting.

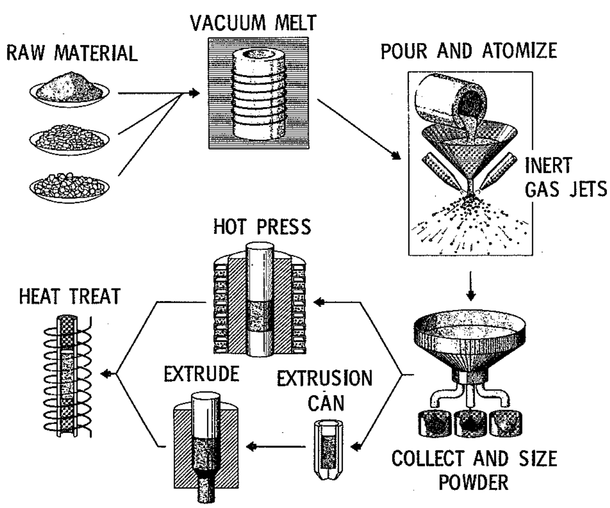


Figure 8.- Prealloyed powder processing.

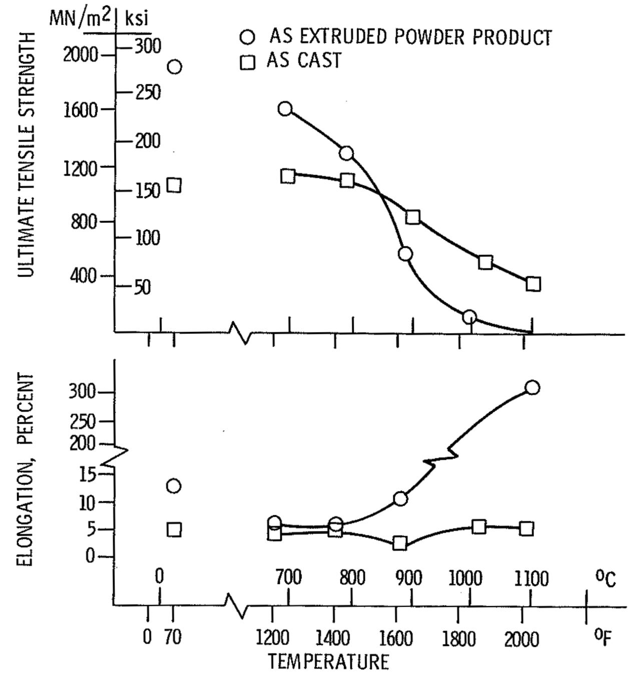


Figure 9.- Properties of NASA-TRW VI-A alloy.

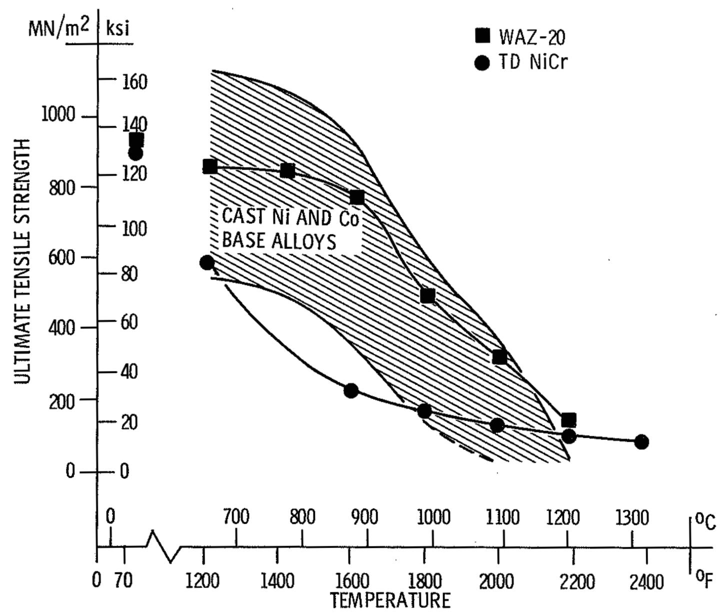


Figure 10.- Strength capability of cast superalloys and a dispersion-strengthened alloy.

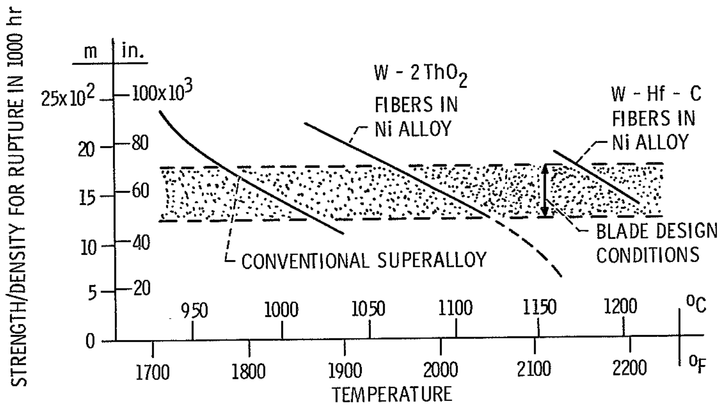


Figure 11.- Metal matrix composites for turbine blades.

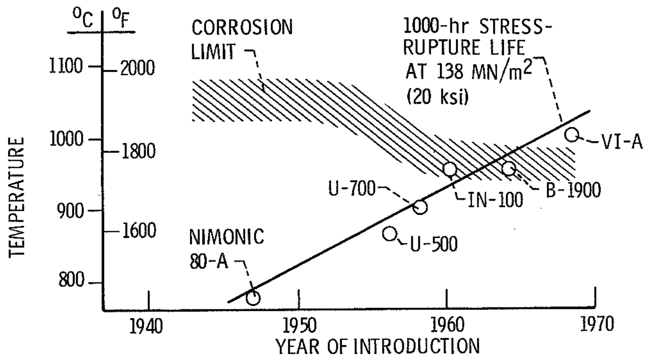


Figure 12.- Interrelation of corrosion and strength characteristics.

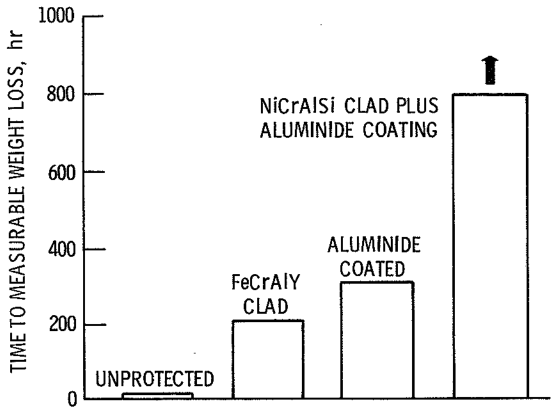
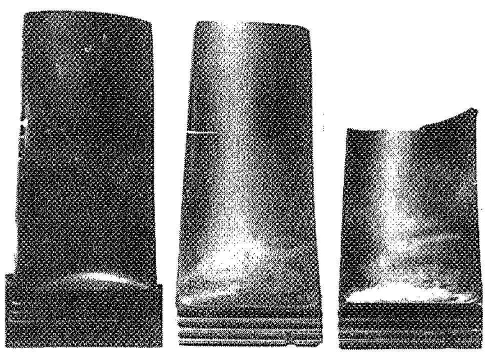


Figure 13.- Effectiveness of oxidation protection systems for alloy IN-100 in Mach 1 burner test.
(Cycle: 1 hour at 1093° C (2000° F); 3 minutes at room temperature)



55 hr 278 hr 306 hr

Figure 14.- Thermal fatigue of blade in engine operating environment.

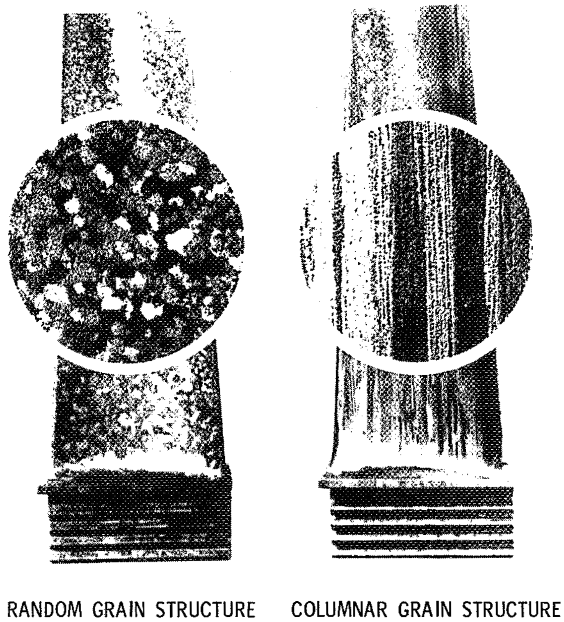


Figure 15.- Controlled grain structure in cast blades.

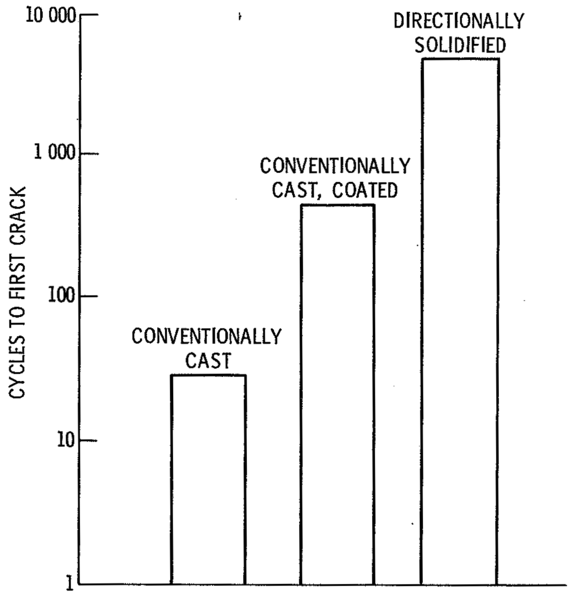


Figure 16.- Thermal-fatigue resistance of alloy IN-100. (Six-minute cycles at 316° C (600° F) to 1088° C (1990° F) in fluidized bed)

SUBSONIC AND SUPERSONIC PROPULSION

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INTRODUCTION

About a year ago a 3-day conference on propulsion was held at the NASA Lewis Research Center. The present paper is a broad overview of that material, and more of the detail is presented in the report of that conference (NASA SP-259). Also, an attempt is made to forecast trends, but without assurance that everyone will agree with the estimates.

Figure 1 provides an outline of the subjects to be discussed. It lists various kinds of airplanes and the types of engines that seem right for them. A conventional-takeoff-and-landing (CTOL) transport of the future could be a near-sonic airplane. It would need high-bypass turbofan engines with bypass ratios between 5 and 10. A short-takeoff-and-landing (STOL) airplane might use either of two engines. If it had externally blown flaps, a very high bypass turbofan would be needed. The bypass ratio would be 10 or higher. If it had an augmentor wing instead, a medium-bypass turbofan would be used. The bypass ratio would be down around 4 and the fan air would be ducted inside the wing to the augmentor flap. For some months now the aircraft industry has been studying vertical-and-short-takeoff-and-landing (V/STOL) aircraft that are powered by integral lift fans or else by remote lift fans. In the latter case a separate engine supplies working fluid to the lift-fan unit. A future supersonic transport (SST) might use low-bypass turbofan engines. The bypass ratio would be down around 1.5 or less. For general aviation a geared turbofan could provide the right balance between performance and cost.

Table I lists the engine types that are illustrated in figure 1 and some areas of advanced technology that are important for each engine. Some of these areas could be important for several engines, but for conciseness they will be discussed only once. For the CTOL engines the focus will be on noise, turbine temperature, and pollution. A major problem with both types of STOL engines is the lift-augmentation noise. For V/STOL engines the requirements of component packaging introduce some new problems. Important areas for an SST are cycles, inlets, compressors, and nozzles. General aviation could gain a real boost from low-cost concepts for producing turbine engines.

The noise discussion will begin with a brief consideration of FAR 36, the FAA regulation concerning noise. This regulation specifies the maximum noise at three measuring stations, shown in figure 2. On airplane approach, the measuring point is 1.85 km (1 n. mi.) prior to touchdown. For a 3° glide slope, the aircraft altitude at this point is 113 meters (370 feet). On takeoff the measuring point is 6.48 km (3.5 n. mi.) from the point of brake release. Altitude at this point depends on the particular aircraft and flight procedures used. The third measuring point is on a line parallel to the runway at a distance of 0.65 km (0.35 n. mi.) for four-engine aircraft and 0.46 km (0.25 n. mi.) for three-engine aircraft. FAR 36 specifies the permissible noise levels at each of these points as a function of aircraft gross weight.

In order to meet these noise goals, the sources and characteristics of engine noise must be understood. Figure 3 shows the internal noise sources associated with the turbomachinery. The noise producers are the fan stage, compressor, and turbine. Of these, the fan stage is usually the primary noise source. If fan noise is quieted enough, the compressor and turbine can become important sources. Noise from these sources radiates from the inlet and from the fan and core exhaust ducts. Fan noise is controlled at the source by the fan design. A low-noise fan results from a low fan pressure ratio and tip speed and a large separation between the fan rotor and stators. Also, the number of rotor blades and stator vanes is selected to keep the propagation of fan discrete-tone noise at a minimum.

Figure 4 shows the external or jet noise sources. This noise is caused by the mixing of the high-velocity exhaust jets with the ambient atmosphere. It radiates primarily from the aft part of the engine. Jet noise is strongly dependent on jet velocity, and the best way to reduce it is to keep the exhaust velocities low. Mixer-type nozzles have been used with some success to reduce the jet noise of high-velocity jets; however they have not helped appreciably at the lower velocities.

The spectral composition of these different noises will be examined next. Figure 5 is an engine spectrum with one-third-octave sound pressure levels shown as a function of frequency. This is the sound signature at a particular position with respect to the axis of the engine. At other positions the pattern is different. This directivity effect becomes important when determining the noise from an airplane flyover. The noise below about 800 or 1000 Hz is primarily associated with the jet. It is characterized by relatively low frequencies and the absence of discrete tones. The fan noise is composed of the higher frequencies, above 1000 Hz. It is characterized by discrete tones that occur at the fan blade passage frequency and its harmonics. The discrete tones are the cause of the fan engine's whine. Blade passage frequency is determined by the number of fan blades and the fan rotational speed. For many fans it falls in the frequency range from 1000 to 3000 Hz. There is another kind of fan noise called multiple pure tones or buzz-saw noise. This noise is shown by the narrowband analysis of the data in figure 5 in the frequency range near 1600 Hz. It is characterized by discrete tones that appear at multiples of the shaft speed of the fan. It is associated with the shock-wave system on the blades of transonic rotors and is influenced by irregularities from blade to blade. The frequencies of these tones are generally spaced so closely that a one-third-octave analysis will not resolve them. They are thus seen best in narrowband data.

Two terms that arise in noise discussions are PNdB and EPNdB, both of which involve the psychological response of people to sound. PNdB is a summation of sound pressure levels that have been weighted to account for human response. The largest weighting occurs in the frequency range from 2000 to 4000 Hz. Higher and lower frequencies are weighted less because people find them less annoying. EPNdB is a modification of PNdB to account for discrete-tone content, which is more annoying than broadband sound, and for length of exposure to the sound. EPNdB thus includes the effects of airplane speed and flight path, as well as a correction for discrete tones.

There is an important difference between fan noise and jet noise. The difference is that fan noise is generated internally and can be suppressed by acoustically treating the internal flow passages of the engine. Figure 6 shows how acoustic treatment might be employed. The engine shown is a cutaway of the NASA Quiet Engine that has the low-noise features described earlier. These are a large separation of the fan rotor and stators, a low-tip-speed fan, and the proper number of rotor blades and stator vanes. The engine has a bypass ratio of about 5.5, and because of the low exhaust velocities the jet noise is relatively low. Fan noise can be further reduced by acoustic treatment. In the inlet are three treated splitter rings and a treated outer cowl. In the fan duct are a single splitter and treated inner and outer walls. In addition there is treatment in the core engine inlet for compressor noise and in the core exhaust duct for turbine noise. It is possible that, with sufficient treatment, the fan noise and other noise could be suppressed to such a point that jet noise would again become the controlling noise. Further noise reduction would then require reducing the jet noise. One way that this might be accomplished is through the use of variable-area exhaust nozzles. Increasing the nozzle area reduces jet noise by decreasing the jet velocity. The effect is similar to throttling the engine but the process may be more efficient. The use of variable-area nozzles can have an impact on noise only if fan noise is suppressed so that jet noise is dominant.

Acoustic treatment such as that shown in figure 6 has been experimentally tested. Figure 7 shows a full-scale 1.8-meter (6-foot) inlet that has been tested on the fan rig at Lewis Research Center. The inlet has three treated splitters and a treated outer cowl. With this inlet about 13 PNdB of sound suppression was obtained.

Lewis Research Center has been working on a low-noise engine incorporating the sound reduction features that have been discussed. This engine, called the Quiet Engine, is being built under contract with the General Electric Co. The engine has been undergoing tests with one of the candidate fans for the past month or so. Figure 8 shows one of the candidate fans for the engine, fan C. It is 1.83 meters (6 feet) in diameter and has a design pressure ratio of 1.6 at a tip speed of 472 m/sec (1550 ft/sec). There are 26 fan blades. This fan is currently being tested on the fan rig. Figure 9 shows the Quiet Engine on the test stand at General Electric's Peebles facility. The engine is being tested with fan A, which has a design pressure ratio of 1.5 at a tip speed of 354 m/sec (1160 ft/sec). It has 40 blades. The fan was tested on the fan rig earlier in 1971 and was selected for the engine tests on the basis of its acoustic and aerodynamic performance.

Table II shows EPNdB values calculated from preliminary data for the Quiet Engine at takeoff and approach. The calculation is for a four-engine aircraft such as the 707 or DC-8. Data are shown for the Quiet Engine with an untreated nacelle and with a treated nacelle. The treated nacelle incorporates the treated inlet described previously. The fan duct has a single treated splitter and treated inner and outer walls. For reference, the EPNdB levels specified by FAR 36 and the levels of a 707 or DC-8 airplane are also shown. The unsuppressed engine is about 6 EPNdB below FAR 36 requirements at both measuring points. With suppression, the levels are 11 and 10 EPNdB below FAR 36 at takeoff and approach, respectively.

Noise estimates for the near-sonic airplane are shown in figure 10. Perceived noise levels in PNdB are shown relative to FAR 36 as a function of design fan pressure ratio. PNdB rather than EPNdB values are shown because of the difficulty of correcting for discrete-tone noise and of estimating flight path and speed for "paper" engines. Cycle studies of engines for this airplane have shown that the limiting noise occurs during airplane approach, when the engines are throttled back and fan noise is dominant. The curves show the level of fan approach noise after 15 PNdB of suppression. It appears that the FAR 36 goal of 106 EPNdB could be met with a two-stage fan having a design pressure ratio of about 2.25, and that a level 10 PNdB below FAR 36 could be reached with a single-stage fan having a design pressure ratio of about 1.7. A noise increment of 5 PNdB is shown between one- and two-stage fans. This difference is probably close to the maximum that could be expected. The exact difference may be less and is the subject of some controversy. Additional work will be required to resolve this point. Unfortunately, reaching these noise levels causes airplane performance to be reduced.

Figure 11 shows how advanced technology might pay the price of having lower noise in a near-sonic airplane. Relative direct operating cost (DOC) is plotted against noise level, with an indication of the FAR 36 noise requirement and noise levels 10 and 20 dB lower than FAR 36. In this figure the airplane is fixed while the engine is changed to achieve lower noise levels. With current technology, a low DOC can be achieved and FAR 36 can be satisfied. If lower noise is required, a different engine that weighs more and uses more fuel must be used. Thus, the price of lower noise is higher DOC. From right to left along the current-technology curve, fan pressure is decreasing and bypass ratio is increasing in order to lower turbomachinery noise and jet noise. Also, more suppression is used in the inlet and fan ducts. The break in the curve accounts for the change from a two-stage fan to a one-stage fan. FAR 36 minus 10 dB is achieved with a 7.5-percent increase in DOC. The upper shaded band shows how the situation improves as noise technology advances. The lower bound assumes a reduction in source noise of 5 dB and an improvement in noise suppression of up to 20 dB.

The lower shaded band indicates that further improvements are possible with advances in turbine technology. Turbine temperatures as high as 1810 K (2800° F) were used with advanced turbine cooling schemes and better high-temperature materials. With advances in both these technologies, FAR 36 is achieved with a 6-percent reduction in DOC. For a 20 dB lower noise, the DOC increase is about 3 percent.

Rapid increases in turbine inlet temperature have been occurring in the past and will continue in the future, as shown in figure 12, which is a trend plot of turbine inlet temperature over the years. The temperatures shown are at takeoff and the time is the year when each of several engines was first introduced into service. In the early sixties the temperatures were low, 1145 to 1310 K (1600° to 1900° F), with those of military engines being greater than those of commercial engines. The blading used then was uncooled, and materials limited the temperature levels. In the late sixties cooling was brought into service. This allowed a step increase in temperature without exceeding the limits of the materials. Cooled turbines first showed up in rigs, then in military engines, and finally in commercial engines. As indicated previously, temperatures on the order of 1810 K (2800° F) will be needed in future transports. If the trend plot of figure 12 is a reasonable indicator, these temperatures will first show up in military engines in the middle of this decade and then in commercial engines in about the mid-eighties.

The methods that are used to cool turbine blading must result in blades that are structurally sound, are reasonable in cost, require a low cooling-air rate, and have little adverse effect on turbine aerodynamic performance. Figure 13 shows the three methods of blade cooling that are currently being considered as most attractive. The first method is convection cooling, where the heat is transferred from the metal to the cooling air by means of convection. The second method is film cooling, where the cooling air, after some convective heat transfer, passes out of the blade through a series of holes and forms a local film of air to keep the hot gas from the blade. The third method is transpiration cooling, where the blade is built of porous material through which the cooling air oozes out and blankets the blade, thus keeping the hot gases away. Examples of blading using these schemes are shown in figure 14. The blade on the left utilizes a combination of convection and film cooling. Some of the cooling air goes out the trailing edge and some goes out through vertical rows of holes on the pressure side of the blade. The blade on the right is of the transpiration type, with a porous shell attached to an inner strut by means of electron beam welding.

Figure 13 also shows the effect of temperature on the relative coolant flow required for the three methods. The convectively cooled blade is limited in temperature capability because of the large amounts of cooling air required. The transpiration cooling method offers the best potential of all. However, its future in terms of being a practical method is still uncertain. Film cooling, on the other hand, permits the use of increased temperatures with acceptable coolant flow and can be accomplished with a structurally sound blade. However, to exploit the film cooling concept fully, many rows of holes must be used rather than the two rows shown in figure 14. Much work still is required to determine the best cooling configuration for high temperatures. It is probable that the ultimate configuration will utilize a combination of the methods described here.

As attempts are made to increase the performance of engine components, the pollutants the engine produces must be considered. The new federal air quality standards are supposed to be achieved by the mid-seventies. Regulations for emissions from automobiles have already been established, and similar standards for jet aircraft emissions are expected to be established near the beginning of 1972. The airlines and aircraft industry have anticipated these regulations for aircraft emissions and have already made considerable progress in reducing smoke in the exhaust of jet aircraft, as shown in figure 15. The upper part of this figure is a photograph during takeoff of a 727 aircraft that is equipped with the original JT8D engines. The bottom part of the figure shows the takeoff of a similar aircraft that has been retrofitted with JT8D engines with redesigned combustion chambers that reduce smoke emissions. Similarly, more recent engines such as the CF6 and JT9D have been designed to have smoke emissions that are well below the visible threshold.

Regulations on aircraft emissions are expected to include limits on gaseous pollutants which are produced during a defined landing-takeoff cycle. This cycle would cover all aircraft operations below an altitude of 915 meters (3000 feet). A limit on smoke emissions would be established by setting a maximum value on the SAE Smoke Number. Table III lists these major pollutants, together with proposed techniques for reducing them. Hydrocarbons and carbon monoxide are highest during engine idle, while oxides of nitrogen and smoke are highest during takeoff. A range of emission values at these critical operating conditions is shown for typical commercial engines in use. The concentrations of the gaseous constituents in the exhaust are expressed in terms of the grams of pollutant per kilogram of fuel burned. High hydrocarbon and carbon monoxide emissions during idle are due to inefficient combustion. Inefficient combustion is caused by a combination of poor fuel atomization, lean fuel-air ratios, and low combustor pressure and temperature. High nitric oxide emissions at takeoff are caused by the chemical reactions of oxygen and nitrogen at high flame temperatures. Smoke density is also higher at takeoff because of high pressures and rich fuel-air ratios.

In order to meet standards expected in the mid-seventies, hydrocarbons and carbon monoxide emissions during idle would have to be reduced by as much as 90 percent, while nitric oxide emission during takeoff would have to be reduced by as much as 75 percent. It is hard enough to design a good combustor without having to make it free of pollution. Therefore research is in progress to gain a better understanding of the problem and to devise new combustor design techniques to reduce emissions without sacrificing engine performance and reliability. Some of these combustor design techniques are pointed out in table III.

One approach is to change the fuel-injector design. Improving fuel atomization by using an air-assist fuel nozzle can significantly reduce unburned hydrocarbon and carbon monoxide emissions at idle operating conditions by improving combustion efficiency. This might be done as shown in figure 16. Here a conventional dual-orifice fuel nozzle is modified so that during idle high-pressure air is injected through the secondary orifice. Only small amounts of air are needed to make it work. For higher power settings the air would be shut off and secondary fuel would be injected in the conventional manner.

Improved fuel-injector designs such as air-blast nozzles may accomplish the same purpose. NASA has performed a great deal of research on the swirl-can element shown in figure 17. Each element, which is about 5 cm (2 inches) in diameter, consists of three basic parts: a carburetor, where air and fuel are introduced; a swirler, where the fuel and air are further mixed and a swirl is imparted to this mixture; and finally, a flame stabilizer, where combustion is initiated and maintained. Combining these elements into a modular array as shown in figure 18 produces a relatively short-length combustor. This particular combustor is 107 cm (42 inches) in diameter and consists of 120 individual swirl-can elements arranged in three concentric annular rows. The swirl-can combustor has the potential for operating efficiently over a wide range of fuel flows. It does this by premixing the fuel and air and by distributing combustion uniformly across the combustor.

Another method being studied to reduce emissions is modification of the combustion-zone designs. Fuel staging can be used as shown in figure 19. During engine idle or relight the objective is to operate at locally higher fuel-air ratios in the primary zone to achieve higher combustion efficiency without increasing overall engine fuel flow. This might be accomplished by providing separate fuel manifolds and controls for separate radial zones of fuel injection, as in the swirl-can combustor shown in figure 19. During idle, fuel would be injected into only the top row of swirl-can elements. At higher power settings, fuel would be injected into all three rows.

Fuel staging may be combined with the concept of airflow distribution control, also shown in figure 19. Combustion efficiency is improved at idle by shifting the combustor airflow distribution so that less air is introduced into the upper region of the primary zone, thereby increasing the local fuel-air ratio in addition to lowering local velocities. To do this job by means of variable geometry is possible but is not desirable because of the high-temperature environment. An alternate technique for varying this airflow is to equip the diffuser with wall bleed as shown in figure 19. Tests with wall bleed have demonstrated that the radial velocity profile may be shifted to either the hub or the tip by independently controlling the quantity of bleed on the inner and outer walls of the diffuser. During idle, when combustor pressure drop is low, wall bleed would not be used; however, the combustor inlet would be designed unsymmetrical with respect to the combustor liner in order to allow most of the air to bypass the upper region of the primary zone. During takeoff and cruise, when the combustor pressure drop is higher, wall bleed would be used to adjust the inlet radial velocity profile so as to introduce more air into the primary zone. It is anticipated that this could be done with about 4 percent bleed flow. Engine cycle efficiency would not be sacrificed because this bleed air could be used for turbine cooling. The same technique could be applied to improve altitude windmill relight capabilities.

The methods just described can reduce hydrocarbon and carbon monoxide emission during idle by increasing combustion efficiency to levels of the order of 98 percent. However, these improvements may not be good enough to meet the expected standards without also reducing engine idle time.

Over the past several years of combustion research, one goal has been to reduce combustor length in order to minimize combustor wall-coolant requirements and to minimize the overall engine weight. Fortuitously, the short-length combustor technology developed for these purposes provides a combustor with a short dwell time that has been shown to be effective in reducing the formation of oxides of nitrogen (table III). The chemical reactions that produce nitric oxide are considerably slower than the combustion reaction. Therefore, reducing dwell time prevents the nitric oxide reaction from going to completion and less nitric oxide is formed. Tests performed on the swirl-can combustor described previously have shown nitric oxide emissions significantly lower than those from a conventional combustor operating at similar conditions.

Fuel prevaporization or fuel-air premixing reduces nitric oxide formation by reducing flame temperature. Reducing flame temperature reduces the rate of the chemical reactions that form nitric oxide. Nevertheless, the reduction of oxides of nitrogen in high-pressure-ratio engines may require a further reduction in flame temperature by the use of water injection.

Techniques for eliminating visible smoke plumes by improving fuel-air mixing and avoiding fuel-rich regions are already being utilized in advanced gas-turbine engine designs. However, these methods that are used to reduce smoke also reduce altitude relight capabilities. Present low-smoke engines have sufficient relight performance, but there may be a problem in the future because of higher compression ratios or requirements to reduce particulate quantities even in nonvisible plumes. The techniques described to improve combustion efficiency at idle could help alleviate this design problem.

In general, the compressor pressure ratio has a big effect on these pollutants. Hydrocarbons and carbon monoxide emissions at idle are more difficult to control for low-pressure-ratio engines, while nitric oxide emissions and smoke at takeoff are more difficult for high-pressure-ratio engines. The trend in large jet engines to higher pressure ratios suggests that in the future the control of both oxides of nitrogen and smoke may become a more challenging problem.

Table IV summarizes some characteristics of CTOL engines. It compares a current engine with a low-pollutant engine of the future. A near-term goal is to reduce noise by 10 dB, while a longer term goal is to reduce noise by 20 dB. The higher thrust-weight ratio of the future engine represents a 23-percent improvement; this engine uses the advanced materials described by J. C. Freche in a previous paper. Although cruise SFC of the near-sonic engine is up somewhat, the 3-point rise in engine efficiency represents a 10-percent gain over the current value. With advances in turbine technology the future engine could be cruising at temperatures as high as 1810 K (2800° F). The overall pressure ratio may rise to 36 and the bypass ratio to 9. Depending on what is learned from fan-noise studies, the future engine will have a fan pressure ratio between 1.5 and 1.9.

STOL ENGINES

As indicated in table I, the discussion of STOL engines will deal primarily with lift-augmentation noise. Figure 20 shows the two STOL propulsion systems of principal interest. In the externally blown flap system, the air and gas from the turbofan engines blow over and under the large double-slotted flap. This action provides the required high lift, but unfortunately increases the noise level considerably. In the augmentor-wing STOL airplane, there is ducting inside the wing to get the engine fan air to the augmentor flap. Once again, the flap produces high lift but at the expense of a new noise source. The internally blown flap system, which is not shown, is similar to the augmentor wing but a single large flap is used rather than the augmentor flap. To work properly, the augmentor flap and internally blown flaps require a pressure ratio greater than 2. Thus, a multistage fan with a pressure ratio over 2 is required. The cycle for the engine is selected so that engine core velocity is quite low. Thus, jet noise from the engine does not constitute a problem.

For the externally blown flap (EBF) system, a very high fan bypass ratio would be selected to keep engine noise low, but an additional noise source is present. This is the noise caused by the high-velocity exhaust jet scrubbing over the flap system. Figure 21 shows a half-scale model of an EBF wing that has been used to measure this noise. The engine is simulated by a coannular nozzle in which the two flows can be varied independently. To give an idea of the size of the model, the nozzle is 58 cm (23 inches) in overall diameter. The noise from the model has been measured as a function of jet velocity and of flap angle. Figure 22 shows some typical measurements. Sound pressure level is shown as a function of frequency for undeflected flaps and for flaps deflected to the 30°-60° position. The primary jet velocity was 236 m/sec (775 ft/sec) and the secondary jet velocity was 189 m/sec (620 ft/sec) for both the undeflected and deflected flaps. The data show the noise to be primarily at the low end of the frequency range. The curves also show the rather large increment in noise when the flaps are lowered to the 30°-60° position, caused by the exhaust jet scrubbing over the flap surfaces. One approach to reduction of this noise is the use of mixer nozzles to lower the velocity of the jets prior to contact with the flap system. Figure 23 shows a mixer nozzle that is being tested on the EBF wing model. Preliminary experiments indicate that reductions of several decibels are possible with this approach. Fan noise is not anticipated to be an insurmountable problem for EBF engines because of their low fan pressure ratio and tip speeds. The use of acoustic treatment will probably be limited by the sensitivity of the engine to losses at this low fan pressure ratio.

The augmentor wing (AW) will be discussed briefly. As indicated earlier, it needs a high fan pressure ratio, and thus fan noise coming out of the inlet is a big problem. One type of suppressor that could be used is a choked inlet, as shown in figure 24. The sketch indicates a multiple grid near the cowl lip, which accelerates the flow to high speeds so that acoustic waves from the fan are trapped. As indicated in the figure, the relative noise ahead of the inlet depends on the throat Mach number, and the noise level drops sharply at sonic speeds. Other mechanical devices can produce this same effect, but they all share the same kinds of problems, some of which are listed in the figure. One of them is distortion. As the throat Mach number is increased, distortion increases, and that could be very bad for the engine. In addition the inlet pressure recovery decreases so there is a loss of thrust. Another problem is the mechanical complexity that is needed to change the flow area for different power settings. This mechanism has to be fast and reliable for emergency power changes. It is expected that these problems will be solved with more work, and the sonic inlet looks like a good choice for this type of engine.

If a sonic inlet is used to eliminate fan noise, the controlling noise source is the jet noise from the augmentor flap. Figure 25 shows the augmentor-wing model used for noise tests. The wing chord of the model is 345 cm (11 ft 4 in.) with flaps retracted, or about full scale for a small airplane. Data have been obtained for several nozzle slot heights as a function of nozzle pressure ratio. In figure 26 typical variations of sound pressure level around the wing are shown. As the data show, with the flap deflected there are two noise lobes or peaks centered about the axis of the jet. With the flaps undeflected these lobes would occur at about 45^{0} above and below the horizontal plane. With the flaps deflected, however, the lobes are also deflected, so that one is directed vertically. This accentuates the noise on the ground during a flyover.

There are several possible ways in which this noise may be reduced. One direct way is by acoustically treating the internal surfaces of the flaps. Another involves changing the basic slot nozzle into a series of small elements. This approach is similar to that used with standard engine nozzles for noise reduction.

Figure 27 shows noise estimates for both the EBF and AW STOL propulsion systems. PNdB estimates for a sideline distance of 150 meters (500 feet) are shown as a function of fan pressure ratio for the EBF and as a function of nozzle pressure ratio for the AW. The unsuppressed noise is shown by the upper curve for each configuration. For the EBF with a mixer nozzle, the curve shows that the goal of 95 PNdB can be met at a fan pressure ratio of about 1.25. At this pressure ratio and with limited acoustic treatment, the fan noise should not be a factor. While the trends obtained with mixer nozzles have shown promise, they have not yet achieved the suppression needed to reach the 95 PNdB goal. This is an area for future research. For the augmentor wing, the 95 PNdB goal is shown to be obtainable with multielement slots and acoustically treated flaps at a nozzle pressure ratio of about 2.4. It is assumed that fan noise is not a factor with a sonic inlet, even though a two-stage fan at a pressure ratio of about 2.8 is required. As with the EBF system, the required level of suppression has not been demonstrated although the trends obtained are promising. This too is an area for more research.

Table V summarizes characteristics expected for the EBF engine and the AW engine. For both, noise at 150 meters (500 feet) will be 95 PNdB. Engine thrust-weight ratio will be about 5 for the AW aircraft and somewhat greater than 5 for the EBF aircraft, both of which will use the advanced materials discussed earlier. Because of its higher bypass ratio, the EBF engine will have considerably better specific fuel consumption than the AW engine. Turbine temperature and overall pressure ratio will be moderate for both. Bypass ratio for the AW engine will be at today's level but that for the EBF engine will be 10 or higher. Fan pressure ratio for the EBF engine will be 1.3 or lower to meet the noise goal, while that for the AW engine will be 2.3 to 2.8 and its noise will have to be quieted with a sonic inlet.

V/STOL ENGINES

The V/STOL engines must serve multiple functions and, as indicated in table I, component packaging poses some new problems. An example configuration illustrating the required engine functions for a VTOL aircraft is presented in figure 28. This aircraft is in no way intended to represent a preferred system. Three engine functions are shown being accomplished by separate engine systems. The lift fans must provide the necessary thrust to lift the aircraft off the ground. The control fans are required because there is no forward speed to provide aerodynamic control forces during takeoff and landing. Finally, there are the engines for the cruise part of the mission. In an actual aircraft, it is highly desirable that all these functions be provided by the same engine system.

Noise constraints dictate the engine system design for a civilian VTOL transport. It becomes necessary to use high-bypass-ratio fans with low pressure ratios of about 1.25. With these pressure-ratio requirements, the fans themselves become rather well defined. The reasons for considering several different fan systems lie in the question of how to drive the fans.

In general, the drive systems can be put into two categories: integral fans and remote-driven fans. Figure 29 shows the integral-fan engine. It is in principle a conventional two-spool high-bypass-ratio turbofan engine. The single-stage low-speed fan has the potential of being very lightweight because of the possible use of advanced composite materials. The fan is shown with the flow to the compressor split off from the bypass flow. Such a geometry is attractive because it permits the pressure rise in the core section to be lower than that in the bypass section, and thereby allows a smaller inner radius for the fan. This means a smaller diameter engine. The inner spool has a multistage compressor driven by a single-stage turbine. The combustor is a reverse-flow type, which shortens the length of the engine. The fan is driven by a multistage turbine. Because of speed constraints on the fan and because the turbine diameter is much less than that of the fan, the turbine blade speeds and stresses are very low. The turbine aerodynamic design is way beyond that of conventional practice because the blade speed is low and the work output per unit of generator flow is high. Turbines of this type are called "high work factor" turbines, these factors being over twice those of current cruise-engine turbines. The principal effect of a high work factor on the turbine is a reduction in efficiency. Studies are currently under way to determine what efficiency levels can be expected and to explore possible methods of keeping efficiency as high as possible.

Figure 30 shows examples of the remote-drive system. In this system the power source is located remotely from the fan and the pneumatic energy is transferred to the fan module through ducts. These ducts can be located in such a way that the fans and power sources can be interconnected to reduce the engine failure and control problems.

Two remote systems are shown. The first is a gas-generator system, wherein a conventional turbojet is used. Rather than being used for thrust, the hot pressurized gas is diverted through the ducting to the fan module. The other system uses an air pump. This air pump would be a low-bypass-ratio, high-pressure-ratio, turbofan engine. The pressurized air, which is cooler than the gas-generator exhaust, is routed through the ducts into an auxiliary combustor, which heats the air just before it gets to the fan module. The gas coming out of the air-pump core could be used for lift, control, or cruise thrust.

A cross-sectional view of one of the fans is shown on the lower part of figure 30. The fan rotor, stator, and scroll and a single-stage tip turbine are shown. The configuration is basically the same for both the gas-generator and air-pump systems.

The discussion here has been confined to a description of the various fan systems of potential use for lift-fan transports. The ultimate choice of system is a function of many factors, some related to the engine (weight, fuel consumption, volume, depth) and others that are related to the aircraft operation (hover control, engine out, transition performance, etc.) Studies are currently in process to aid in the selection of the most promising system.

Table VI shows the characteristics of integral and remote engines for future V/STOL aircraft. Both will result in a noise level of 95 PNdB at 150 meters (500 feet). The integral engine will probably turn out to be 25 percent lighter than the remote. The right-hand column lists the characteristics of the power source for the remote engine. Turbine temperature is moderate and overall pressure ratio is low. Fan pressure ratio should be 3.6. A bypass ratio of 0 to 2 indicates that the power source can be a turbojet or a turbofan. Getting back to the comparison of integral and remote systems, fan pressure ratio will be about 1.25 to meet the noise goal, while bypass ratio will be 13 for the integral and 10 for the remote. The integral engine enjoys a 10-percent hover SFC advantage over the remote engine.

SST ENGINES

Moving from the subsonic area into the supersonic area, the discussion, as indicated in table I, will cover cycles, inlets, compressors, and nozzles. Figure 31 shows the performance penalty entailed in trying to meet FAR 36. With an afterburning turbojet (ABTJ) a long range can be achieved, but only with a very noisy engine. Noise can be reduced by oversizing the engine and taking off at partial power, but then the airplane range falls off rapidly. Somewhat better results can be achieved with an afterburning turbofan (ABTF), but the range penalty that accompanies a noise reduction to FAR 36 is still excessive. With the addition of a jet suppressor giving 6 dB suppression for a 6-percent thrust loss, it appears that the afterburning turbofan could meet FAR 36 with a moderate range loss.

With such an engine it is very important to incorporate efficient lightweight inlets and exhaust nozzles. During supersonic flight the inlet produces more of the thrust than any other part of the engine. As shown at the top of figure 32, the inlet uses multiple shock waves at supersonic cruise to slow the air to subsonic speeds before it enters the engine. The oblique shocks are produced by the centerbody and cowling, and the last wave is a normal shock which can be moved around by changing the back pressure at the end of the diffuser. For high performance it is pushed up to the minimum area called the throat. At a Mach number of 3 the static pressure in the diffuser is about 30 times the free-stream pressure. These high pressures push against the diffuser walls and produce about 70 percent of the total thrust.

At off-design speeds one of the problems with the inlet is that the throat area is too small. Therefore the cowl or centerbody has to be changed to increase it by a factor of 2 or more as indicated in figure 32. At some of the off-design speeds, a bypass door is also needed to spill some air that the engine can't use. At takeoff speeds the inlet flow area must be increased even more so that the distortion of the air caused by the sharp lip of the cowl will not stall the engine. In this case the bypass system could operate in reverse as auxiliary inlets. During flight at low altitudes it also is necessary to choke the inlet to suppress compressor noise. Obviously there is a need for high-speed inlet controls which are integrated with the engine and aircraft control, and it is expected that directions will be supplied by onboard digital computers.

A fundamental factor in inlet design is whether these changes in geometry will be effected with two-dimensional or with axisymmetric centerbodies. Another fundamental choice is the relative amounts of external and internal compression to be used ahead of the throat. External compression is the flow-area contraction ahead of the cowl lip, and internal compression is that occurring between the cowl lip and the throat. One big advantage of increasing internal compression is that it reduces the cowl angles and hence the drag.

Both decisions affect supersonic cruise range, as is shown in figure 33, where range is presented as a function of internal contraction. Cruise speed is assumed to be around Mach 2.5 to 3.0. Although there are only a few data points, it looks as though high internal contraction helps axisymmetric inlets more than it helps two-dimensional inlets. This is because it is easier to achieve high pressure recovery without a great deal of boundary-layer bleed inside the axisymmetric inlet. It should be pointed out, though, that this figure only takes into account the aerodynamics of the inlet, and certainly the mechanical factors such as structural weight also affect this kind of comparison. The final choice depends on the airplane and engine being used. In either case, however, the trend to high internal contraction is obvious in the figure.

A big problem, though, with internal contraction is that disturbances can cause the shock waves to become unstable. When this happens the normal shock pops out ahead of the cowling and the drop in pressure causes engine stall. This unstart can be triggered by wind gusts or by sudden changes in engine airflow.

Two solutions to this problem which probably can be developed are shown in figure 34. The control system which senses the normal-shock-wave position and which varies the overboard bypass flow could be speeded up to absorb some of the engine disturbances. Another aid is to use a large bleed region in the cowl ahead of the normal shock. This bleed passage is back-pressured by pressure-relief valves to prevent flow from being spilled during cruise. If the normal shock is then perturbed ahead of the throat, the higher pressures in the bleed duct automatically open these valves and stabilize the shock waves by spilling some of the flow. More effort is needed, though, to make these valves work right as an aircraft system.

During all these gyrations of an inlet, the airflow distortion at the end of the diffuser must stay low so that it will not cause compressor stall. This is an old problem, but a relatively new finding is that the unsteadiness of this distorted flow also affects the stall margin. To illustrate this dynamic distortion effect, the left side of figure 35 shows the steady-state total-pressure contours at the end of an axisymmetric inlet at zero angle of attack. (Here p_t is local total pressure, $\bar{p}_t$ is average total pressure, and $p_{t,\infty}$ is free-stream total pressure.) The normal shock is pulled back of the throat into the subsonic diffuser. The dark areas are low pressures and the light areas are high pressures. The distortion pattern is radial and normally would not be too bad in the engine. If these pressures are measured with high-speed transducers the flow patterns change drastically, as shown on the right side of figure 35. The dynamic distortion shown here contains large circumferential zones which are much more likely to cause engine stall. This particular dynamic pattern was the worst one found just before the engine stalled.

This dynamic distortion effect is difficult to quantify in developing an inlet to be compatible with an engine. One thing that will help is a better method of controlling flow separation in the subsonic diffuser. In some cases vortex generators have been very effective if they were located properly. The high-speed controls discussed earlier will also be a big help in avoiding conditions that cause high distortion. They might also be used to derate the engine operation temporarily in order to increase the stall margin if the distortion seems to be getting too great. This distortion of the flow coming into the engine face can have an adverse effect on the compressor. Although the problem of distortion is not new, the sensitivity of the compressor to distortion is more critical as compressor advancements are made. For example, major improvements in the compressor in terms of compactness and low weight have occurred through the use of transonic stages. With such compressors, however, the tolerance to distortion is reduced. Therefore, it is necessary to evolve new approaches to compressor design that will increase the distortion tolerance while still allowing transonic stages. One approach, called casing treatment, that shows promise is illustrated in figure 36. On the left side of the figure is shown a typical compressor stage with a grooved ring installed in the casing at the rotor tip. Figure 37 is a photograph of half of this ring and shows the grooves, which in this case go circumferentially.

The effect of this treatment on the stage performance is shown on the right side of figure 36, where the pressure ratio is presented as a function of weight flow. Shown in the figure are the design speed curve and the stall line, both with and without casing treatment, and the operating line. With the treatment, there is a substantial shift in the stall line. For example, the flow margin at the operating point is about 3 percent without treatment, but approximately 8 percent with the treatment. Also, the stall pressure ratio has been improved considerably. The success of this approach has been so good that the concept is being incorporated in certain engines currently under development.

There is still much work to be done in this area. Although the advantages of the treatment are already being utilized, the reasons for the benefits are still not understood. Through additional efforts an understanding should be achieved, and these and other approaches to improvement of the compressor stall margin will be exploited to their fullest extent.

The exhaust nozzle is another part of the engine that becomes more important as the flight speed is increased. As shown in the upper left of figure 38, the nozzle could be a divergent ejector at supersonic cruise. At a flight speed around Mach 3 the nozzle pressure ratio is about 40, so the exit area is about four times the throat area. Since the static pressures inside the nozzle push on the walls and produce about 20 percent of the total thrust, nozzle efficiency has a big effect on aircraft range. At off-design speeds, though, the nozzle pressure ratio can be as low as 2 or 3, and therefore this divergence must be eliminated so that (as shown in the upper right of the figure) the nozzle is essentially convergent. A fairly complicated mechanism is required to decrease the exit area down to one-fourth of its original size. In addition, the throat area may vary by about 40 percent because of changes in engine power. During takeoff the jet noise is a horrible problem, and somehow the nozzle must change geometry to suppress the jet noise. The concept shown in the lower left of figure 38 is multiple tubes inside a mixing shroud. And finally, after landing the nozzle must reverse the thrust. A clamshell (lower right in fig. 38) is only one of the devices that might be used for this purpose. Like inlet design, nozzle design is a complex trade-off between the aerodynamic and mechanical requirements.

Some of the better ideas for nozzle design are shown in figure 39. On the left is a variable-flap ejector which works in the manner illustrated in the upper part of figure 38. An obvious disadvantage is the mechanical complexity which is indicated by the large number of hinges. Another problem is the seal leakage for all these moving flaps. The nozzle in the middle of figure 39 uses auxiliary inlets at low flight speeds to bring air inside the divergent shroud. Thus the shroud does not have to close as far and the mechanism is simpler. At supersonic cruise these doors would be closed and the nozzle would again work like the variable-flap ejector. An entirely different approach is shown on the right in figure 39. It is a plug nozzle that eliminates most of the hinges and seals. At high speeds the jet is expanded from the throat in the annular flow passage between the plug surface and the cylindrical shroud. At lower speeds the shroud is retracted upstream so that the exit area is decreased. To change the throat area the plug could be translated relative to the convergent flap. This plug concept appears to have many advantages for this kind of aircraft.

The off-design performance of these nozzles depends upon the interaction of the internal and external flows. To complicate matters, the external flow is distorted depending on the manner in which the engine is installed on the airplane. Figure 40 illustrates the kind of podded engine installation which seems good for this type of airplane. An F-106 has been modified so that engine pods can be hung directly beneath the wings. Each of the different nozzles shown in previous figures can be flight tested on these engine pods. This test technique is particularly useful at transonic speeds since the wind-tunnel models normally used have to be so small to avoid wall interference effects.

A jet-noise suppressor is another part of the nozzle that produces as many different designs as there are designers. Spokes, chutes, tubes, spades, scoops, shrouds, and liners can be used in various combinations. Some suppressors suitable for the plug nozzle are shown in figure 41. At the top is a chute arrangement, both with and without an acoustic shroud. At the bottom are two retractable suppressors – spokes and tube bundles that can be stored in the plug cavity. They are supposed to break up the big jet into a large number of small jets. This procedure reduces the mixing length downstream of the nozzle where the noise is being generated and changes its directivity. Also, the noise is shifted to higher frequencies which are more easily absorbed by the atmosphere. These suppressors are shown installed on the F-106 aircraft for flight tests to find the effects of external flow on their performance.

SST NOISE

The nozzles that are shown in figure 41 offer some potential for reducing SST jet noise; however the noise reduction is accompanied by a substantial thrust loss. Figure 42 shows the range of experience with mixer-type nozzles. Noise reduction in PNdB is shown as a function of thrust loss in percent. Data are shown from flyover tests and from static tests of various mixer nozzles. Several lines having different values of the ratio

ΔPNdB/ΔF are also shown. Engine throttling, represented by the dashed line, gives about 1 PNdB reduction in noise for a 3-percent thrust reduction. The suppressor nozzles are somewhat more effective than this. The data obtained to date from flyover tests fall within the band for current technology, with values of ΔPNdB/ΔF between 0.5 and 1.0. Most data from static tests fall within the band for preliminary research, with ΔPNdB/ΔF values between 1.0 and 2.0, but some values are larger than 2.0. The objective of future research will be to obtain these higher levels in full-scale tests.

Table VII compares a current prototype afterburning turbojet for an SST with a possible future afterburning turbofan. With the turbofan and a 6-dB jet-noise suppressor, it will be possible to lower noise to 108 PNdB, the FAR 36 requirement. The thrust-weight ratio is expected to improve 20 percent both for the installed engine and for the bare engine on the basis of the use of advanced materials. With advances in turbine cooling technology, turbine temperature may rise to 1810 K (2800° F). For the fan engine, a moderate overall pressure ratio will suffice, and a bypass ratio of 1.5 with a fan pressure ratio of 2.4 is expected. The afterburning turbofan promises the same high level of engine efficiency as the turbojet, 42 percent.

GENERAL AVIATION ENGINES

As indicated in table I, the discussion of general aviation aircraft will speculate on the feasibility of low-cost concepts. The small size and weight of the gas-turbine engine and its suitability for high-speed flight make it very attractive for advanced light aircraft. For the small private plane, however, the price of current small gas-turbine engines is too high. At the power levels required for high flight speeds, reciprocating turbo-charged piston engines are also quite costly. The Lewis Research Center has, therefore, been investigating means for reducing the cost of gas-turbine engines by a factor of, perhaps, 3; that is, reducing the price to the range of 1 to 2 dollars per newton (5 to 10 dollars per pound) of thrust.

In this program both turbojet and fan-jet engines are being designed and built. The fan jet provides greater aircraft range, greater takeoff thrust, and lower noise levels, and for light aircraft major interest has centered on this type of engine. A fan-jet engine configuration which is being considered at Lewis is shown in figure 43. This engine employs a geared front fan 38 cm (15 inches) in diameter, a five-stage compressor 25 cm (10 inches) in diameter, and a two-stage axial turbine. It uses a 485-kW (650-hp) gear box having a 2:1 speed reduction ratio for the fan drive. This design provides the fuel economy of the fan-jet engine and the low noise of a low-tip-speed fan, while still retaining a single-shaft two-bearing design for the core engine. The design also features sheet-metal construction of all the major rotor components, which will be further discussed later.

In order to achieve economy, the general approach has been to use very simple engines, designed to operate at moderate temperature and pressure ratio. This approach reduces the tip speeds, the stress levels, and the number of expensive stages required. For the fan-jet engine, a cruise turbine inlet temperature of only 980 K (1300° F) is used, along with a moderate engine pressure ratio and bypass ratio. These conditions will produce 4890 newtons (1100 pounds) of takeoff thrust and a specific fuel consumption of 0.094 kg/hr-N (0.92 lb/hr-lb) at a flight speed of 720 km/hr (390 knots).

These engine characteristics could provide a very high performance light airplane. A four-place light twin, for example, could take off in 305 meters (1000 feet) and have a cruise range of 1600 km (870 n. mi.). It is evident that achieving cost economy with such an advanced high-performance airplane will require fabrication development comparable to that being undertaken for the engine. This, however, must be the subject of other programs and later discussion.

A small turbojet engine has also been designed in accordance with this economy approach. This engine, shown in figure 44, has been designed for 2890 newtons (650 pounds) of sea-level static thrust and is being built in cooperation with the U.S. Navy in a program to determine the applicability of low-cost turbojet engines for missiles and drones. This engine uses a four-stage cast compressor and a single-stage cast turbine. The front bearing support, the compressor casing, and the rear bearing support are all simple castings. The combustor housing and liner are tubular sheet-metal assemblies. All these components are shown in more detail later.

This engine will have a specific fuel consumption of 0.133 kg/hr-N (1.3 lb/hr-lb) at a flight speed of 912 km/hr (495 knots). It has an external diameter of 29 cm $\left(11\frac{1}{2}\text{ inches}\right)$, and it will weigh only 40 kg (90 pounds). While it was designed primarily for application to drones, it could also provide a very useful range and payload for a light aircraft and could readily be developed for that purpose. The engine is currently being assembled and performance testing will soon begin.

Reduction of manufacturing costs is the essential factor involved in finally obtaining low cost on either of these gas-turbine engines. The remainder of this discussion will be concentrated on the techniques that are being investigated for achieving such cost reduction. The general approach is to use manufacturing methods in which the engine components can be quickly manufactured, using relatively inexpensive machinery and equipment, and keeping the man-hours of labor to a minimum. The first procedure meeting these requirements is investment casting of complete disk and blade assemblies for the compressors and the turbine. The cast compressor and turbine rotor which have been fabricated for the Navy turbojet program are shown in figure 45. Each of the four axial compressor stages were investment cast in a single piece and joined together by circumferential electron-beam welding. This compressor assembly was then electron-beam welded to a cast shaft. The compressor assembly was cast from 17-4 PH, a relatively inexpensive material which may be readily cast and which has a simple heat-treat cycle. Very sound castings with accurate blade profiles have been obtained, and these have been spin tested and have demonstrated a very good over-speed safety factor. The assembly technique based on electron-beam welding has, in general, proved to be very suitable and has resulted in sound attachments with very little distortion.

A similar approach is being followed for construction of the turbine rotor, which is also shown in figure 45. This assembly has been cast from Inconel 713 and, again, very sound, accurate castings have been obtained. This rotor has also been tested in the spin pit and a good over-speed safety factor has been obtained.

An alternate approach for obtaining low-cost axial stage rotors which is also being investigated is illustrated in figure 46. Here, combined disk and blade assemblies are stamped out of sheet metal and the blade profiles are formed by coining. To obtain the total number of blades required, a pair of bladed disk assemblies is fitted together. As illustrated in figure 47, these are then fitted into mating rings to form a complete rotor.

Several of these sheet-metal compressor rotors have been built and tested in our spin-test rig. During these tests, strain and clearance measurements were taken to compare with the design values for the rotor. The measured values correspond very well with calculated ones and certain versions of this rotor have been tested to 25 percent above their design speed, indicating a good strength margin and safety factor.

Sheet-metal construction techniques similar to those shown for the compressor rotor are also being used to build fan rotors and axial-flow turbine rotors. In general, the results achieved during tests of these rotors have agreed with expectations and indicate that the sheet-metal construction technique is attractive for fan and turbine rotors as well as compressors.

Fabrication development and performance testing has also been accomplished on a low-cost annular combustor, as shown in figure 48. This combustor is constructed from rolled cylinders with a simple pattern of punched holes. Commercial perforated sheet is used and this is oriented in a special way to provide a cooling airflow layer which keeps the liner substantially below the gas-stream temperature. The combustor has achieved good combustion efficiency as well as an attractive temperature-variation pattern factor. In addition, the low metal temperatures, well below the gas temperature, permit low-alloy materials to be used, and this combustor is therefore considered to be very promising for application to low-cost gas-turbine engines.

Accessories are also very important to the overall cost and size of small jet engines, since they tend to be large and expensive. The most critical accessory is the engine speed and fuel control, which must provide capability for rapid thrust response without surging the compressor or exceeding the allowable turbine temperature. The control is a key to the safety, reliability, and cost of the entire jet engine, and has been extensively investigated during this program.

The working parts of the simple hydromechanical control which have been investigated are shown in figure 49. This fuel control is based upon use of a zero-gradient pump-speed-sensing technique. A small gear pump, shown in figure 49, provides a fuel flow output proportional to engine speed. From this, a speed signal is obtained which works in conjunction with a hydraulic control and computation circuit to provide both speed governing and surge and blowout limitations. The figure also shows the two valves which provide the speed signal and speed governing, and two pressure-controlled valves which control the fuel flow limits during acceleration.

This control has been extensively analyzed and has been built and tested in actual operation of a jet engine. During these tests it has provided all the expected control functions and it also produced a very smooth acceleration of the engine with short response time.

For comparison, figure 50 shows the parts required for the fuel and speed controller of a current turbojet engine. The zero-gradient pump control is seen to have a smaller number of working parts. It is also simpler to assemble and adjust, it does not require close fits, and it has the cost advantage of being direct acting and not requiring servo-actuators.

Although other engine parts and accessories have been investigated, they will not be discussed here. In summary, it may be said that design and manufacturing experience has now been obtained with all the major components required for a gas-turbine engine suitable for application to general aviation. Complete engines will soon be tested and performance data and first-hand operational experience obtained.

Table VIII compares characteristics of a low-cost geared turbofan with a piston engine. The turbofan is somewhat quieter and has a 34-percent higher thrust-weight ratio. Its one deficiency is an 80-percent increase in cruise specific fuel consumption. However, the lower fuel cost and higher cruise speed of the turbofan engine result in a fuel cost per unit distance only 62 percent of that for the piston engine. The last line in the table indicates that the geared turbofan enjoys a comparable advantage in initial cost.

CONCLUDING REMARKS

If the noise of conventional transports is to be cut to a level well below FAR 36, much work is required to design quieter turbomachinery by developing more effective suppression. To offset the economic penalties of low noise, the engine needs to be lighter and more efficient. Therefore better materials and cooling methods will be needed. In order to have a clean engine, new combustors may be necessary. For STOL airplanes, hard work will be required to quiet the noise associated with propulsion lift. The noise goal is so difficult that some real advances in propulsion technology will be needed. The V/STOL engines need to be lighter and more compact. Better materials and high-work turbines are both important here. An SST engine needs clever design of the variable inlets and exhaust nozzles and their controls. It also needs as much jet-noise suppression as possible. For general aviation, a geared turbofan offers spectacular performance, and surprising reductions in cost seem possible.

If there is one point that stands out in an overview of propulsion, it is that noise limits and, to a lesser extent, pollution limits are designing future engines.

TABLE I.- AREAS OF ADVANCED TECHNOLOGY

Engine type	Important technology areas
CTOL	Noise, turbine temperature, pollution
STOL	Lift-augmentation noise
V/STOL	Component packaging
SST	Cycles, inlets, compressors, nozzles
General aviation	Low-cost concepts

TABLE II.- ENGINE-NOISE COMPARISON

	Takeoff EPNdB	Approach EPNdB
707 or DC-8	116	118
FAR 36	104	106
Quiet Engine:		
Unsuppressed	98	100
Suppressed	93	96

TABLE III.- PROPOSED TECHNIQUES FOR POLLUTION ABATEMENT

Pollutant	Critical operating condition	Typical emission levels, g/kg fuel	Technique
Hydrocarbons Carbon monoxide	Idle Idle	7 to 75 30 to 77	Air-assist fuel injection Improved fuel injectors Fuel staging Airflow distribution control
Oxides of nitrogen (as NO_2)	Takeoff	13 to 40	Reduced dwell time Fuel prevaporization Fuel-air premixing Water injection
Smoke	Takeoff	SAE Smoke No. 20 to 65	Improved fuel-air mixing Avoid fuel-rich regions

TABLE IV.- WHAT TO EXPECT IN CTOL ENGINES

Parameter		Current	Future
Noise, PNdB		106	96 to 86
Thrust-weight ratio		6.0	7.4
Engine efficiency, %		30	33
Turbine temperature	K	1490	Up to 1810
	^{o}F	2220	Up to 2800
Overall pressure ratio		30	Up to 36
Bypass ratio		4.4	Up to 9
Fan pressure ratio		1.6	1.5 to 1.9

TABLE V.- WHAT TO EXPECT IN STOL ENGINES

		Externally blown flap	Augmentor wing
Noise at 150 m (500 ft), PNdB		95	95
Thrust-weight ratio		>5	≈5
Cruise SFC	kg/hr-N	0.066	0.087
	lb/hr-lb	0.65	0.85
Turbine temperature	K	1370	1370
	^{o}F	2000	2000
Overall pressure ratio		21	21
Bypass ratio		10 or higher	4
Fan pressure ratio		1.3 or lower	2.3 to 2.8

TABLE VI.- WHAT TO EXPECT IN V/STOL ENGINES

		Integral	Remote	
			Lift fan	Power source
Noise at 150 m (500 ft), PNdB		95	95	
Thrust-weight ratio		12	9	
Turbine temperature	K	1645	1105	1560
	^{o}F	2500	1530	2350
Overall pressure ratio		10		12
Fan pressure ratio		1.25	1.25	3.6
Bypass ratio		13	10	0 to 2
Hover SFC	kg/hr-N	0.037	0.041	
	lb/hr-lb	0.36	0.40	

TABLE VII.- WHAT TO EXPECT IN SST ENGINES

Parameter	Prototype	Future
Noise, PNdB	128	108
Thrust-weight ratio:		
Installed	4.1	4.8
Bare engine	6.5	7.8
Turbine temperature K	1480	1480 to 1810
Turbine temperature °F	2200	2200 to 2800
Overall pressure ratio	12	10 up
Bypass ratio	----	1.5
Fan pressure ratio	----	2.4
Engine efficiency, %	42	42

TABLE VIII.- WHAT TO EXPECT IN GENERAL AVIATION ENGINES

Type	Piston	Geared turbofan
Noise at 0.46 km (0.25 n. mi.), PNdB	84	78
Thrust-weight ratio	3.1	5.9
Cruise SFC kg/hr-N	0.051	0.092
Cruise SFC lb/hr-lb	0.50	0.90
Rel. fuel cost per unit distance	1.0	0.62
Turbine temperature K	-------	980
Turbine temperature °F	-------	1300
Overall pressure ratio	-------	6
Bypass ratio	-------	2.5
Fan pressure ratio	-------	1.3
Cost for 4450-N (1000-lb) thrust	$17 400	$11 000

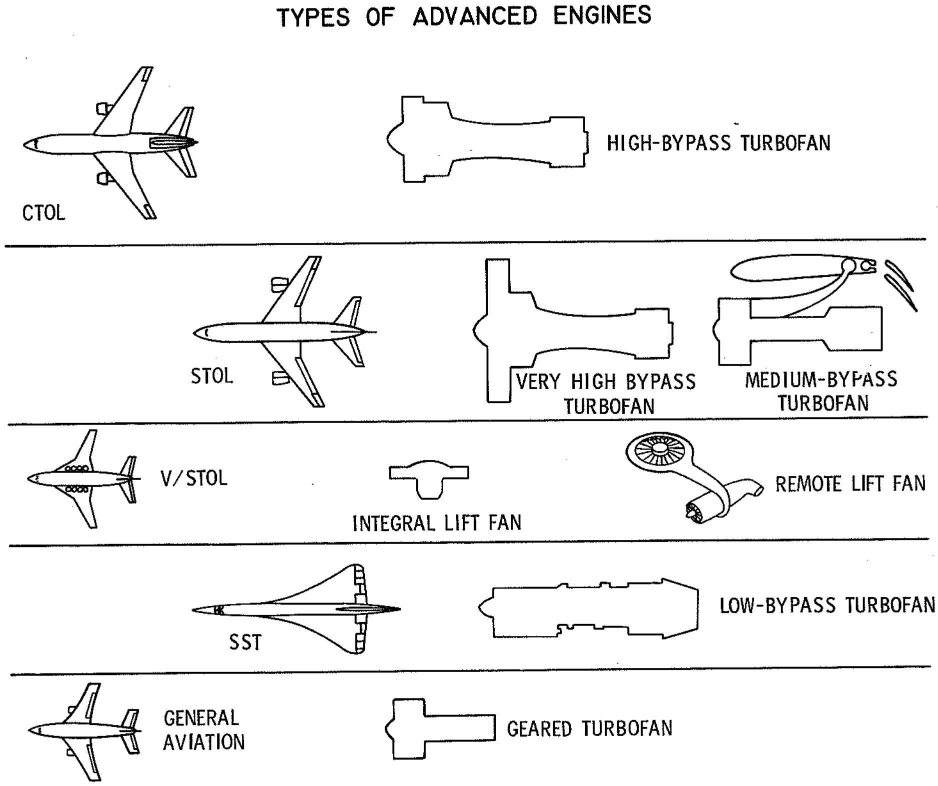


Figure 1

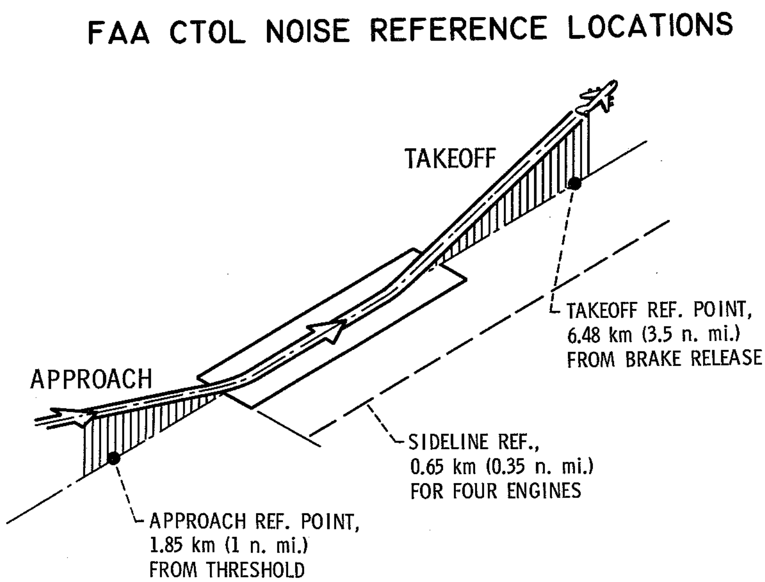


Figure 2

INTERNAL NOISE SOURCES

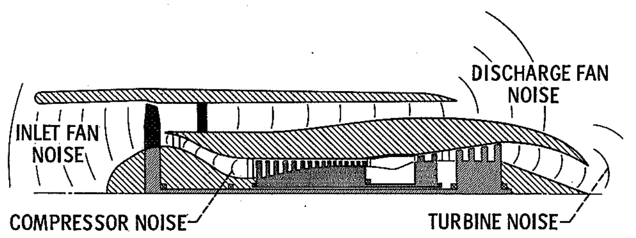


Figure 3

EXTERNAL NOISE SOURCES

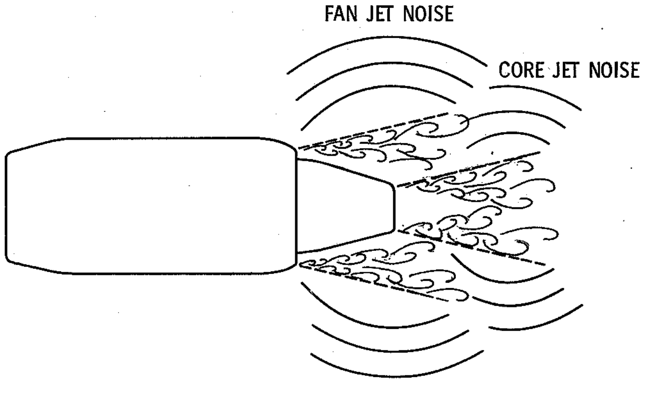


Figure 4

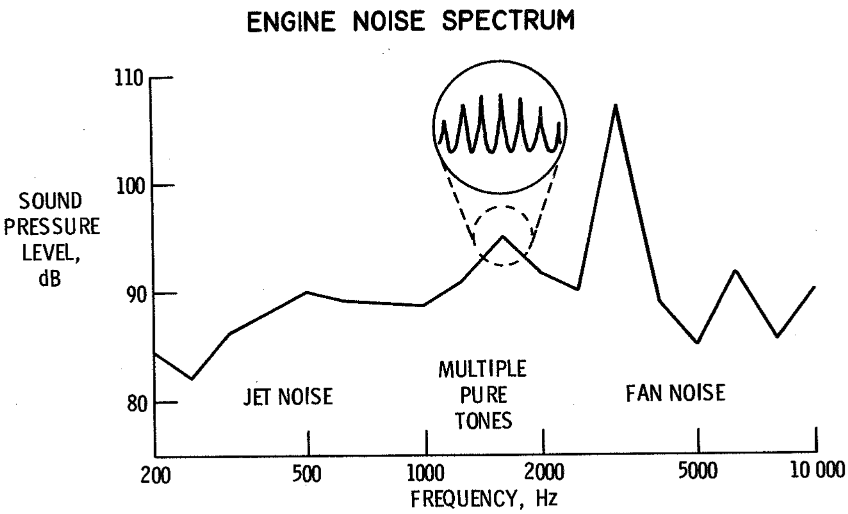


Figure 5

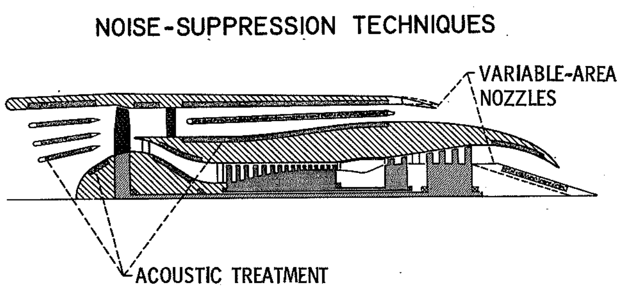


Figure 6

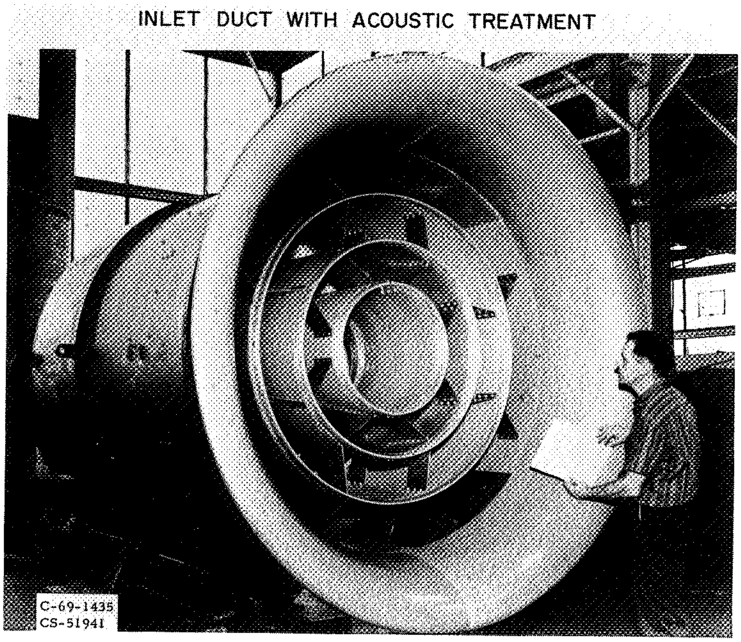
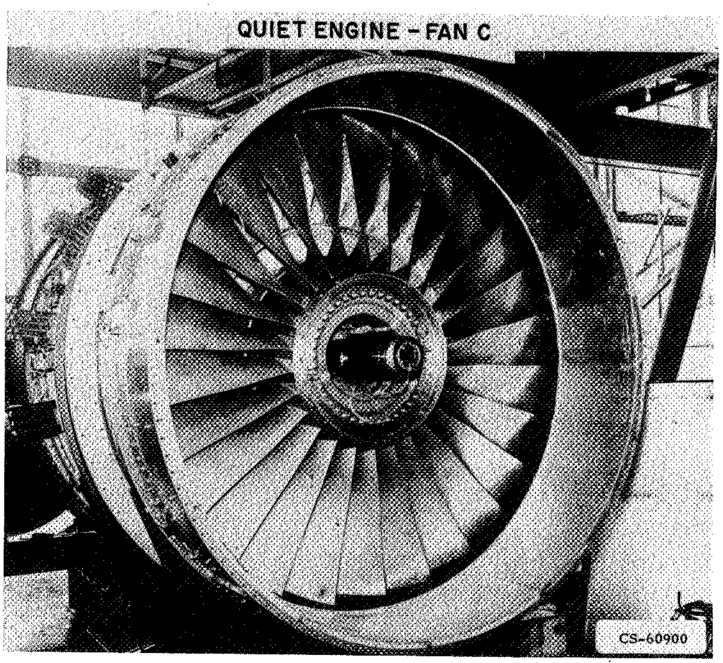


Figure 7



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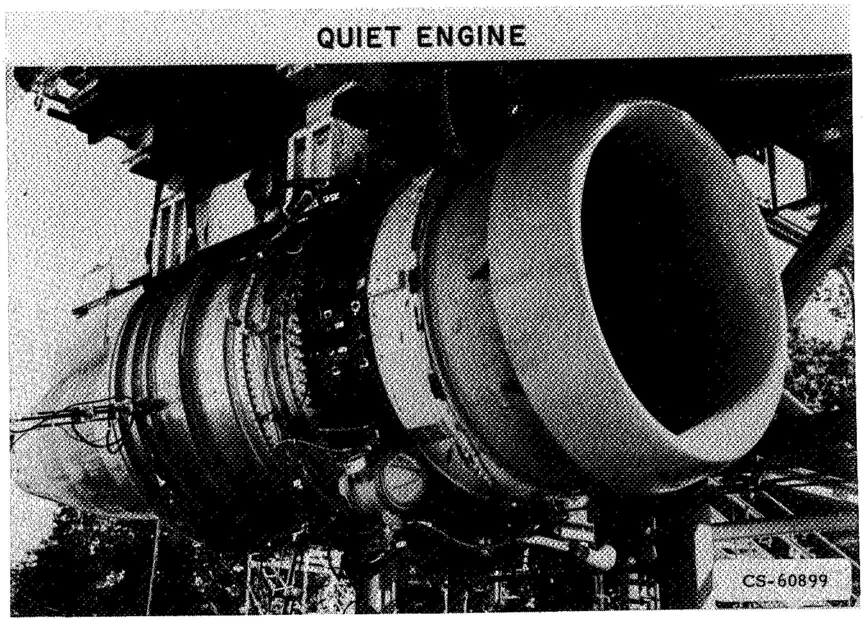

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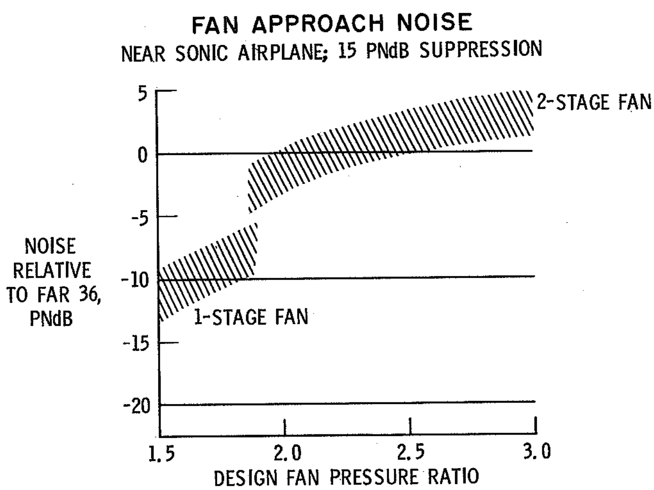

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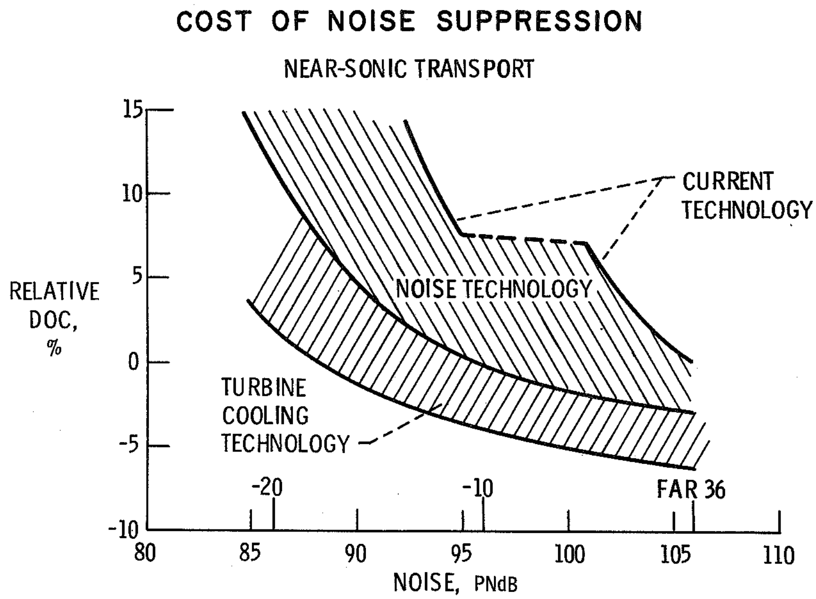

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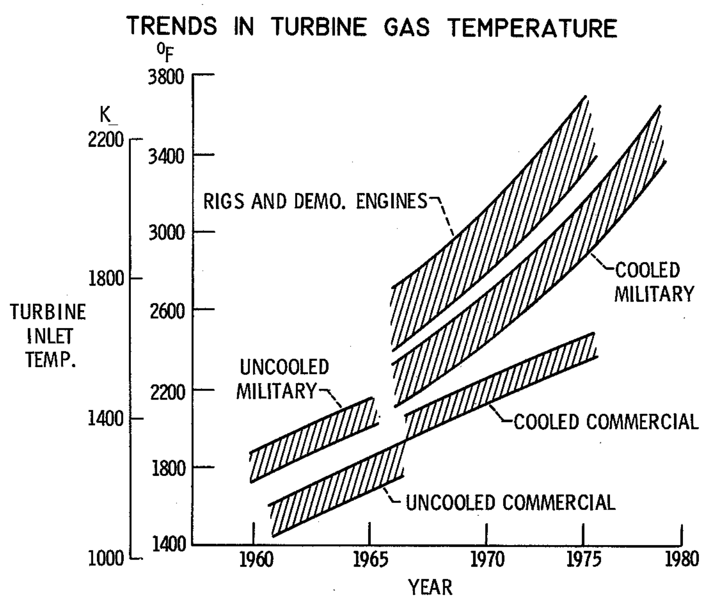


Figure 12

POTENTIALS OF COOLING METHODS

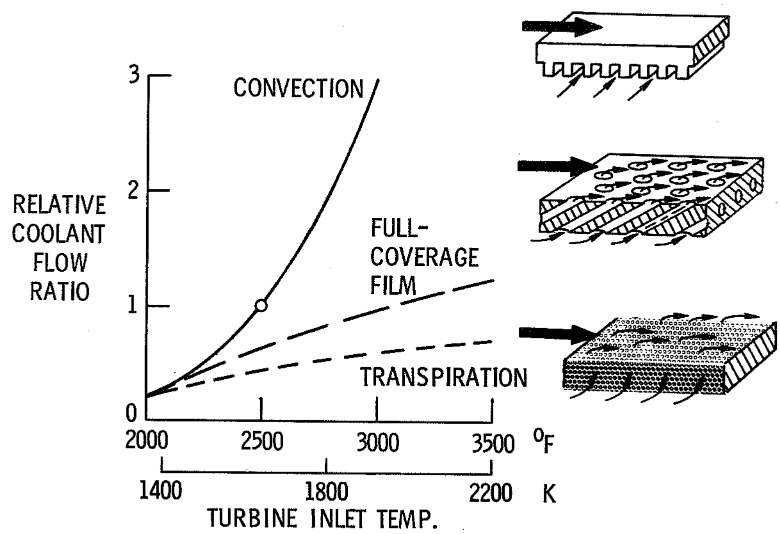


Figure 13

ADVANCED COOLING CONCEPTS

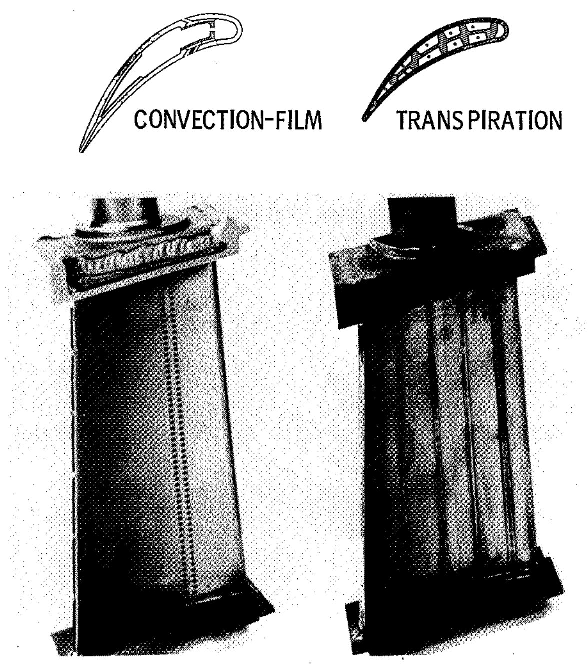
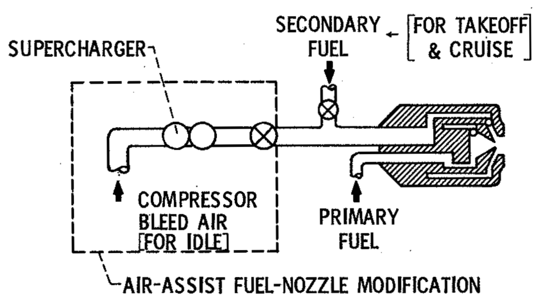


Figure 14

Figure 15

CONTROL OF EMISSIONS BY FUEL-NOZZLE DESIGN



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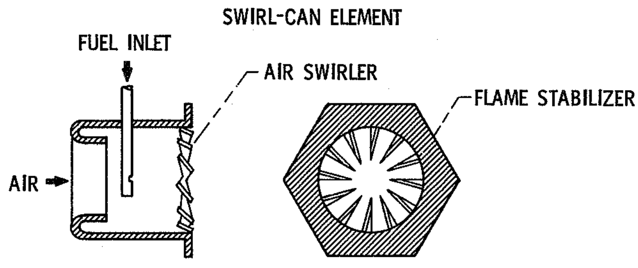

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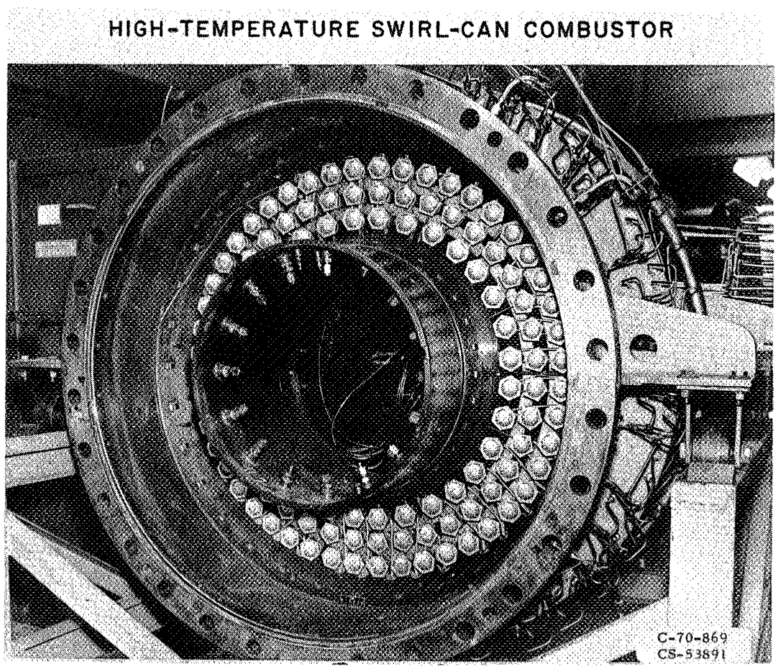


Figure 18

CONTROL OF EMISSIONS BY COMBUSTOR DESIGN

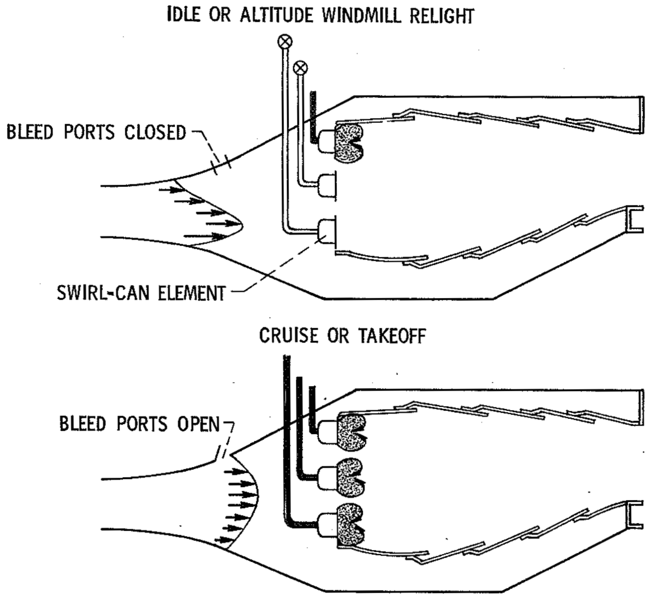


Figure 19

STOL PROPULSION SYSTEMS

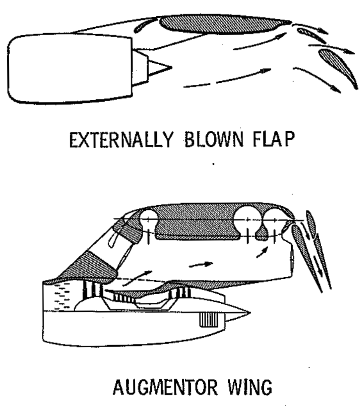


Figure 20

NOISE TEST HARDWARE FOR EXTERNALLY BLOWN FLAP

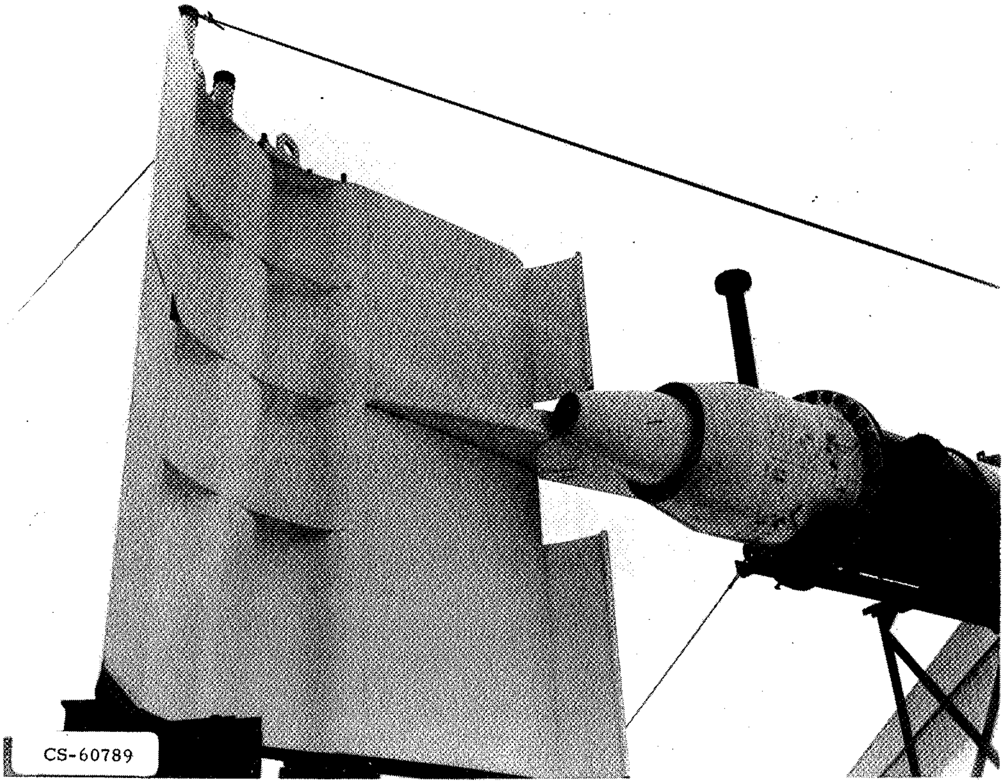


Figure 21

FLAP INTERACTION NOISE

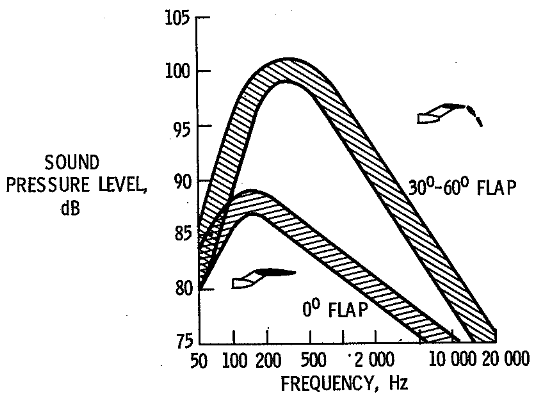


Figure 22

EXTERNALLY BLOWN FLAP WITH MIXER NOZZLE

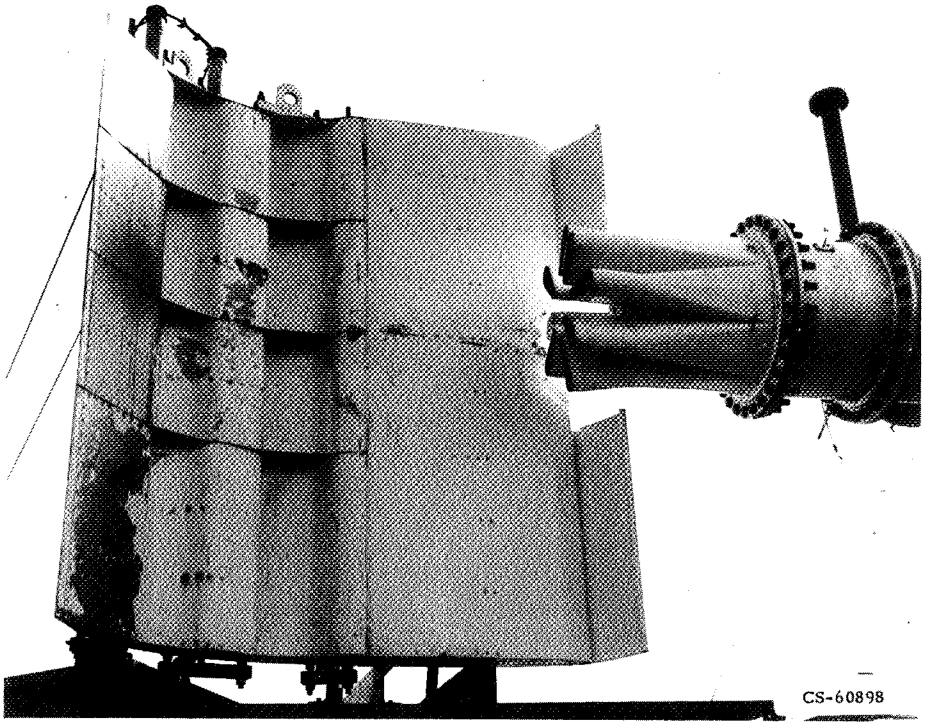


Figure 23

PERFORMANCE OF SONIC INLET

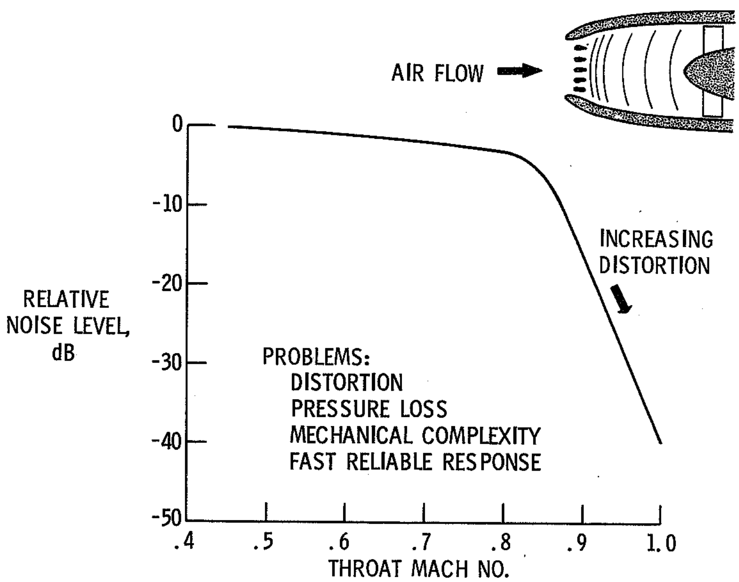


Figure 24

AUGMENTOR-WING NOISE TEST FACILITY

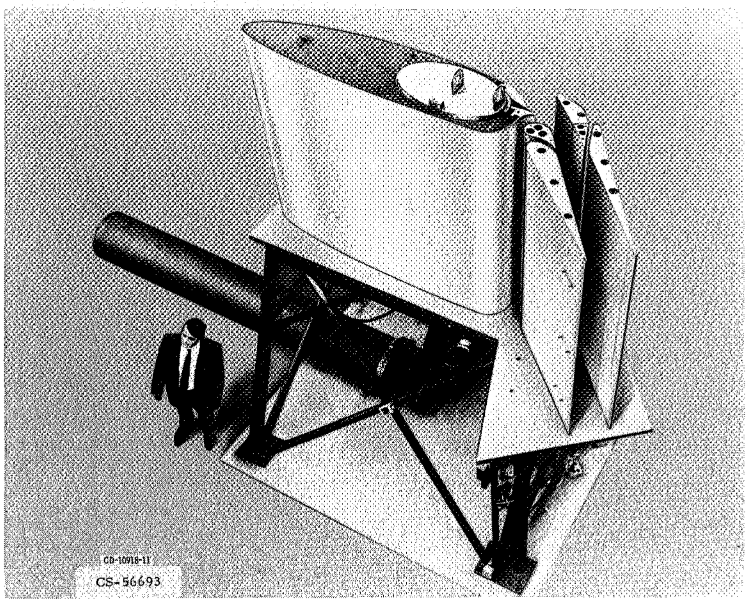


Figure 25

AUGMENTOR-WING NOISE

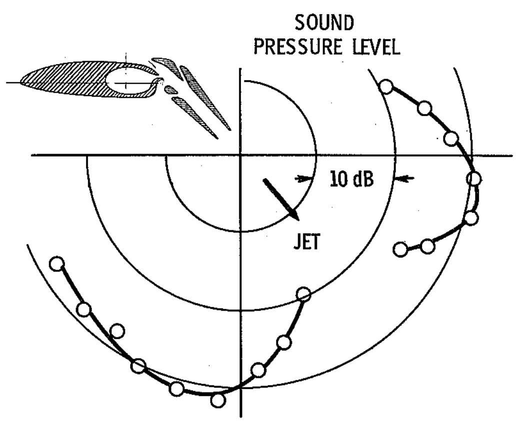


Figure 26

STOL NOISE ESTIMATES

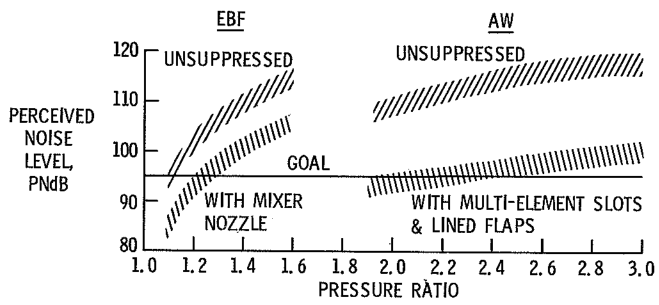


Figure 27

VTOL PROPULSION FUNCTIONS

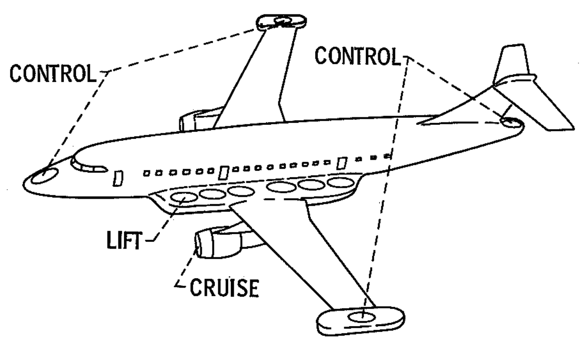


Figure 28

INTEGRAL-DRIVE LIFT FAN

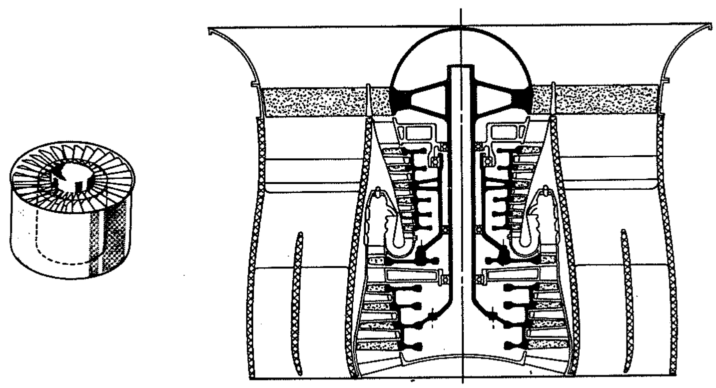


Figure 29

REMOTE-DRIVE LIFT FAN

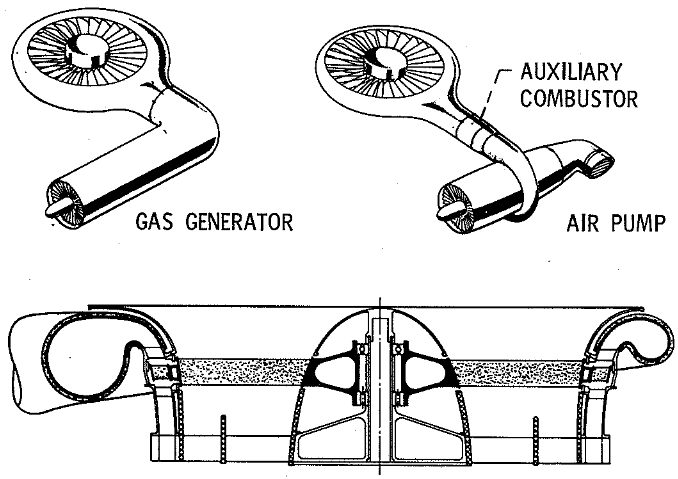


Figure 30

RANGE VERSUS NOISE FOR SST

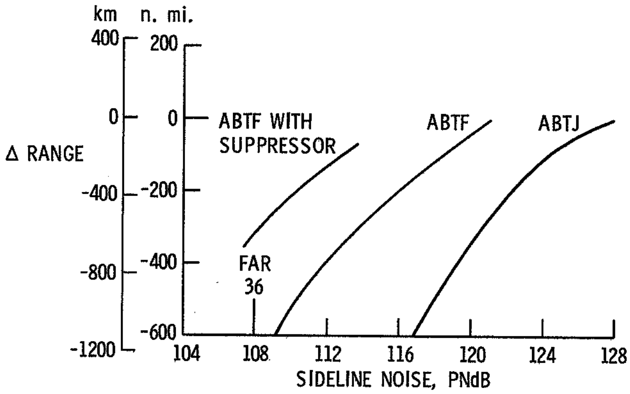


Figure 31

INLET DESIGN REQUIREMENTS

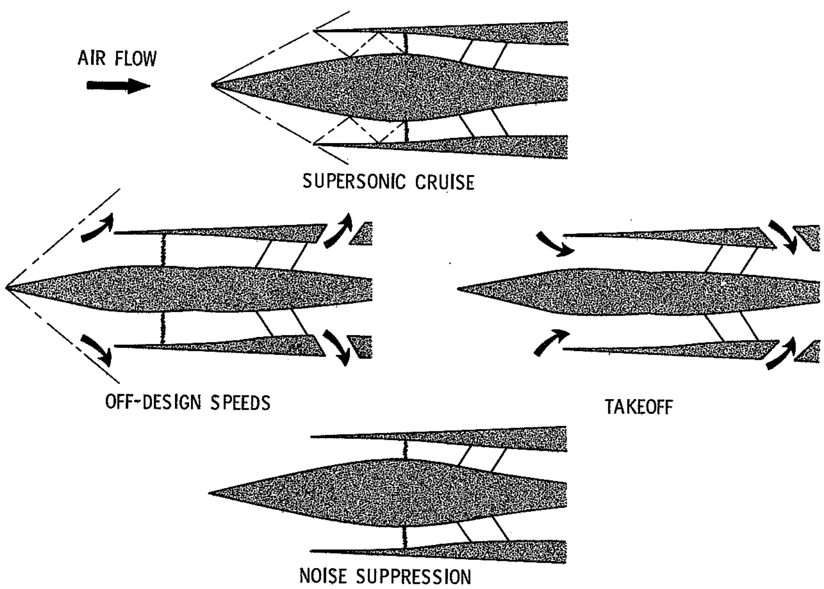


Figure 32

INLET EFFECTS ON RANGE

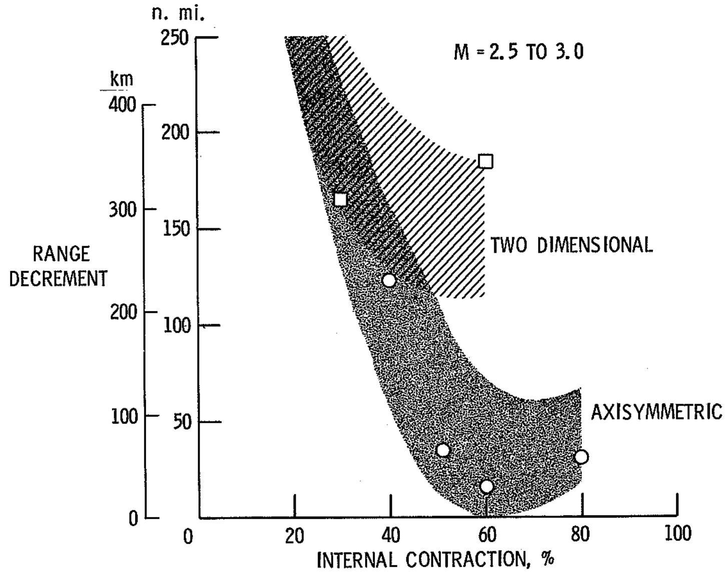


Figure 33

INLET STABILIZATION

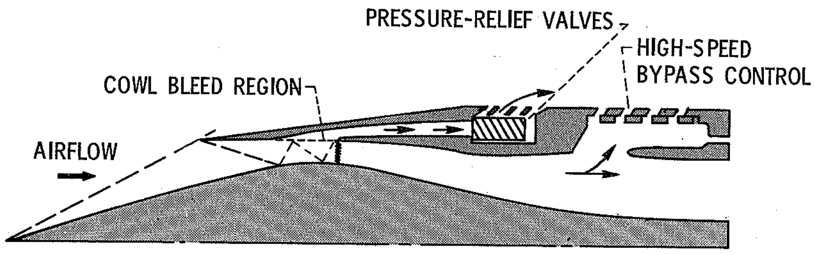


Figure 34

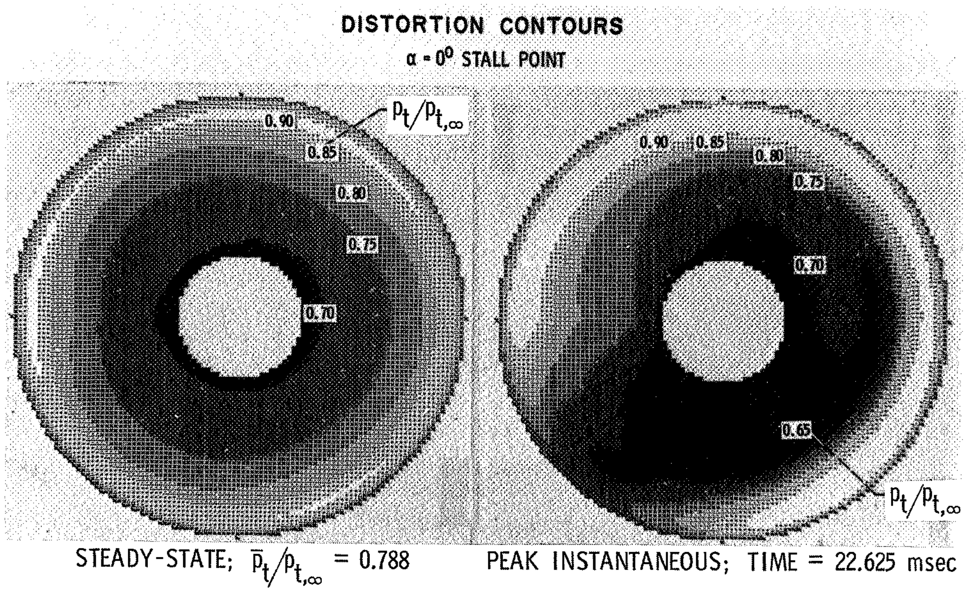


Figure 35

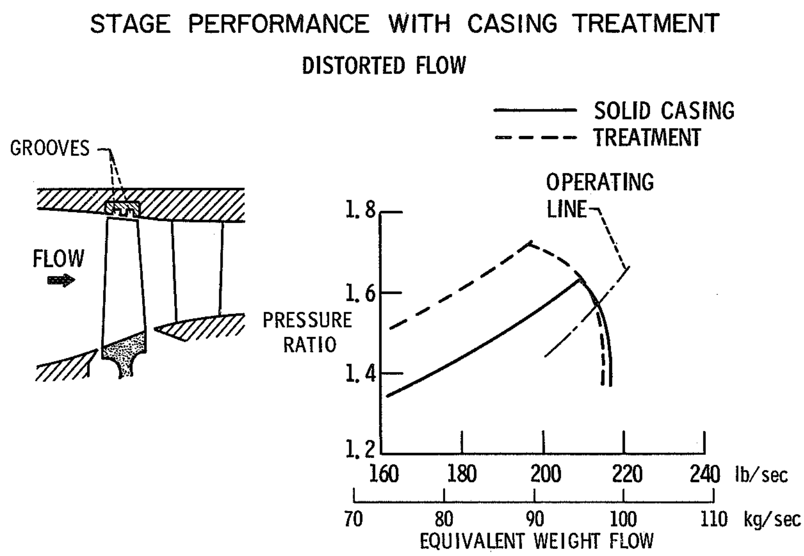


Figure 36

COMPRESSOR CASING GROOVED INSERT

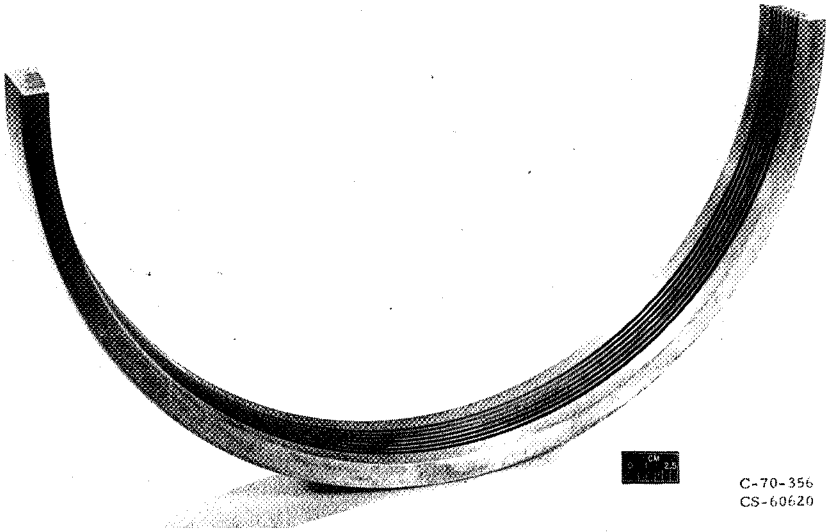


Figure 37

NOZZLE DESIGN REQUIREMENTS

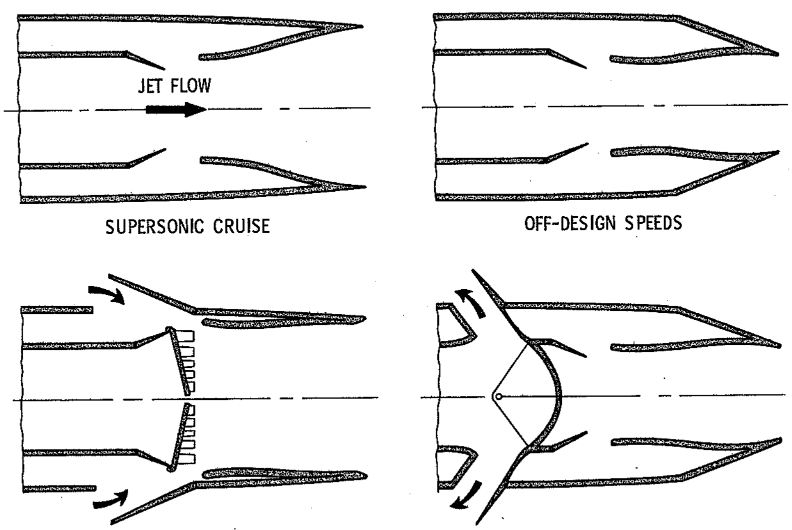
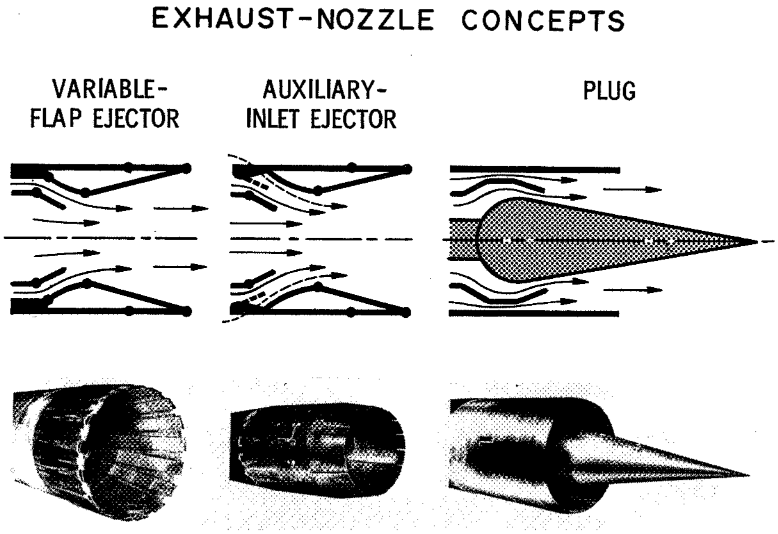


Figure 38



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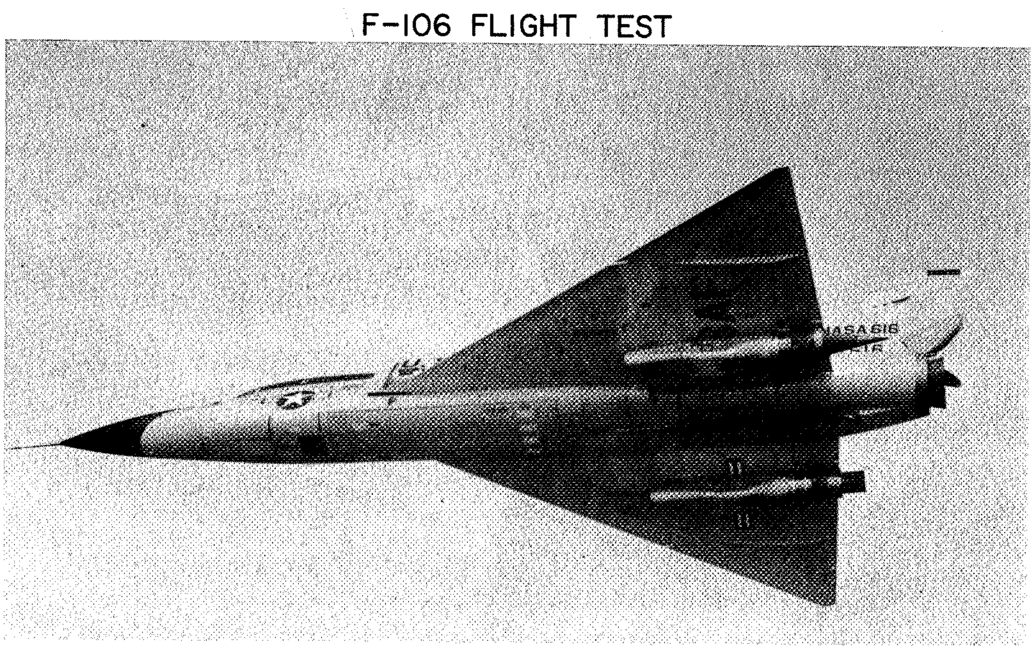

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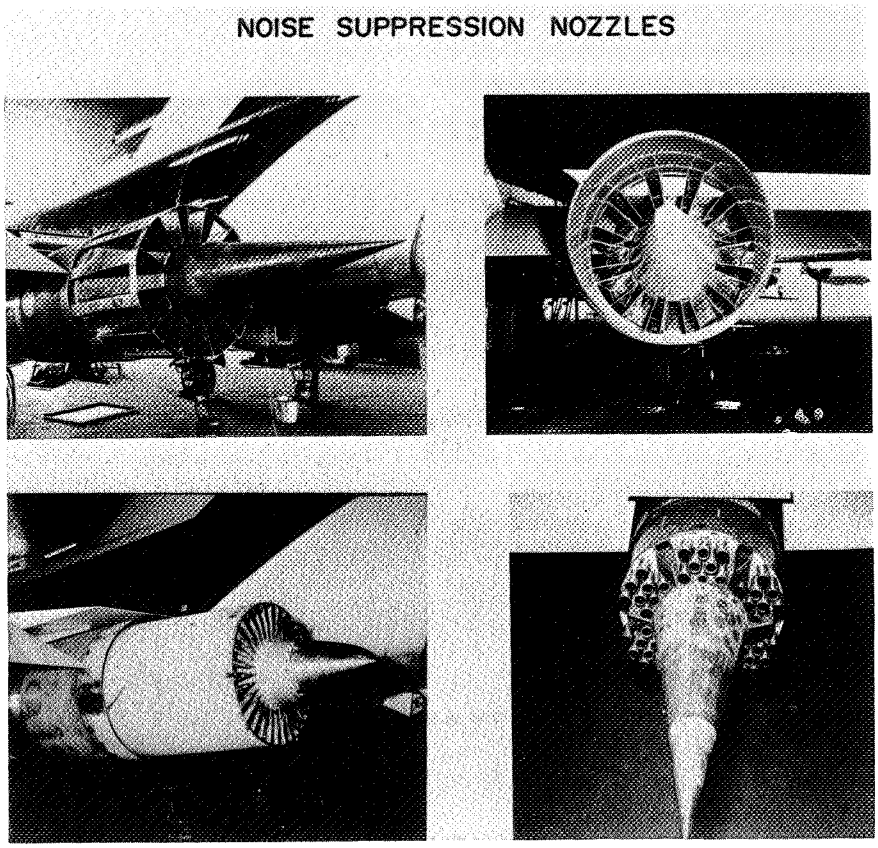

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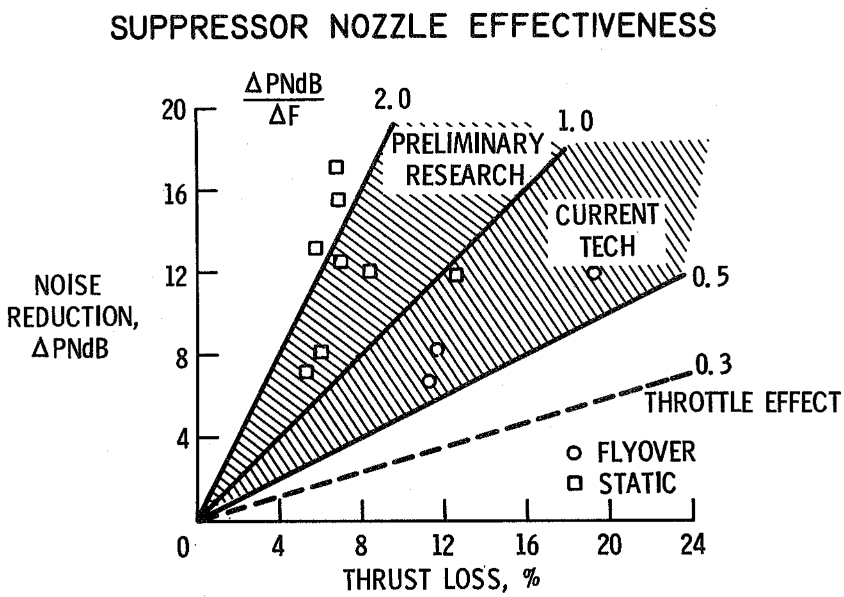

Figure 42

GEARED FAN-JET CONFIGURATION

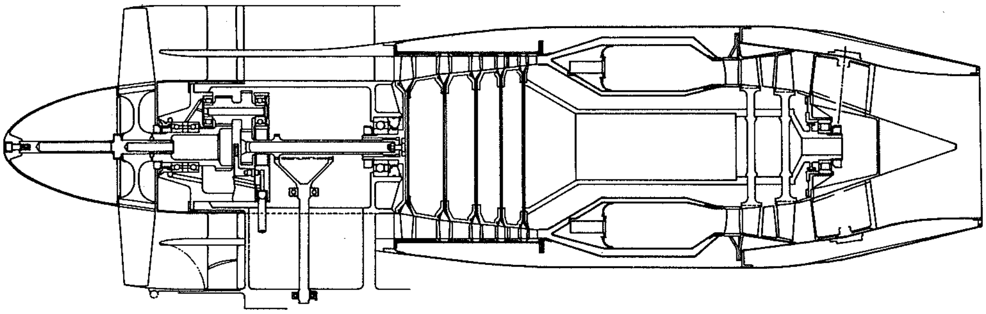

Figure 43

NAVY ORDNANCE ENGINE

2890-N (650-lb) SEA-LEVEL STATIC THRUST

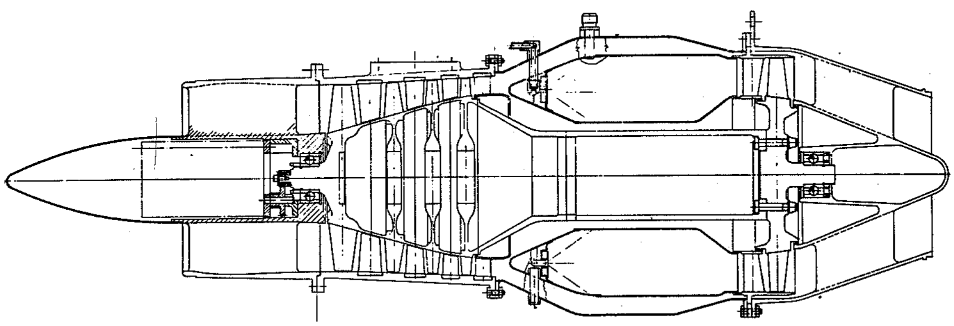

Figure 44

CAST COMPRESSOR AND TURBINE ROTOR

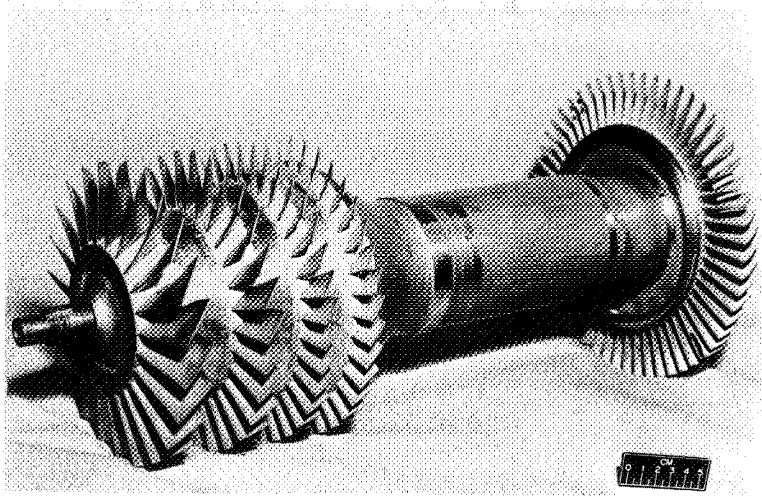

Figure 45

SHEET-METAL COMPRESSOR
ROTOR COMPONENTS

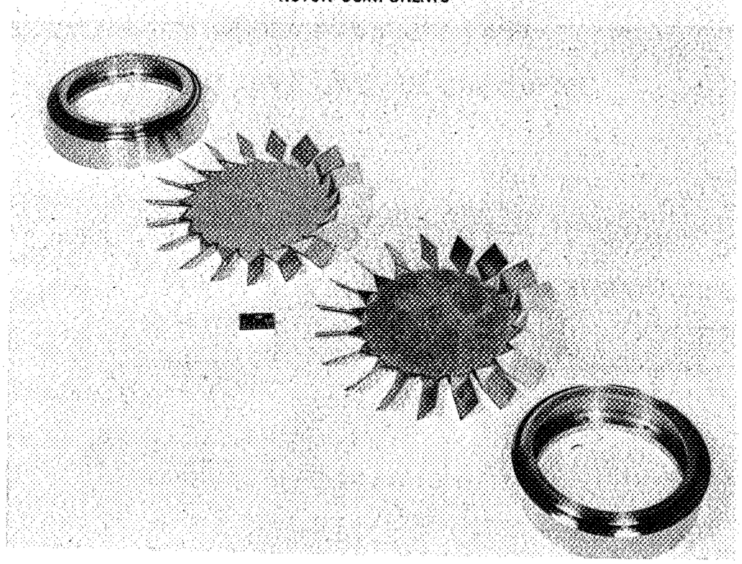

Figure 46

ASSEMBLED SHEET-METAL COMPRESSOR ROTOR

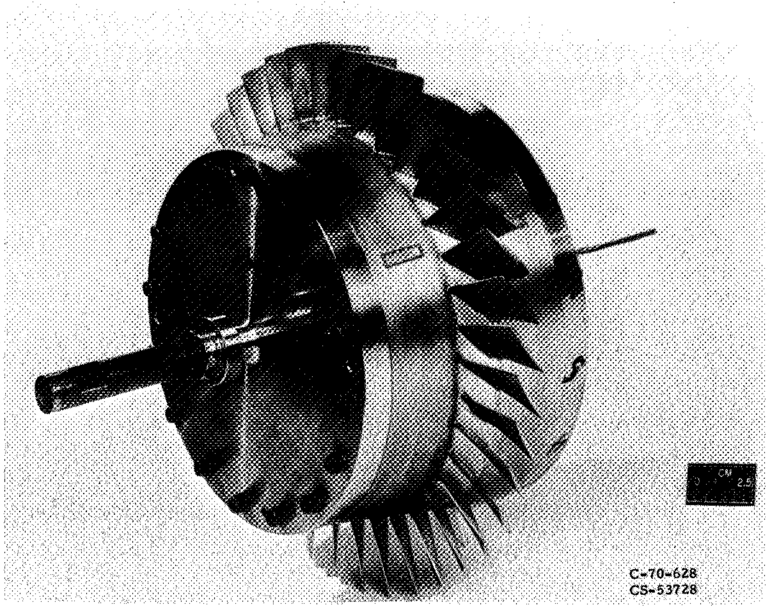

Figure 47

Figure 48

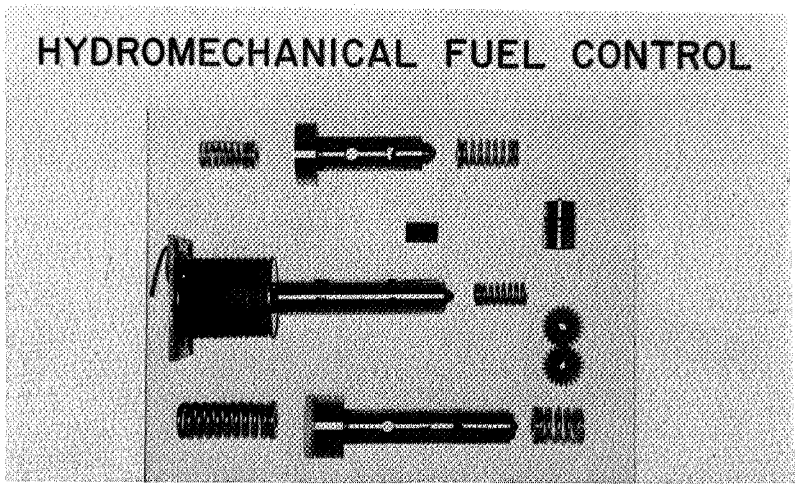

Figure 49

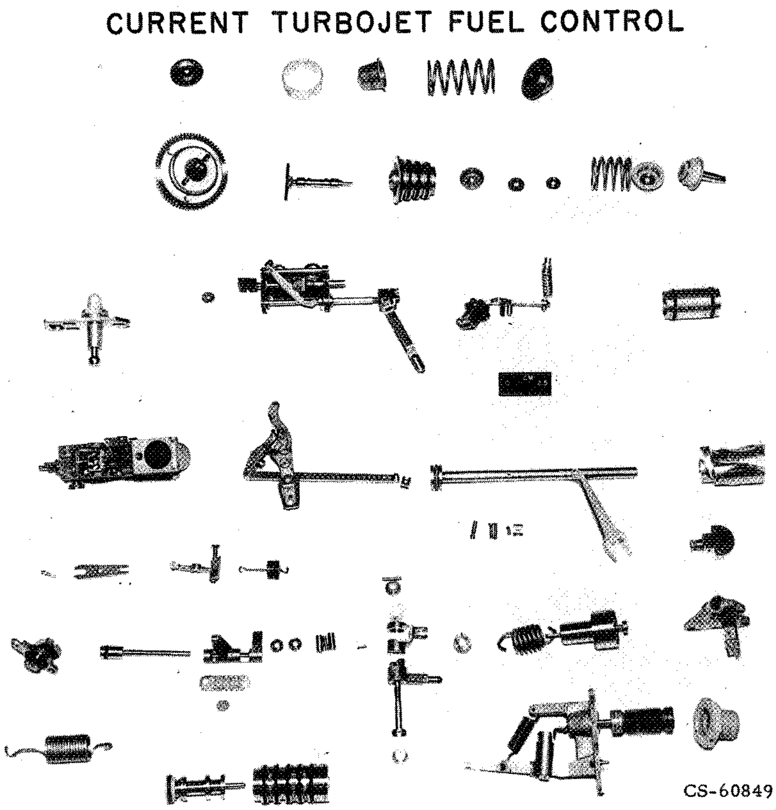

Figure 50

HYPERSONIC AIR-BREATHING PROPULSION SYSTEMS

By John R. Henry and H. Lee Beach
Langley Research Center

INTRODUCTION

During the past decade exploratory research on concepts for hypersonic air-breathing engines has been pursued in substantial research and development programs in the United States, and a considerable technology base has been established. Summaries of this work, together with the present status and estimates of future prospects, are given in a number of papers, for example, references 1 to 5. Several of these engine concepts have been shown to be feasible, relative to the aerothermodynamic performance of the engine cycle, by investigations of small-scale research engines and components. Little effort, however, has been devoted to the development of concepts and hardware for hypersonic engines and propulsion systems which will integrate with an airframe and which will have satisfactory flight operating characteristics and performance for a given mission. Furthermore, some of the engine types investigated are more applicable to space and military missions than to civil cruise missions. The National Aeronautics and Space Administration has initiated a research program designed to satisfy some of these needs, and the principal approaches to this work are outlined in reference 6.

The present paper presents an overview of the various possibilities for air-breathing propulsion systems for cruise applications and gives an assessment of their present status and their future potential. Clearly, there is a need for exploratory research and imaginative solutions in many areas in order to realize the apparent potential of air-breathing engine types. In this regard, several promising approaches to the design of the airframe-integrated scramjet (supersonic combustion ramjet) are described in some detail; this work has been initiated in the NASA program. Emphasis is placed on scramjet development because it has inherent qualities, such as high performance over wide ranges of flight speeds and geometric and mechanical simplicity, which make it a leading candidate for many hypersonic applications.

ENGINE TYPES FOR HYPERSONIC CRUISE

Air-breathing engines traditionally have been considered most effective for cruise in the subsonic and low supersonic speed range primarily because of their high specific impulse performance (high thrust per unit fuel weight flow), a parameter which has a first-order effect on the range of a cruise vehicle. Turbojet and turbofan engines enjoy this preference in spite of having characteristically high weights and low installed thrust-weight ratios. Installed propulsion system weight obviously is important since a saving in weight can be traded for increased fuel load and range, provided the sacrifice in specific impulse performance is not too great. A large number of other factors are important to various degrees, such as operating characteristics, system and mechanical simplicity, and structural and cooling requirements.

For cruise at high supersonic and hypersonic speeds, additional factors must be considered. There is an upper flight-speed limit on turbine-type engines; this limit generally is considered to be about Mach 3. (See ref. 1.) For hypersonic cruise the designer is faced with the problem of choosing a propulsion system which will operate satisfactorily over the entire speed range; such a system might consist of two separate engine types, or a combined engine which packages two types in one hardware assembly, or a composite engine which integrates several engine cycles in such a way that two or more can operate simultaneously on the same propulsive fluid. The ultimate choice will depend on future progress in these areas and a preferred weighting of the several performance, operating, and design factors previously noted. However, in order to choose a path along which to plan future advances, current knowledge must be used to assess the status and potential of the more promising types.

Cruise Mission in Mach 3 to Mach 5 Range

A propulsion system consisting of the turbojet (operational capability to Mach 3) and the ramjet (operation above Mach 1) provides the highest performance potential for cruise vehicles operating at high supersonic to low hypersonic Mach numbers. This high performance characteristic is exemplified in recent studies at NASA in which vehicles with hydrogen-fueled turbojet-ramjet propulsion systems were predicted to achieve ranges on the order of 8000 to 9600 kilometers (5000 to 6000 miles) under realistic assumptions of weights and aerodynamics. (See paper no. 25 of this compilation.)

Technology requirements for designing either the turbojet or the ramjet are well established. Turbojet development is, in fact, very advanced, and extended application to a hypersonic vehicle should require only a straightforward effort. Ramjet technology, although not at the level of turbojet technology, is sound enough to allow ramjet engines using hydrocarbon fuel to have been flown to Mach 4. (See ref. 1.) In addition, a hydrogen-burning ramjet employing regeneratively cooled chamber walls consisting of reinforced formed tubes has been developed and has been extensively tested. (See ref. 2.)

The question of whether the turbojet and ramjet should be installed in the combined wraparound turboramjet fashion or in a separately operated engine configuration depends on the detailed system analysis for a particular vehicle to evaluate relative weights, controls, and engine drags. It also depends on the continued progress of turboramjet development which is discussed in reference 1. A recent investigation of an all-body hypersonic cruise aircraft with two-dimensional inlets touched on this subject (ref. 7) and revealed that the wraparound turboramjet was severely penalized, relative to separate engines, by higher weights and engine dimensions.

Ultimately, the biggest task facing designers of a Mach 3 to Mach 5 cruise transport is the definition of efficient configurations for propulsion-system installation on the vehicle. This definition involves the optimization of aerodynamics, propulsion, and structures and is essentially a development task. A turbojet and ramjet propulsion system on a Mach 3 to Mach 5 cruise vehicle is therefore judged to be a near-term possibility relative to other types that require significant research effort.

Cruise Mission in Mach 6 to Mach 12 Range

At Mach 6 and above, the containment of flow and cooling requirements associated with high pressures and temperatures are problems for subsonic combustion ramjets, and dissociation losses cause ramjet performance to drop off rapidly. For flight at these high Mach numbers, the substitution of a supersonic combustion ramjet (scramjet) for a subsonic combustion ramjet gives higher performance. The scramjet can also be operated in the subsonic combustion mode and provides the capability for high performance in the Mach 3 to Mach 5 range as well.

Over the last decade a number of small-scale scramjet research engines have been developed; these engines have been described in references 1 and 2. Practical levels of scramjet performance closely approaching the values predicted on the basis of isolated high-efficiency component data have been demonstrated in these actual test engines. The feasibility of several engine conceptual design approaches likewise has been substantiated, including the dual-combustion mode and thermal-compression engines. For the former, stable and controllable conversion from subsonic to supersonic combustion and vice versa during continuous engine operation has been demonstrated.

Off-design combustor operation in the research engines utilizing a staged fuel-injection technique to control the heat-release rate along the combustor length has been shown to be efficient. In addition, successful ignition techniques have been substantiated at Mach numbers corresponding to temperatures too low for autoignition. Solutions for component iteraction problems, such as combustor operation downgrading inlet performance and causing inlet unstart, or other operational difficulties have also been developed.

For the NASA hypersonic research engine, flight-weight fuel and control systems have been developed and tested in a "breadboard" configuration to the point of substantiation, and a flight-weight fuel-cooled, structural model engine has been designed, fabricated, and tested. (See ref. 8.)

The scramjet research engine projects have not been concerned with the development of designs and configurations which would integrate with the vehicle to provide satisfactory flight operating characteristics. Such a design would require a low amount of cooling, high performance with fixed geometry (if possible), and good aerodynamic characteristics. Approaches to the development of integrated scramjet concepts will be discussed in detail in later sections.

Composite Engines for Use in Either Mission

Candidate propulsion systems for cruise in the Mach 3 to Mach 12 range include not only pure air-breathing systems but also composite rocket—air-breathers as well. (See refs. 1 and 9 to 11.) The attractive feature of the composite is its potential to provide a favorable trade-off between the high-efficiency (in terms of specific fuel consumption) air-breathers and the high trust-weight ratios of rockets. The mixed-cycle capability of these engines is utilized from takeoff to about Mach 2.5 to Mach 3 or during the acceleration phase.

An engine concept which seems feasible for cruise is the supercharged ejector ramjet (SERJ) described in references 1 and 9 and shown schematically in figure 1. It can be described in simple terms as a ramjet with rockets and fan enclosed and operates in a fan-assisted (supercharged) air-augmented rocket mode for takeoff, in a ramjet mode for cruise, and in a fan mode for landing.

Component technology for the air-breathers discussed previously and for rockets (ref. 12) indicates good feasibility for the SERJ; subscale testing on an ejector ramjet (ref. 11) substantiates this viewpoint. The fan is necessary because most near-term missions require a significant subsonic cruise and loiter capability. At high speed its effective disposition from the airstream becomes a problem; a satisfactory solution has been found for lower Mach numbers (ref. 13), but further investigation, concentrating perhaps on physical removal of the fan, is required for higher Mach numbers. Possible help in this regard is the speculation of reduced cruise and loiter requirements in the vicinity of airfields for hypersonic cruise vehicles. (See ref. 1.)

Another concept with applicability to the acceleration phase in rocket—air-breathing composites involves the inclusion of a liquid air cycle to provide the oxidizer for the ejector mode. The possibility of efficiently pumping liquid air to high pressure could lead to a reduction of turbomachinery weight and high thrust-weight ratios (ref. 1) if the heat-exchanger weight can be kept down.

Although rocket—air-breathing composites have the potential for providing payload (range) capability significantly greater than those of either rockets or air-breathers, actual predictions of near-term performance potential (ref. 9) do not verify this capability.

Indications are that specific impulse levels are of the magnitude anticipated, but large engine weights yield lower thrust-weight ratios than desired. Since, however, most studies of these composites have been focused on launch-vehicle applications, they may not have been shown to their best advantage for cruise. Basic research and development will be required to realize the indicated potential for cruise application; these studies should include the exploration of new schemes for improving present concepts, with particular emphasis on the fan (SERJ), atmospheric fouling, and effective catalysts for the para- to ortho-hydrogen conversion. (See ref. 9.)

Special Future Types

In addition to the hypersonic engine types and configurations already considered, several other concepts warrant consideration in the future. One of these, the lift fan, has been proposed as a means of providing for takeoff and landing and thereby removing the requirement for large lifting surfaces on hypersonic vehicles. This would allow cruise at hypersonic speeds at lower altitudes (higher dynamic pressure), which would favor more efficient engine operation.

Another prospect for the future is the external combustion scramjet. In such an engine, the combustion process is not contained but takes place at a specified location on the underside of an imaginatively designed vehicle. Very favorable aerodynamics could conceivably result from the proper coupling of the vehicle configuration and the pressure rise due to combustion. Cooling requirements would also be enhanced since a larger part of the engine surfaces could be radiation cooled.

Recently, nuclear rockets, with air-breathing acceleration to a safe altitude for reactor firing, have been proposed as propulsion-system candidates for hypersonic transports. They would operate as boost-glide vehicles with flight times associated with Mach 25 cruise (ref. 14). Current feeling seems to be that the gaseous core nuclear rocket (GNR) is the best choice for achieving the necessary high specific impulse at a reasonable thrust-weight ratio. The feasibility of a nuclear hypersonic propulsion system has yet to be successfully demonstrated, and serious questions about safety arise (particularly, for flights over land); the concept, nevertheless, deserves further investigation.

It is apparent that much work remains to be done before a finalized propulsion-system design for even a first-generation hypersonic cruise vehicle can be completed. At the present time, turbojet-ramjet and turbojet-scramjet systems enjoy an advantage both from a performance and technology standpoint, but the door of the future is wide open for new and innovative propulsion ideas such as those described. Regardless of the engine components, however, it has become clear that a hypersonic propulsion-system design cannot be accomplished independent of vehicle design.

KEY ISSUES IN AIRFRAME—INTEGRATED-SCRAMJET DESIGN

The design of practical scramjet engine configurations for application to prototype hypersonic vehicles and missions requires advances in technology in three principal areas. First, internal design features have to be developed which will conform to the restraints imposed by integration with the airframe and practical flight operating characteristics. Secondly, concepts for the design of fuel-cooled structures systems are required which will extend engine life to thousands of cycles, in contrast to the 136 cycles demonstrated by the NASA hypersonic research engine. This problem is discussed in paper no. 9 of this compilation and will not be treated herein. Thirdly, engine configurations are desired which have cooling requirements significantly less than the heat sink available in the hydrogen fuel flow needed for thrust. This engine cooling characteristic will provide the hypersonic-vehicle designer with several notable options, as discussed in reference 6, which result from the potential for cooling parts of the vehicle by using the excess heat sink in the fuel. The first and third items are discussed in this paper; however, an understanding of the general implications of airframe integration on the engine design is needed first.

GENERAL INTEGRATED DESIGN FEATURES

Current vehicle concepts under study feature a propulsion installation consisting of several engine modules mounted flush to the underside and toward the aft end of the vehicle. (See paper no. 5 of this compilation.) With this arrangement the shock wave produced by the bow of the vehicle compresses the engine airflow upstream of the inlet and thus reduces the compression required by the inlet and reduces the engine size needed for a given amount of thrust by a factor of about 3 at Mach 10 and a factor of 1.5 at Mach 3. The space available for the propulsion system located between the vehicle undersurface and the bow shock is several times wider than it is high; this fact suggests an arrangement of several rectangular engine modules side by side. An unfavorable aspect of the design is the relatively thick turbulent boundary layer generated on the forebody of the vehicle, which must either be diverted around or ingested by the engines. Current preliminary studies indicate that the scramjet engines can ingest the boundary layer without significant detrimental effects. Analyses show that the transfer of part of the engine inlet function to the vehicle forebody results, typically, in a contribution by the forebody to the thrust of roughly 35 percent, which emphasizes the requirement for an efficient design.

The other major geometric design feature of the integrated configuration which is external to the engine modules is the vehicle afterbody, which is used as an extension to the engine nozzle as noted in paper no. 5 of this compilation. This feature permits much higher effective engine exhaust velocities with relatively modest area expansions in the nozzle hardware itself and with low engine external drag. Effective designs for the afterbody will provide contributions to the thrust varying from roughly 25 percent to 35 percent over the Mach number range and thus make the afterbody design equal in importance to the forebody.

Basically, the engine-airframe integrated design features just described involve the entire underside of the vehicle in the production of propulsive thrust. The driving force leading to these design concepts is that at hypersonic flight speeds in spite of the high energy potential of the hydrogen fuel, very large engine airflows are required to produce sufficient thrust. If integrated designs were not used and the engine were completely self contained and pod mounted in the free stream in the more conventional manner, the size, weight, and drag of the engines would be too high to be practical. On the other hand, there are promising approaches to the design of the integrated engine modules as discussed in the next section.

ENGINE MODULE DESIGN

The general arrangement for an engine module of the type described in paper no. 5 of this compilation is shown schematically in figure 2. The sketch applies to flight up to Mach 3. The turbojet and its ducting are shown embedded in the body of the vehicle with a fixed-geometry scramjet mounted underneath the turbojet installation. A variable-geometry inlet and an adjustable door would be used to match the inlet airflow to the turbojet requirements. The adjustable door also would be used to close off the turbojet ducting above a flight Mach number of 3.0. The scramjet engine is not provided with a closeoff door; the studies of reference 6 indicate that it is not a requirement. In present concepts air would flow through the scramjet engine at all flight speeds, the scramjet would provide all propulsion above Mach 3, and combustion would be initiated in the scramjet engine in a subsonic mode in the transonic flight range to reduce afterbody drag and would be continued at low supersonic speeds to contribute thrust. The use of the scramjet engine in this speed range would require an ignitor in addition to an appropriate combustor and fuel-injector design.

A similar schematic is shown in figure 3 for scramjet operation above Mach 3. The fixed-geometry scramjet concept illustrated is the main object of a research and development program at the Langley Research Center, and it will be used to illustrate design techniques for airframe-integrated engines. The sidewalls of the inlet are the main compression surfaces and are provided with sweptback leading edges. This feature provides an open window in front of the leading edge of the cowling, which is the bottom surface, through which flow can spill downward during the inlet-starting process at low supersonic Mach numbers; inlet starting consists of establishing supersonic flow in the inlet by a transient aerodynamic process requiring large amounts of spillage. With sweptback leading edges, planes of constant flow properties throughout the engine tend to be parallel to the leading edges, which is the reason for designing the fuel injection struts with the same amount of sweep. The combustor is terminated at a downstream station represented by a swept line parallel to the inlet leading edges. For inner modules the engine also would be terminated at this plane; however, for outboard modules the sidewalls would be continued to the nozzle exit in order to avoid spillage laterally. Changes in flow area throughout the engine occur in the lateral or spanwise direction to accommodate the requirements of the thermodynamic cycle.

SCRAMJET INTERNAL DESIGN

Inlet

The purpose of the engine inlet is to reduce the velocity of the entering airstream and thereby convert the dynamic pressure to static pressure for efficient operation of the thermodynamic cycle. The design parameter that provides an index to the amount of velocity reduction is the inlet aerodynamic contraction ratio. This ratio is defined as the ratio of the engine airflow area in front of the inlet to the flow area at the inlet throat, which in this case is located in a swept plane at the fuel injector strut station. For any scramjet design the optimum value of inlet contraction ratio has to be determined according to thrust requirements and other design criteria. (See ref. 15.) A well-known general guideline in this area is too much contraction ratio produces such high static temperatures at the entrance of the combustor that excessive dissociation of the molecular species occurs in the combustion process. The dissociation absorbs significant amounts of energy which then cannot be converted to thrust. This phenomenon partly is responsible for the fact that supersonic combustion is preferred to subsonic combustion at flight speeds above approximately Mach 6. Optimum values of contraction ratio tend to increase with flight speed; normally for fixed-geometry inlet designs this type of variation is automatically produced by a characteristic of decreasing amounts of flow spillage from the inlet with increasing flight speed. However, there is a limit to the amount of contraction ratio that is possible for a given fixed-geometry inlet design and still have a starting capability at the low end of the Mach number range; therefore, the need for variable geometry in the inlet throat, which would lift this restriction, should be examined.

Engine cycle performance analyses have been conducted for a full-scale engine and a flight trajectory corresponding to a free-stream dynamic pressure of 47.9 kN/m^2 (1000 psf). Typical values of engine component efficiencies and inlet flow spillage were assumed together with a vehicle bow shock corresponding to 8° of turning and a stoichiometric fuel-air ratio. The results are given in figure 4 in terms of specific impulse as a function of flight Mach number. Specific impulse I_{sp} is defined as thrust per unit fuel flow and is one of the prime parameters which determine range for a cruise vehicle. With fixed inlet geometry, contraction ratios ranging from 6 to 10 were assumed to be possible, depending on the inlet design. With variable geometry a high contraction ratio value of 25 was assumed for the Mach range between 8 and 10. As the flight speed is reduced below Mach 8 along the dashed curve, the contraction ratio is also lowered below 25 to avoid choking the inlet throat. The results indicate that the effect of variable geometry would be to increase the performance by a maximum of only 15 percent. The associated penalty would be increased system complexity, seal and joint problems, and increased cooling requirements. All these penalties would involve increases in weight which would tend to cancel the performance increase. In view of these results, variable geometry does not appear to be justified.

Fuel Injector and Combustor

The scramjet configuration illustrated in figure 3 features several struts which contain fuel injectors mounted in the stream. The purpose of this type of design is to reduce the required length of the combustor in order to produce configurations with low wetted area, low cooling requirements, and low weight. This principle is illustrated further in figure 5, which represents a horizontal cross section through the combustor. Rows of circular fuel injectors are located on each side of the three diamond-shaped struts as well as on each of the two sidewalls. In a supersonic combustor the fuel is injected into the stream with as little disturbance to the airflow as possible in order to keep the total-pressure losses at a low level. As a result, the fuel-air mixing occurs by a turbulent mixing or diffusion process as the flow proceeds along the combustor length. If the ignition and reaction lengths are not significant, the combustor length required for efficient performance is equal to the mixing length required for all the injected fuel to reach a stoichiometric mixture. For the two-dimensional case illustrated, this length is on the order of 20 times the width of the airstream tube carbureted by opposed fuel-injector rows or 20 times the gap S between the struts. The use of fuel-injector struts is the primary difference from most designs for the small-scale research engines previously mentioned. In the research engines, the fuel injection generally was entirely from the walls and thus would require a combustor length four times that shown in figure 5. The combustor lengths could be shortened still further by using more than three struts, but the reduction in wetted area reaches the point of diminishing returns. The use of fuel-injection struts has been demonstrated to be practical in an experimental investigation of a large-scale supersonic combustor by Metzler and Mertz (ref. 16).

Optimization of Fuel Injector and Combustor

The use of fuel injector struts provides a potential for short combustor designs; however, the injector configuration and arrangement still has to be optimized for the proposed range of operating conditions and integrated with the combustor design. Analytical design techniques are required to accomplish these tasks satisfactorily. Considerable technology has been developed in the Langley program in basic fluid-mechanic type of investigations of fuel injection and mixing with and without reaction. (See refs. 17 to 35.) Some of these results (ref. 34) have been correlated by use of an integral parameter, the mixing efficiency, which is defined for some downstream station in a duct as the ratio of the amount of fuel flow that would burn if ignited to the total amount of fuel flow injected. (See ref. 35.) The mixing efficiency, therefore, is directly comparable to combustion efficiency. In reference 34, a simple empirical correlation is established for mixing efficiency in terms of injector geometry, the distance downstream, and the dynamic pressures of the injected fuel and air streams. Comparisons are made in figure 6 between predictions of mixing efficiency (fraction of fuel mixed), based on the empirical correlation, and measured values of combustion efficiency (fraction of fuel burned) for a variety of supersonic combustor designs tested with full duplication of flight conditions at the entrance to the combustors (ref. 35). The two efficiencies agree within a maximum deviation of ±0.1; this degree of scatter is to be expected since the mixing efficiency correlation ignores a number of secondary effects on mixing, such as pressure gradient, effect of reaction, combustor design details, and so forth. This analytical technique is useful in taking a first cut at optimizing designs. Sophisticated theories are also under development which will provide higher accuracy and more flexibility; these theories are described in the next section.

Thermal-Compression Concept

The fuel-injector struts of figure 5 are all located in a plane normal to the flow direction; by arranging the struts or fuel injector bodies in a sweptback pattern, additional compression of the inlet airflow is possible by means of a thermal-compression effect illustrated in figure 7. The sketch is similar to figure 4 of reference 36 and represents the upper half of a supersonic combustor designed for a flight Mach number of 8. The inlet is a fixed-geometry inlet for which the lowest Mach number obtainable in the throat or at the combustor entrance is on the order of 3.0. It has been designed, however, to produce the nonuniform Mach number distribution indicated in order to be compatible with the concept. At each of the fuel injector struts, represented by the long triangles, combustion is initiated, and the expansion of the combustion regions maintains the compression shocks originating at each strut. These shocks provide added compression of the airflow approaching the combustion regions so that at all points P, the

Mach number is reduced to 2.5 from the values at the combustor entrance and the static pressure is increased to 2.5 atmospheres (101.3 kN/m^2 = 1 atmosphere) from an average value of 1.2 atmospheres. This added compression would provide a significant increase in the thrust performance. The design of a thermal compression combustor obviously requires sophisticated design techniques which can be used to analyze local conditions within and surrounding a combustion region. Theories of this type have been developed in the Langley Research Center program and include provisions for nonuniform flow distributions, fuel-air mixing and reaction, merging mixing patterns, and so forth. Experimental programs on supersonic combustion are in progress to evaluate and substantiate the turbulent mixing parameters in these theories. Langley Research Center also is conducting a cooperative program with New York University (by means of an NASA grant) to develop a concept for a thermal compression scramjet. Dr. Antonio Ferri, who is the principal advocate of the thermal-compression concept, is supervising this program. During the program the problems involved in designing a practical engine of this type will be identified and will furnish guidance for the more basic research.

REDUCED ENGINE COOLING REQUIREMENTS

Design Techniques

Thermal-protection systems for the scramjet engine for cruise vehicle applications will be of the regenerative type with the internal walls formed by cooling jackets through which cold fuel will be circulated to absorb the heat. The heated fuel will then be injected into the airstream and burned; the heat transferred through the engine walls is thus conserved. A system of this type is described in reference 8. One of the prime objectives of the research and development program on integrated scramjet concepts is to develop configurations which will have cooling requirements which are substantially less than the heat sink available in the fuel flow required for thrust. The two principal approaches to solving this problem are the reduction of the wetted area or wall area to be cooled and the reduction of the heat-transfer rate.

An improved method for estimating heat transfer in supersonic combustors has been developed, based on a modification and extension of the method of reference 37. The analysis features provisions for the effects of pressure gradient and nonequilibrium boundary-layer velocity distributions. The accuracy of this method has been substantiated by use of heat-transfer data measured in supersonic combustors. (See ref. 38.) This analytical technique has been used to illustrate the effectiveness of several design techniques in reducing the cooling requirements of the combustor, which is responsible for on the order of one-half the total engine cooling load. The results are presented in figure 8.

The curve at the top of the figure indicates the effect on combustor cooling requirement of using fuel injection struts to reduce combustor length, as noted in the discussion of figure 5. The use of three struts clearly will produce a large part of the reduction in cooling requirement that is possible by use of this technique.

The two plots at the bottom of the figure are associated with reducing heat transfer by reducing the pressure gradients and pressures in the combustor. For Mach 6 flight, if supersonic combustion is used instead of subsonic, the cooling requirement is reduced by over a factor of two with no significant change in performance. For the Mach 8 case, if a value of area ratio across the combustor of 2.5 is used instead of 1.4, a significant reduction also is obtained with a slight loss in performance, less than 100 seconds of specific impulse.

Another method of reducing heat transfer is to maintain low temperatures and pressures next to the combustor wall over most of the length. This flow situation can be produced, without reducing the values of these parameters in the central portion of the flow, by aerodynamic design features of the inlet and by the fuel-injector design. The direct approach to reducing heat transfer is to insulate the wall by means of coatings, which might need refurbishment at intervals. Coatings could be very effective and research is needed in this area.

Cooling Requirements for Langley 3-D Scramjet Module

A photograph of a mockup of the Langley 3-D scramjet concept is given in figure 9; this configuration was described in the discussion of figures 2 and 3. The near sidewall has been removed from the model, as shown by the dashed lines, to provide a clear view of the internal design. It should be noted that the sweptback oblique shock waves originating at the widewall leading edges turn the flow slightly in a downward direction as well as toward the vertical center plane. The fairings at the top inside corners of the inlet are designed to follow streamlines downstream from these leading-edge shocks at the design Mach number. Research programs are in progress at Langley to develop the inlet, fuel injector, and combustor component designs for the concept; experimental evaluations cover a flight Mach number range from 3 to 8. The program plan includes performance evaluations using small-scale engine models when the component development is completed.

The improved heat-transfer method described previously has been used to predict the cooling required by this configuration for full-scale engines installed on a hypersonic cruise vehicle with a takeoff gross weight of 272 Mg (600 000 lb). The cruise altitudes corresponded to a free-stream dynamic pressure of approximately 16.7 kN/m^2 (350 psf). The results of this analysis are presented in figure 10 in terms of cooling required by the engine referenced to the heat sink available in the fuel given as a function of flight Mach number. The higher curve corresponds to uniform combustion across the duct, and it predicts that the engine will cool up to as least Mach 10. At lower Mach numbers excess heat sink would be available in the fuel for cooling areas of the vehicle. The potential use of actively cooled vehicle structures would provide the vehicle designer with considerable latitude in optimizing the configuration. (See ref. 6.)

The engines on hypersonic vehicles generally are sized to provide an acceleration margin at the top speed; as a result for a cruise condition, the engines have to be throttled to about 72 percent of the stoichiometric fuel-air ratio and therefore 25 percent of the airflow is not burned. If the fuel injectors were designed so that the unburned air flowed along the combustor walls, the cooling requirement would drop to the lower curve, which predicts that the engine would cool up to at least Mach 12.

It should be noted that both curves correspond to an assumption of a constant wall temperature; advanced designs for cooling circuits, which have not been developed yet, would be required to approach this condition. On the other hand, the successful development and use of insulation techniques or coatings would permit further reductions in the cooling load.

CONCLUDING REMARKS

The status of hypersonic air-breathing propulsion technology projected to at least the middle of the 1970 decade indicates that research and development on hypersonic cruise propulsion systems should be based on the use of the turbojet engine for acceleration to Mach 3 and either the subsonic combustion ramjet for cruise at Mach numbers up to 5, or the scramjet for cruise at Mach numbers up to 12. The ramjet-powered cruise vehicle could be a relatively near term development; however, the development of an airframe-integrated scramjet engine requires additional basic research and development to produce engine concepts with appropriate engine internal design features and the desired low internal cooling requirements. Current studies show that these goals are attainable by well-defined approaches. With an effective hypersonic air-breathing propulsion program, the work on the integrated engine concepts could be completed by 1975. The associated technology advances could then be used to develop an advanced research scramjet engine for flight testing and demonstration on a hypersonic research airplane in the first half of the 1980 decade. The development of the research engine could be accomplished in existing ground facilities.

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SUPERCHARGED EJECTOR RAMJET

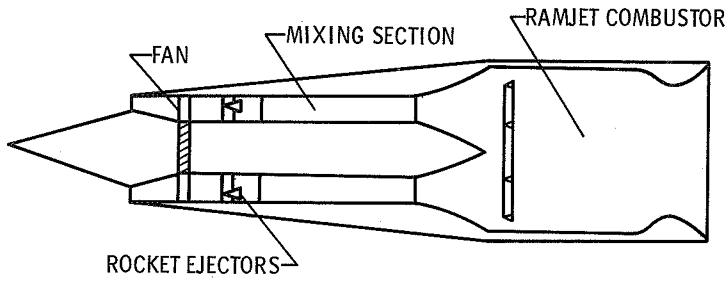


Figure 1

HYPERSONIC PROPULSION SYSTEM SCHEMATIC
TURBOJET OPERATION TO MACH 3

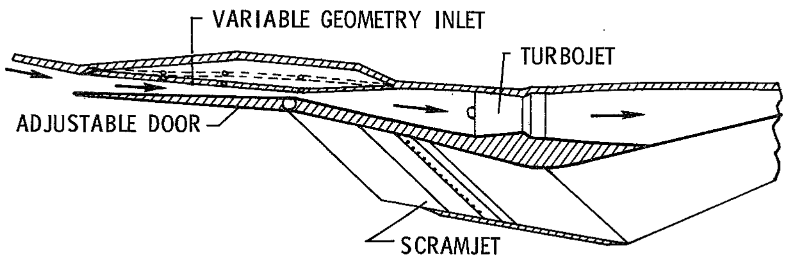
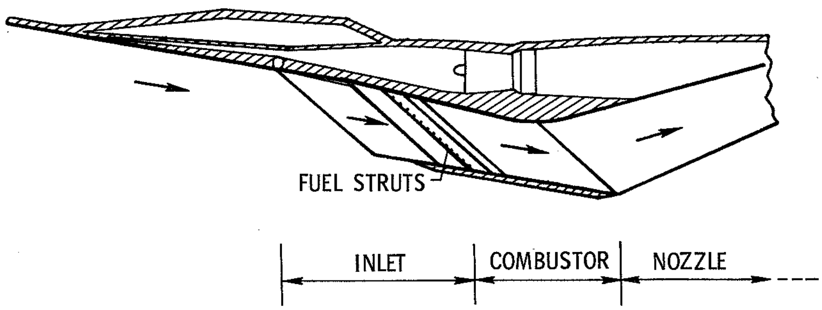


Figure 2

HYPERSONIC PROPULSION SYSTEM SCHEMATIC

SCRAMJET OPERATION ABOVE MACH 3



ISSUES = VARIABLE INLET AND COMBUSTOR DESIGN

Figure 3

EFFECT OF INLET VARIABLE GEOMETRY

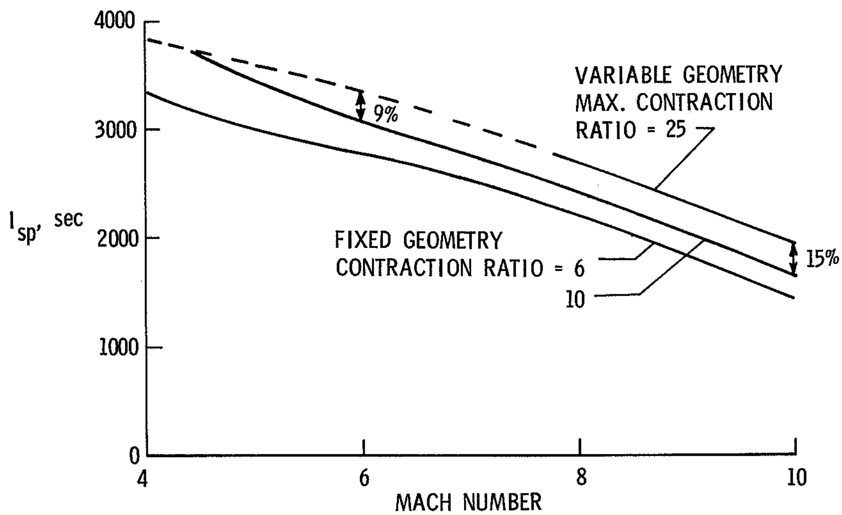


Figure 4

COMBUSTOR LENGTH WITH IN-STREAM FUEL INJECTION
(BASED ON MIXING LENGTH CRITERION)

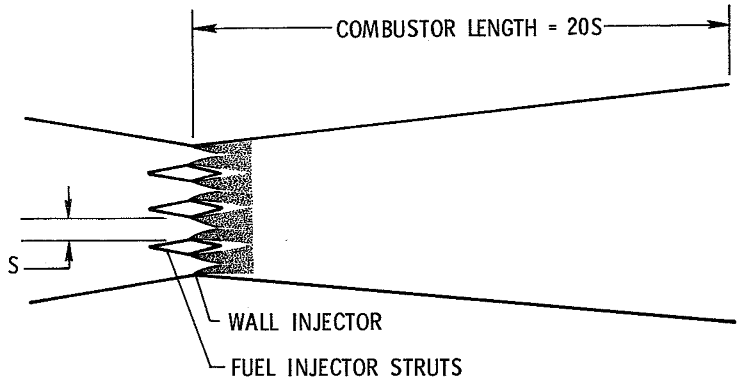


Figure 5

SUPERSONIC MIXING-COMBUSTION CORRELATION

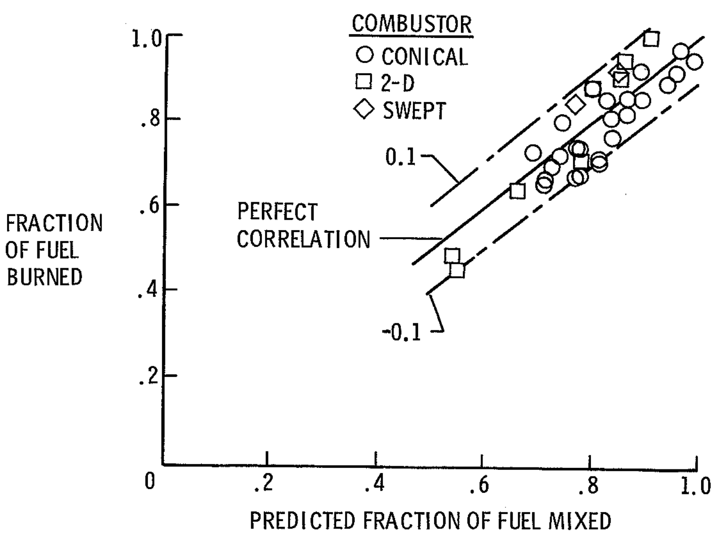


Figure 6

THERMAL COMPRESSION CONCEPT

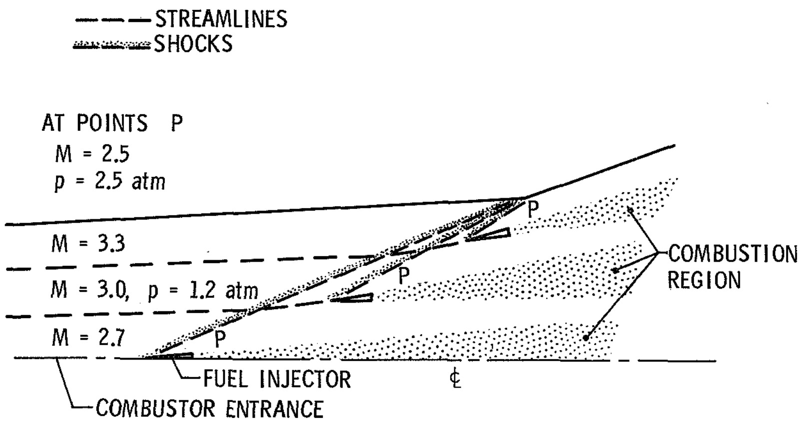


Figure 7

ENGINE DESIGN FEATURES FOR REDUCED COOLING REQUIREMENTS

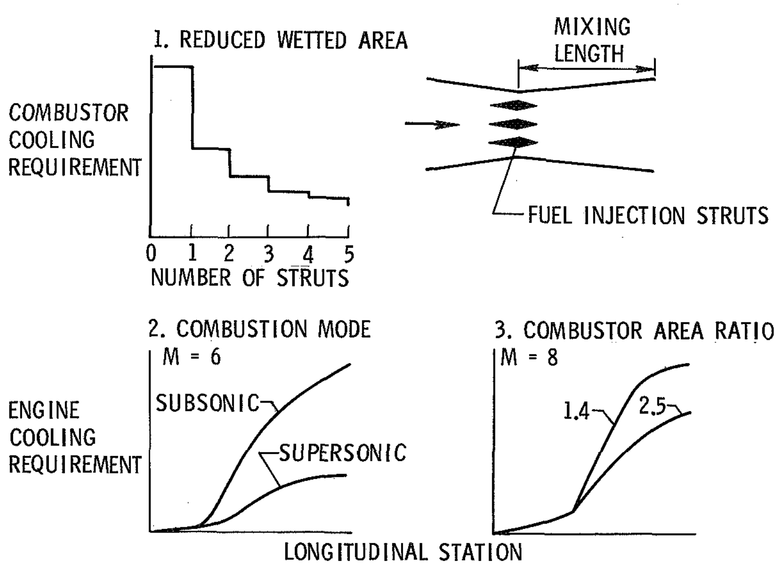


Figure 8

LANGLEY 3-D SCRAMJET MODULE

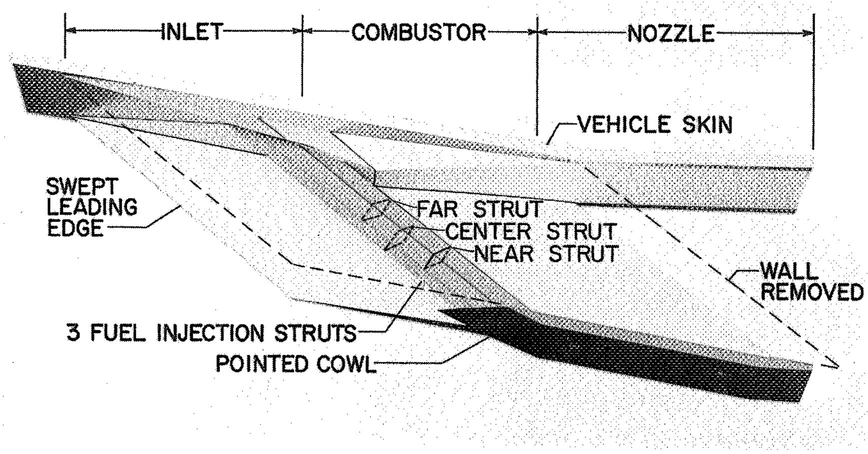


Figure 9

LANGLEY 3-D SCRAMJET MODULE COOLING REQUIREMENTS

CRUISE; ENGINE THROTTLED TO 75% STOICHIOMETRIC

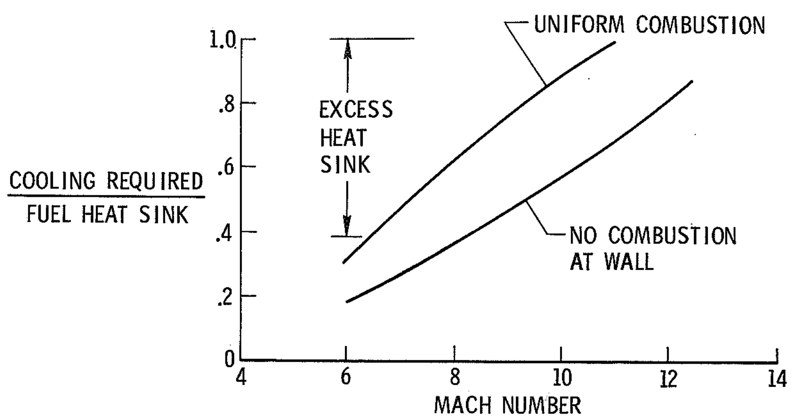


Figure 10

STRUCTURES TECHNOLOGY FOR HYPERSONIC VEHICLES

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SUMMARY

The present paper reviews the technology base that has been established for the primary structural components of hypersonic aircraft and indicates areas where additional research is required if hypersonic cruise flight is to become a practical reality.

INTRODUCTION

For hypersonic aircraft to become a practical reality, techniques must be developed for the design and fabrication of lightweight structures that can withstand repeated and prolonged exposure to the severe aerodynamic heating encountered in hypersonic flight. As for all aerospace structures the minimization of weight is critical; however, the hostile thermal environment and the unique requirements of cryogenic hydrogen tankage introduce additional structural design problems and constraints and add new dimensions to the problems of reliability, maintenance, and life.

The present paper reviews the technology base that has been established for the major structural components (propulsion structure, primary structure, surface structure, and liquid hydrogen tankage) and indicates areas where additional research is required if the goal of regular, manned, hypersonic cruise flight is to be realized.

TECHNOLOGY TRANSFER

Wherever possible, the structural technology generated by other hypersonic vehicles will be applied to the hypersonic aircraft. Figure 1 illustrates the environmental relationship of the hypersonic cruise vehicle to other hypersonic vehicles. The cruise vehicle, shown at the right of the figure, will be exposed to temperatures of the order of 1300° K for flight times of 1 to 2 hours corresponding to ranges of 8000 to 16 000 kilometers and will be expected to have a usable life in excess of 5000 flights. This requirement is in contrast to the vehicles shown on the left which have much shorter exposure times and in most cases only one to a few hundred mission requirements. One-shot ablation-covered entry vehicles provide us with little structures technology applicable to hypersonic cruise flight. The X-15 gave the first significant experience in dealing with elevated temperatures and thermal stress but its heavy heat-sink structure is not suitable for an efficient cruise vehicle. The YF-12 is more

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representative of a lightweight hot structure but with a temperature capability much lower than that required for hypersonic flight. Designs for the space shuttle are aimed at ultimate reusability with a primary structure protected by insulation which, in some cases, also forms the external surface. This approach is not applicable to the longer flight times of the cruise vehicle, which will probably require either a hot or an actively cooled structure.

Although the approach being used on the space shuttle is not directly applicable to the cruise vehicle, much of the technology being developed for the shuttle is of direct benefit as shown in figure 2. The four primary structural components are listed on the left with key items in the development process across the top. The extent of the shaded areas in each block is a rough indication of the technology required for a hypersonic vehicle that can be expected from the shuttle technology program. The propulsion structure will derive some benefit in the area of materials but little else. The hot or cooled structure approach that appears to be required for the hypersonic cruise vehicle is not being pursued for the shuttle; thus, technology transfer in the primary structures area is not large. However, technology being developed for the surface structures and liquid hydrogen tanks along with their thermal-protection systems is directly applicable in many areas. Especially important will be work in the area of concept development and component testing.

PROPULSION STRUCTURES

A major element in its own right and a key to the selection of a structural approach for the primary structure of the airframe is the development of suitable structures for the hypersonic airbreathing propulsion system. This structure is exposed to the most severe environment encountered anywhere on the aircraft. Since temperatures within the propulsion system exceed the capabilities of any known structural materials, active cooling of the structure is mandatory. Regenerative cooling (convective cooling in which the fuel is circulated throughout the structure before being burned) appears to be the most attractive form of cooling since (1) there is no weight penalty for the coolant, provided the cooling requirements remain less than the fuel requirements for propulsion, and (2) the thermal energy absorbed by the coolant is retained by the system and is not vented overboard as with an expendable coolant. Fortunately, cryogenic hydrogen, the most probable fuel for a hypersonic cruise aircraft, is an excellent coolant.

Although little of the shuttle technology is applicable to the propulsion structure, other NASA research, notably the hypersonic research engine project, has led to considerable advances in this field. As part of the project, a complete, hydrogen-cooled scramjet engine structure has been designed, fabricated, and tested in the Mach 7 flight environment of the Langley 8-foot high-temperature structures tunnel. (See ref. 1.) The engine, which is shown in figure 3 is a flight-weight structure that had to meet rather stringent weight requirements as it was originally designed for flight testing on the X-15 airplane. In the course of the wind-tunnel tests, the model was exposed to over 50 thermal cycles and accumulated over 30 minutes of high-temperature testing. The tests have conclusively proven the practicality of cooling an engine in the hypersonic flight regime.

The type of surface used in the construction of the hypersonic research engine is shown in figure 4. Portions of the top faceplate and fin material have been removed in the photograph to reveal the details of the coolant passages. The plate-fin surface is fabricated by brazing preformed fin material between two faceplates. The interrupted flow provided by the offset fins and the small passages are used to promote high heat transfer. The high heat transfer reduces temperature differences between the exposed surface and the underlying structure. These temperature differences are the cause of thermal stresses which lead to fatigue failures.

The design and fabrication of the heat exchanger surface to sustain several thousand cycles of high pressure and temperature is perhaps the number one structural problem for the propulsion system. The relationship between temperature difference and fatigue life for the plate-fin structure is shown in figure 5. The curve which was based on the procedure of reference 2 has been adjusted, by use of a strain concentration factor, to agree with the experimental data which were obtained from references 3 and 4. Even though the depth of the heat-exchanger surface for the hypersonic research engine was made small (1.25 mm) to minimize temperature differences, areas of highest heating still experienced temperature differences in excess of 450° K. Although this value was acceptable for the research engine, which had a design life of 100 cycles, it is clearly unacceptable for viable aircraft engines which must have usable lives of several thousand cycles; furthermore, projections of the curve to lifetimes of several thousand cycles indicates temperature differences that cannot be attained by present approaches. Improved material characteristics, insulative coatings, better designs and manufacturing methods, and lower operating temperatures will improve this situation. These areas will be the focus of research over the next decade.

PRIMARY STRUCTURE

Two competitive approaches are being considered for the primary structure of the hypersonic cruise vehicle. One utilizes an active cooling system to maintain the structural temperature at a level compatible with conventional structural materials (actively cooled structure); the other one permits the primary structure to reach its radiation equilibrium temperature (hot structure). Each of these approaches has distinctive advantages and disadvantages. Additional research is needed for both to develop a base from which a rational selection can be made.

Actively Cooled Primary Structure

Advances in engine technology leading to low cooling requirements, described in paper no. 8 of this compilation, make it possible to consider active cooling of an entire hypersonic aircraft. Cooling requirements for an airframe alone and airframe plus engine are presented in figure 6 as fractions of the cooling capacity of the fuel flowing to the engines. (A value of one or less indicates that the fuel required for propulsion is sufficient to provide the required cooling.) The airframe alone, as indicated by the bottom curve, requires only a small fraction of the total cooling capacity. However, with present engine technology, as indicated by the top curve, active cooling does not appear to be practical. With the advanced engine technology active cooling of the entire aircraft may be feasible over a wide Mach number range as indicated by the lower curve. It should be noted, however, that the thermal analysis assumes engine-cooling performance that is beyond present technology. To achieve the indicated performance, considerable progress must be made in engine structure design and fabrication in terms of cooling efficiency and fatigue life.

Active cooling systems for a Mach 6 cruise vehicle have been examined in some detail in references 5 and 6. One method, slot cooling, is discussed in paper no. 5 of this compilation. An alternative method that has been found attractive, indirect convective cooling, is illustrated in figure 7. In this method, to avoid routing hydrogen throughout the airframe surface, a central heat exchanger is used with a secondary coolant circulated adjacent to the external surface in a closed loop system. Advantages of a cooled structure (discussed more fully in paper no. 25 of this compilation) are that lightweight structural alloys can be operated in an efficient temperature range with thermal stresses minimized to a great extent; thus an efficient structural design can be obtained. In addition, conventional fabrication techniques can be employed to construct components with sharp leading edges and aerodynamically smooth surfaces. Composites including the metal matrix composites will be attractive in this application. The major disadvantage of cooled structures is the system complexity represented by the cooling circuits, heat exchanger, and associated pumping. These systems must be designed to assure safe operation and prevention of catastrophic failures. Obviously, much study will be required to evaluate this approach properly.

Hot Structure

The greatest amount of past research on primary structure for hypersonic vehicles has been devoted to the hot structure approach. The result of a recent study (ref. 7) to determine efficient concepts for hypersonic vehicle wing structures is shown in figure 8. The René 41 structural panels are beaded to obtain maximum stiffness per unit weight and to allow for thermal expansion without thermal stress. Lightweight corrugated heat shields provide the aerodynamic surface and in combination with insulation control the temperature of the structure. The leading edge is segmented to minimize thermal stress. This structure was designed for a Mach 8 vehicle with a range of 8000 kilometers and a lifetime of 10 000 hours. It was found in the study that the major problem areas were thermal stress and material behavior under long-time cyclic exposure to elevated temperatures. In addition, actuators and other mechanisms with their associated joints and seals will require special consideration to withstand the elevated temperature environment. An experimental program to evaluate a hot-wing structure under simulated loading and heating is currently underway at the NASA Flight Research Center.

A major part of the design of hot structures is material selection. The options that the designer has available are indicated in figure 9. The range of applicability of various materials is shown along the vertical temperature scale. The upper curve is representative of the maximum leading-edge temperature as a function of Mach number. The lower curve is representative of the surface temperature for most of the remaining vehicle, as indicated by the shaded area. The boundaries shown are somewhat a matter of judgment but are generally based on long-time material properties such as strength, oxidation, and creep after cyclic exposure to elevated temperatures. Data of the type shown in reference 8 form the basis for this information. The superalloys can be considered to be sufficiently developed and characterized to withstand temperatures up to about 1050° K and would give a Mach number capability approaching 8. The dispersion strengthened alloys such as TD nickel chrome being developed for the shuttle may some time in the future be considered for primary structure to around 1350° K but at present their low strength limits their application to heat shields. There are a host of candidates for leading-edge application, but above 1480° K no material is projected to have properties that will result in a completely reusable vehicle and some refurbishment is necessary.

Since only a small percentage of the vehicle area requires leading-edge materials, special methods of reducing these temperatures are practical. One approach utilizing heat pipes has been reported in reference 9 and its application is indicated in figure 10. Conventional cylindrical heat pipes have successfully operated at the heating rates associated with the leading edge. For leading-edge application, a two-dimensional heat pipe is configured to the shape of the leading edge, as indicated by the darkened area. Heat is absorbed in the stagnation area and rejected along the upper surface. Since the heat pipe operates at essentially constant temperature, a wing temperature distribution in the vicinity of the leading edge is reduced from that shown by the dashed curve to that shown by the solid curve. Resulting temperatures are such that reasonably long life materials can be used. In addition, fairly sharp leading edges are possible with this approach. However, additional studies and tests are required to assess the performance of nonsymmetrically heated, noncircular heat pipes in a dynamic flight environment.

SURFACE STRUCTURE

External heat shields are required for hypersonic vehicles to limit structural temperatures and to control thermal stress. The shuttle technology program is providing basic information for design of these structures. A metallic heat shield currently undergoing wind-tunnel and acoustic tests for shuttle application is shown in figure 11. Besides the basic material problem of oxidation and creep, areas being investigated are dynamic response, heat transfer, hot-gas flow through joints and seals, and overall structural integrity of the panel and insulation package. A strong analytic program is also underway to understand the dynamic and static response of this lightweight structure to a high-temperature aerodynamic environment. Thus, the shuttle technology program should provide the necessary ingredients to design such structures for hypersonic vehicles when materials are properly characterized for the long-time cyclic exposure to elevated temperature.

LIQUID HYDROGEN TANKS

The final area of concern is the special problem of liquid hydrogen tankage. Possible approaches to this problem are shown in figure 12. The fundamental requirement is the prevention of air coming in contact with a tank wall that is at liquid hydrogen temperatures. Otherwise, liquefaction of air and the resulting cryogenic pumping would cause high heat transfer to the fuel. In addition, selective liquefaction might lead to dangerous accumulations of liquid oxygen. A number of solutions to this problem are discussed in reference 10. A vacuum-tight outer shell placed outside the tanks would be an ideal solution but attempts to accomplish this approach in flight-weight construction have not been successful. The only system that has definitely reached the state of the art is the purge system. (See ref. 10.) The purge concept requires the space adjacent to the tank wall to be pressurized with an inert gas. However, it is a complex system, and the usual purge gas, helium, results in high system weight and cost. NASA is sponsoring development of more efficient and less costly systems using the more available gases such as carbon dioxide and nitrogen. Internal foam insulation has been successfully flown on Saturn vehicles and is currently being developed for the space shuttle with at least 100 reuse cycles. The insulation traps a gaseous hydrogen layer which effectively prevents the liquid hydrogen from contacting the tank wall. An external purge may also be required for safety and it must be demonstrated that the system has adequate cycle life.

CONCLUDING REMARKS

Although hypersonic flight on a regular basis may be more than a decade away, much research in the area of structures that is directly applicable is planned or already underway. Major aspects of this program have been outlined, and although details may change, the goals and requirements that have been presented will continue to be the focus for research in the future.

For propulsion structure, adequate lifetime in the presence of thermal fatigue is the number one problem. Attractive ways to accomplish this goal are improved fabrication processes and new design concepts that minimize thermal stress. Successful development of high-temperature insulative coatings can also reduce thermal stress and result in less coolant requirements.

The primary structure represents a significant part of vehicle weight and efforts here are concentrated on achieving the efficient lightweight designs necessary for practical vehicles. Toward this end the potential of actively cooled structures will be assessed, and if they prove attractive, an experimental verification program will follow. Research on hot structures will continue with emphasis on the testing of major components. No matter which approach is used, application of automated design methods will help to identify the most attractive approaches and should result in lighter structures.

Other than the engine, the surface-structure and thermal-protection system are probably subjected to the most severe environment but are not required to carry primary structures loading. The primary goal then is adequate life and structural integrity for these lightweight structures. This goal can be accomplished by application and extension of shuttle technology in the areas of material characterization, acoustic testing, and flutter and vibration analysis and test.

The major goal for the liquid hydrogen tank thermal-protection system is to improve efficiency and reduce complexity. Programs for development of simpler and more efficient purge systems are underway and the shuttle should provide a basis for extension of the internal insulation systems if they prove feasible for long-life cyclic use.

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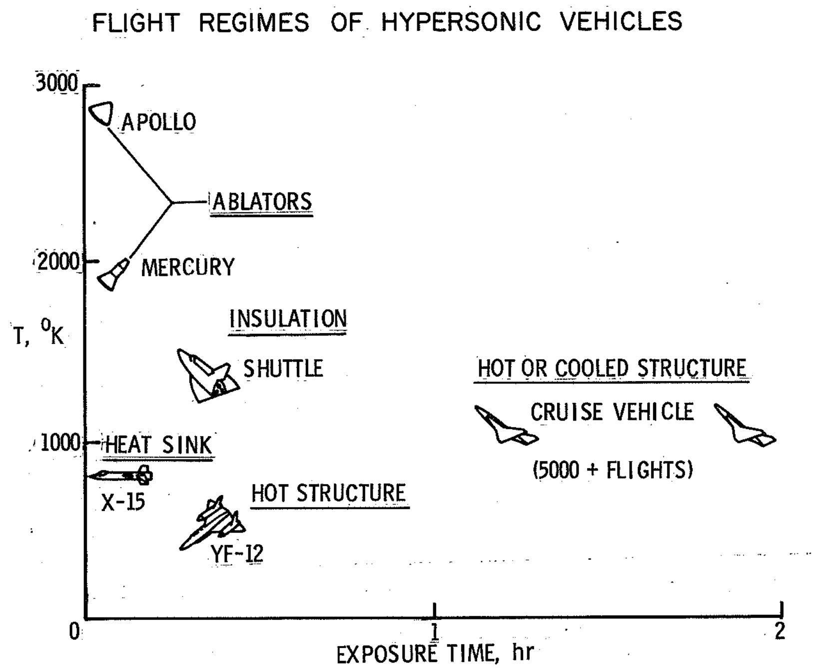

Figure 1

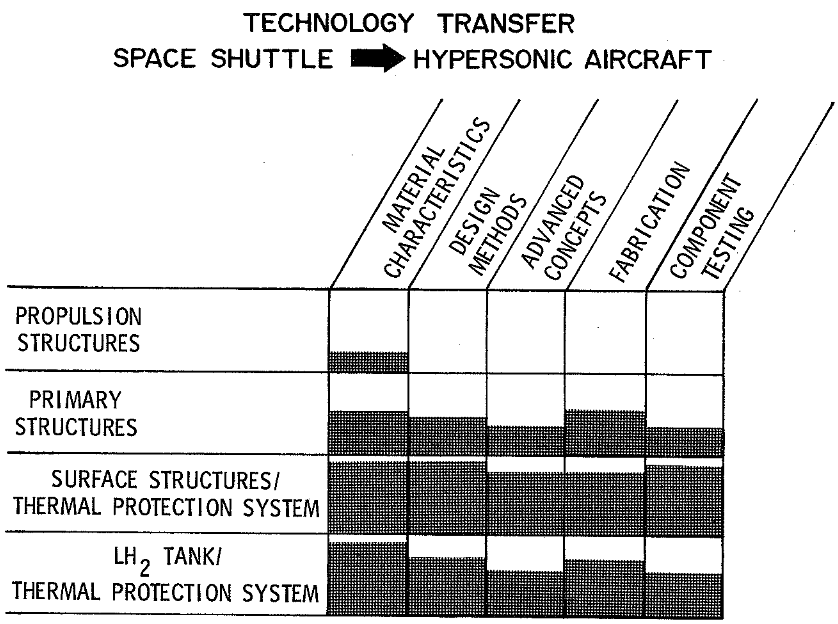

Figure 2

HYPERSONIC RESEARCH ENGINE

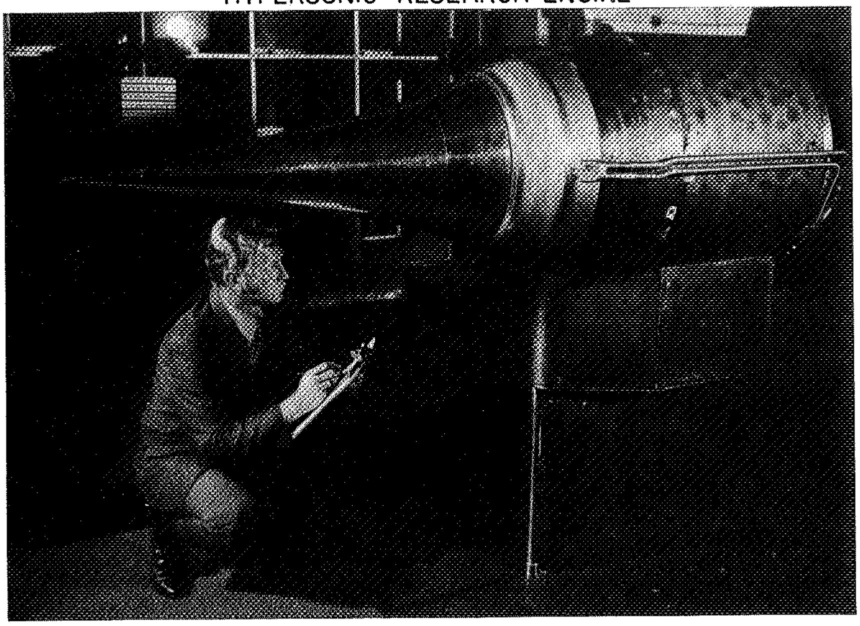

Figure 3

HYDROGEN COOLED SURFACE

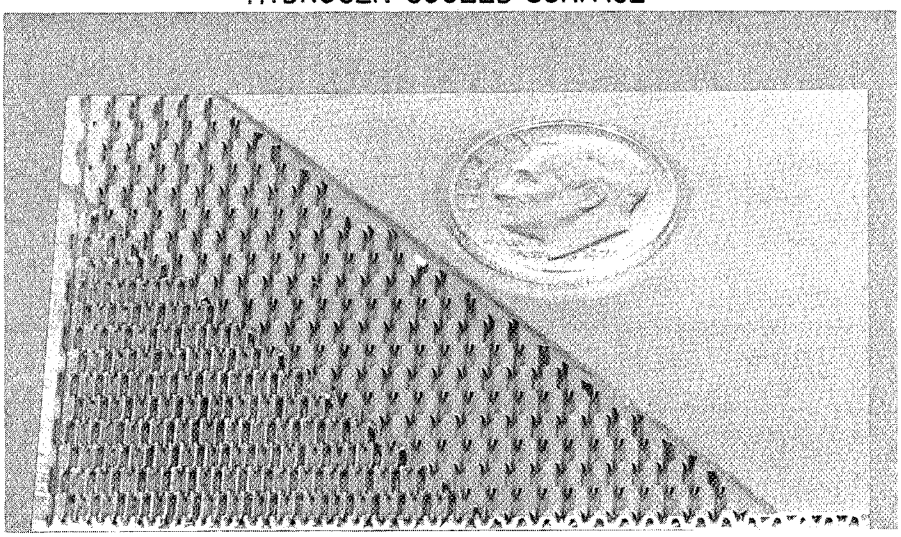

Figure 4

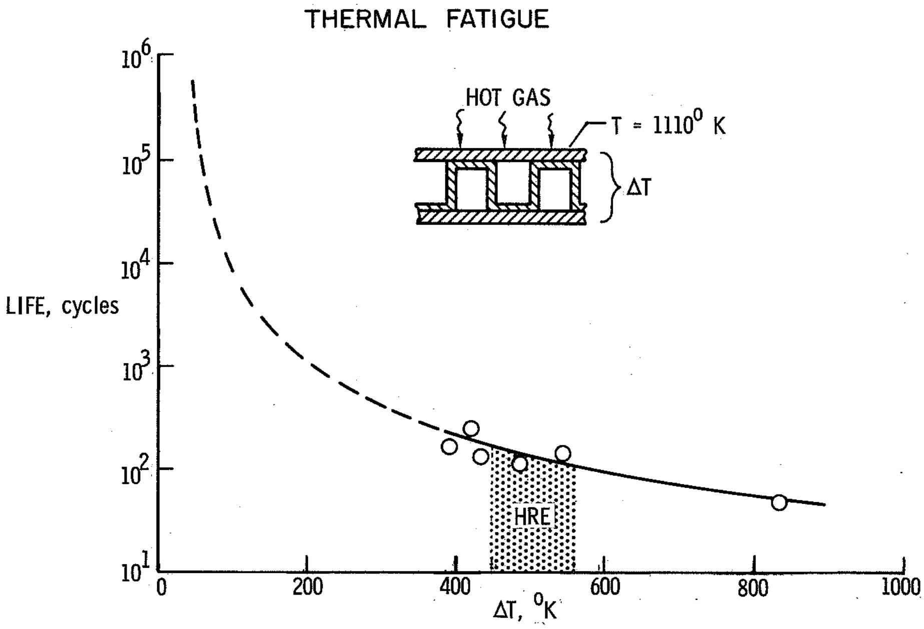

Figure 5

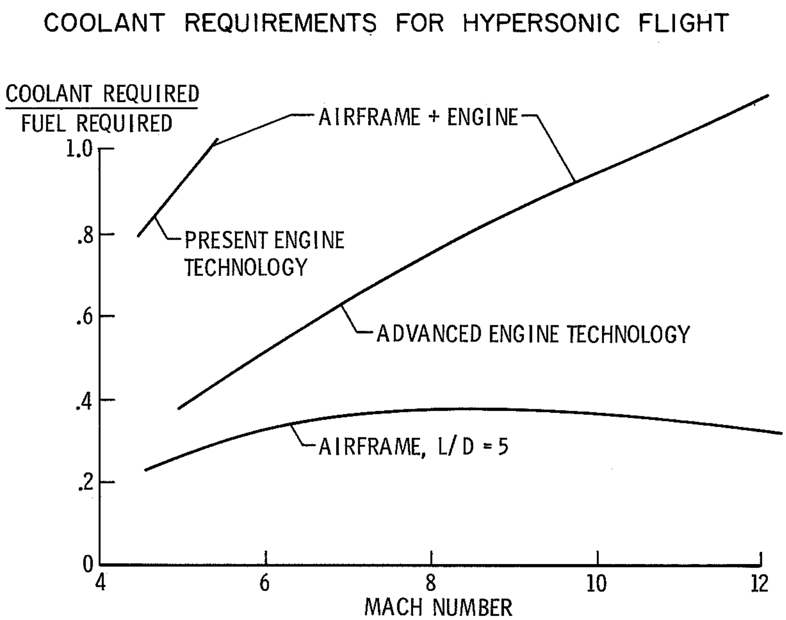

Figure 6

ACTIVELY COOLED HYPERSONIC VEHICLE STRUCTURE

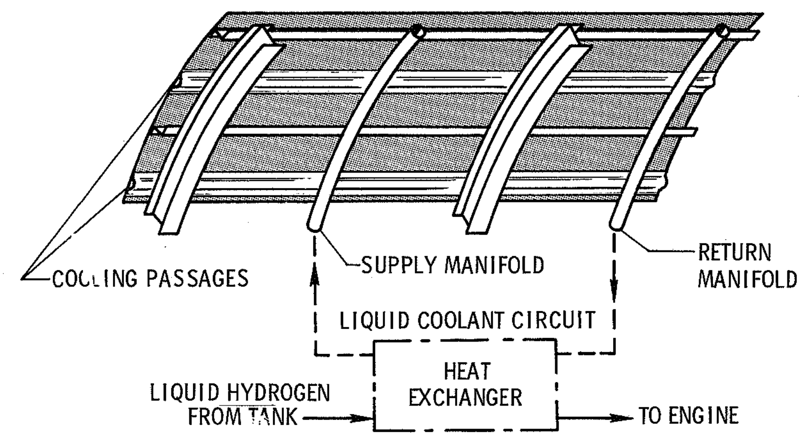

Figure 7

HOT WING STRUCTURES

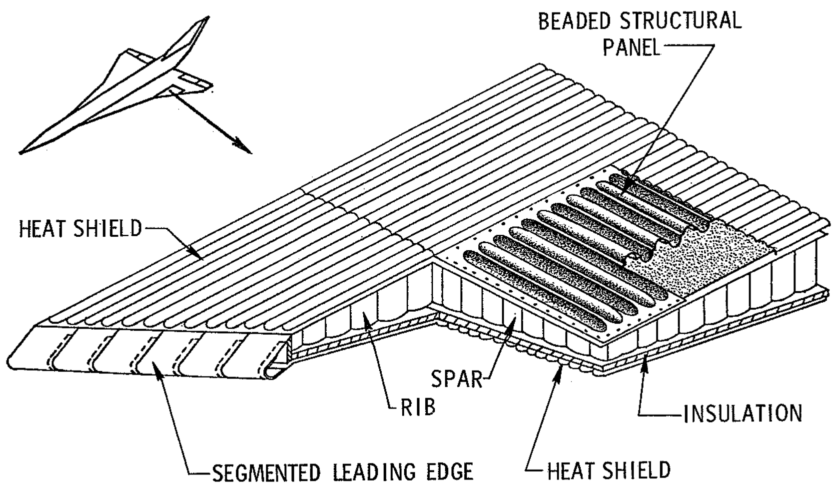

Figure 8

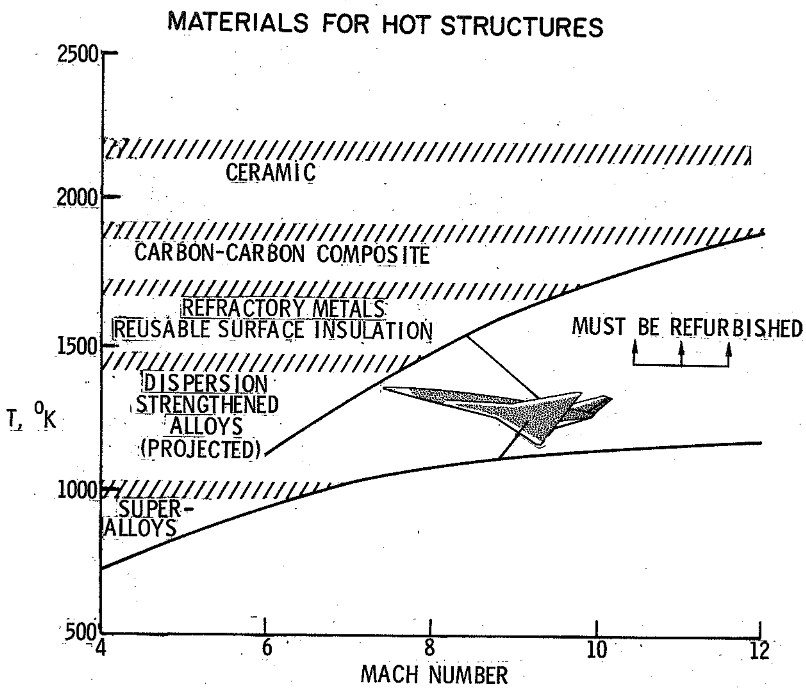

Figure 9

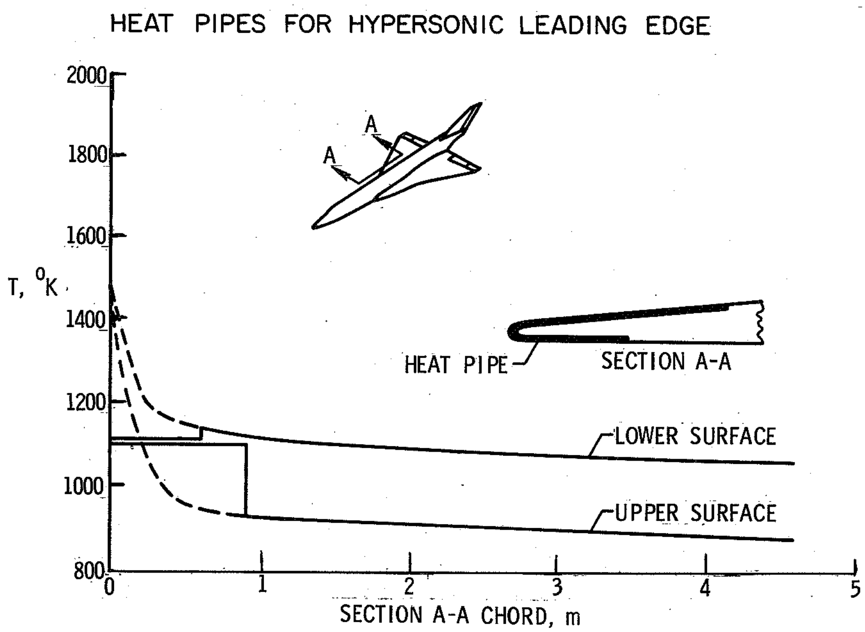

Figure 10

METALLIC HEAT-SHIELD ASSEMBLY

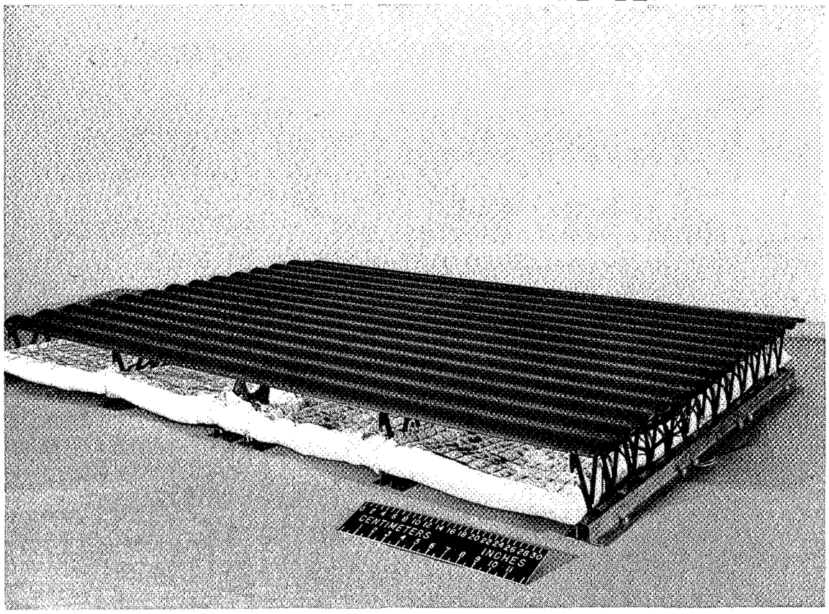

Figure 11

THERMAL PROTECTION OF LIQUID HYDROGEN TANKS

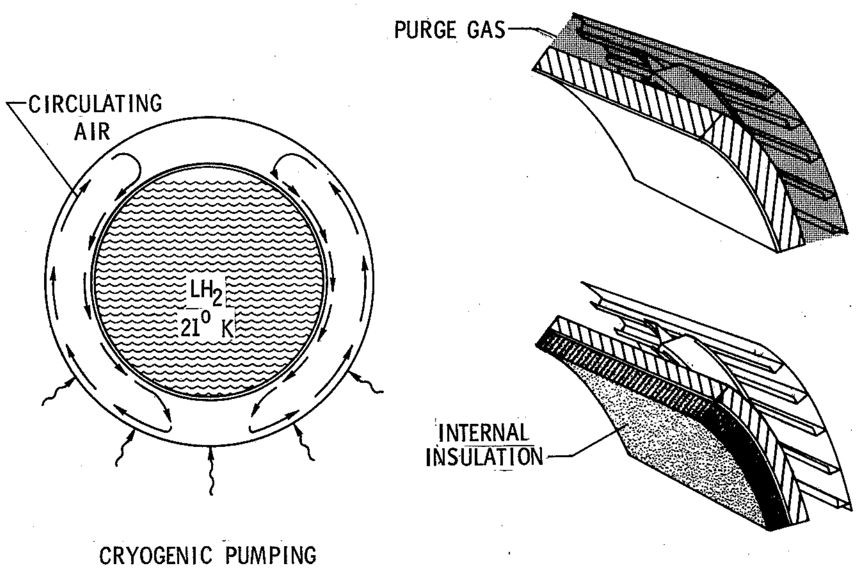

Figure 12

MATERIALS APPLICATION TO CIVIL AIRCRAFT STRUCTURES IN THE SEVENTIES AND BEYOND

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INTRODUCTION

This paper presents both the current status and projections for materials usage for the 1970's and beyond in civil aircraft structures. It will discuss both metal structural alloys and composite materials, with most of the emphasis on composites. Such drivers for future development as fracture toughness and corrosion resistance for metals and response to environment, maintainability, and costs for composites will be considered. Programs which will demonstrate the benefits to be realized such as weight reduction, improved performance, and increased longevity will be presented.

Figure 1 shows the degree to which various materials are used in aircraft structures. At the present time, a typical subsonic aircraft has a ratio of structural weight to gross weight of about 0.27. The structure is predominantly aluminum alloys with steel landing-gear assemblies. Small percentages are titanium alloys and glass-fiber-epoxy composites. During the next 5 years, there will be much development work with advanced composites which will begin to show up in civil aircraft in secondary structures such as flaps, spoilers, and leading and trailing edges. As this usage increases, the structural weight of these components will go down and so will the structural weight of the entire aircraft. By the end of the decade up to one-half the structural weight may be composites with a resulting overall decrease in structural weight to about 80 percent of today's value. During this same time, there is a good chance that titanium alloys may replace steel in landing gears in order to reduce maintenance and increase longevity.

For supersonic transports, a similar picture of material usage is predicted, but with titanium and aluminum alloys interchanged. Again, composites are expected to show a 20-percent overall structural weight saving, but the kinds of composites will be metal-matrix and polyimide-type polymeric matrix for high-temperature usage.

RESULTS AND DISCUSSION

Metals

Some of the material characteristics of metal alloys which may lead to problems in applications to aircraft during the next decade are indicated in table I. Listed are two common structural alloys each for aluminum, titanium, and steel. Stiffness and strength

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values for these alloys are generally satisfactory. The aluminum alloy 2024-T3 could benefit from increased strength, but generally when such strength increases are sought through alloying or new heat treatments, some or all of the other characteristics degrade and become more serious problems. The titanium alloy Ti-6Al-4V, for example, in the annealed state has a strength which is acceptable, but it can be increased markedly by heat treatment. However, hot-salt stress corrosion and fracture toughness, which are already at marginal values, become much more severe with the higher strength heat treatments. Similarly, the steels which generally are highly heat-treated for landing-gear applications become susceptible to corrosion, loss of fracture toughness, and hydrogen embrittlement. Last, weldability, which is more a requirement for advanced manufacturing techniques, is a problem for all the alloys listed except the Ti-6Al-4V; and even here it may cause loss of fracture toughness if not carefully controlled.

During the next decade, developments in metal technology will aim at improving or eliminating the problem areas of corrosion, fracture toughness, and hydrogen embrittlement either by alloy modification or processing variations. It is hoped that these improvements will be achieved without too much loss in strength. Thus the aircraft industry should have the benefits of improved maintainability and longer life from the airframe metals without much change in structural weight.

Composites

As was shown in figure 1, reductions in structural weight are going to occur because of a significant use of composite materials. Some constituents of composite materials are listed in table II. These materials can be combined to form numerous composite materials in which the principal constituents are fibers, matrices, and bonds. The current development status of a variety of fiber, matrix, and bond materials is also shown in table II. Each horizontal row is a listing of constituents arranged in order of their current status. Thus the only established, in-service constituents today are glass fibers, epoxy matrices, and epoxy adhesives. However, in general, combinations are not formed in lines across the rows. Between "Established" and "Research and Development Hardware" are first boron and then graphite fibers, either of which can be used with epoxy matrices or with more development-oriented polyimide and aluminum matrices. Still further away from established hardware are materials which are only beginning to be characterized, but which show a lot of potential for future acceptance.

Although there is an abundance of composite constituents from which to choose, one of the most difficult problems facing a designer today is that of developing an effective means of transferring load into the composite fibers. Several concepts of composite joints are shown in figure 2. Every composite component has to be built today so it can be bolted, riveted, welded, or glued to the rest of the airplane. Thus it generally has to terminate in a metal edge piece or end fitting, and all the load must be transferred from the composite to the metal. The single lap joint is the easiest to fabricate, but is very inefficient in transferring high loads. However, in many stiffness critical applications, where the required design strength is low, the single lap joint is completely satisfactory. For high loads, the most efficient joint theoretically is the scarf joint (fig. 2). However, from a practical point of view, it is difficult to achieve all the theoretical efficiency because a very small-angle taper must be machined on both the metal and the composite, and then the two must be bonded together precisely. Also, unless the composite and the metal are equal in strength, a penalty will be imposed on the stronger. The multiple-step joint, although a little more complex, can be essentially as efficient as the scarf if properly designed with different numbers of composite layers and different step lengths. Failures generally will initiate under tensile loading in all the joints shown, but good design practice can introduce additional plies of composite material in the immediate vicinity of the joint and thus lower the stresses and increase the margin of safety in the joint.

Joints are just one of the problems in the use of composites today. Figure 3 presents some of the other areas which require additional research for composite usage in structures. Strength and stiffness, fatigue and fracture, and design methodology are engineering characteristics of composite laminates which have already been investigated extensively but which still require considerably more investigation. Fatigue of composites is discussed by Herbert F. Hardrath in paper no. 11. New techniques for manufacturing and nondestructive evaluation of finished parts need extensive development. Maintainability and environmental behavior are factors which require long-time exposure to an operational situation. Data for both are lacking today. The biggest factor limiting composite usage will be cost, both the material cost and the cost involved with the interaction between manufacturing, maintainability, and environmental behavior. It is predicted, however, that all the factors which require additional research for composite usage today will be more than 80 percent resolved in the next decade so that the benefits of weight saving in producing lower operating costs and higher return on investment will be realized.

A large number of programs for application of composites to aircraft structural components have been underway in the United States. Results of some of these programs are given in table III to demonstrate the weight saving in various components. Represented are 79 out of a total of about 100 programs. Weight-saving potential of composites is well demonstrated, with wings, fuselages, and tails ranging from 9 to 25 percent; and secondary structures, from 20 to 47 percent. These are not complete wings or fuselages, but rather structural boxes or sections representative of the load-carrying portions. Although the weight savings is well demonstrated, other problems do exist. Not all these components met their design strength, and manufacturing usually was by hand methods, which are costly. Of the few which have been flown on aircraft, the F-14 stabilizer and the C-5A slats are the largest; the F-4 rudders are the most numerous; and a 707 flap has the most flight time, more than 4000 hours. It is obvious that in the next few years more composite structures need to be flown if enough is to be learned about environmental behavior and maintainability to gain the necessary confidence for major commitment to future aircraft. Most of the existing composite programs in this table are sponsored by the Department of Defense and the published data are restricted in distribution, or they are company research and development programs which are proprietary. Therefore, the examples which will be discussed subsequently will of necessity be NASA programs; however, they are indicative of the great potential to be realized from composite materials.

A contractual program with The Boeing Company to develop the technology of metal aircraft structures reinforced with filamentary composites has been underway for several years. Initial results are available in references 1 and 2. The investigation has culminated in the design, fabrication, and test of three large fuselage skin panels representative of civil transport design criteria (fig. 4). These panels were designed to explore composite materials used with the conventional metals in each of three loading conditions, tension, compression, and shear. Each of the three panels was designed to carry the same loading intensity as the existing all-metal panel in current use.

A drawing of the tension panel showing the essential details is given in figure 5. This 1.2- by 3.0-meter honeycomb-sandwich panel has boron-epoxy composite material underneath the aluminum skins which helps to carry the hoop tension and results in a 20-percent weight saving.

A compression panel 0.9 by 2.4 meters with boron-epoxy bonded to the stringers for reinforcement is shown in figure 6. This design demonstrated a 25-percent weight saving based on the total panel weight and was loaded to failure at 115 percent of design ultimate.

Figure 7 is a 1.8-meter-square window-belt shear panel with multidirectional composite reinforcement around the windows. The arrows around the edges indicate the applied shear load. This panel also demonstrated a 25-percent total weight saving and carried 120 percent of design ultimate shear load before it failed.

The next program is a cooperative program with the U.S. Army to develop a composite tail-rotor drive shaft for the UH-1 helicopter. Figure 8 shows one of the first of these drive-shaft segments that has been designed and fabricated with graphite-epoxy. The same aluminum end fittings that are riveted to the production aluminum shaft are adhesively bonded to this composite shaft. Testing to date has shown it to have 3 times greater stiffness and a cyclic torque capability for 15 million engine startups without mechanical degradation. As shown by the bar graph (fig. 8), the graphite-epoxy shaft including the end fittings has only 67 percent of the weight of the aluminum shaft.

Another composites program is underway with the CH-54B helicopter shown in figure 9. This is the Army "Crane" helicopter which has the greatest lifting capacity of any U.S. helicopter. When this helicopter was designed, the 6-meter-long tail cone, or aft fuselage, was designed to satisfy a static-strength requirement as shown in figure 10. When checked for dynamic-stiffness requirements, it was found that more bending material was needed to satisfy the dynamic requirement than was needed for the static strength. The resulting aluminum production design with heavy skins top and bottom weighed 175 kg. Sikorsky Aircraft, Division of United Aircraft Corp., is working under NASA contract to design and fabricate a tail cone which has lighter gage skins top and bottom sufficient to meet the static-strength criteria. The tail cone then has added boron-epoxy strips bonded to the stringers in sufficient quantity to restore the dynamic-bending-stiffness requirement (ref. 3). This design has been fabricated and weighs 118 kg, a saving of 30 percent. Flight testing will be conducted in early 1972 and then the helicopter will be delivered to the Army for routine flight service. The composite material behavior will be monitored closely for the first 2 years at least.

A somewhat different program aimed at getting flight-service experience with composites is the C-130 center wing box shown in figure 11. The C-130 transport airplanes have experienced a rapid accumulation of flight-hours in Air Force service, and a number of them have been retrofitted with a "beefed-up" aluminum center wing box. A recent Lockheed study (ref. 4) has indicated that in place of the beefed-up aluminum, about 230 kg of boron-epoxy bonded to the skin and stringers of this 11-meter-long box, as shown, can reduce the stress levels and increase the fatigue life as much as the aluminum retrofit design – but with a 13-percent weight saving.

A number of benefits are expected to be achieved with the C-130 composite-reinforced wing in a program which will build three wing boxes, ground test one, and install two in airplanes to be flown in regular Air Force service. The long-time service experience will be the principal benefit of the C-130 program and will prove the operational capability of the composite application. The design will demonstrate the enhancement of structural performance, in this case an improved fatigue life. Fabrication of three full-scale wing boxes will prove the feasibility of manufacturing. The flight service will demonstrate the composite effectiveness without the risks inherent at the present time in an all-composite approach.

The previously mentioned figures and tables have shown examples of the principal benefits to be derived from application of composites: reduced weight with equal or better stiffness, static strength, and fatigue life, but without regard to cost. The strong interaction of cost and weight is indicated in figure 12. The example shown is worked for a particular 1.1-meter-long tubular column of boron-epoxy (B-E) which is designed to carry a compressive load of 790 kN. Three designs are shown: all-titanium, all-composite, and 33 percent composite-67 percent titanium. Most of the weight saving that can be achieved is obtained with a relatively small amount of composite. In order to get the maximum 60-percent weight saving for this column, the total cost increases from $400 to $1000. Reductions in costs of composite material in the future will tend to flatten this curve and make composite structures more competitive with metal aircraft structures. Within the decade, the prices on all the surviving composites are predicted to come down to around $55 to $70 per kg ($25 to $30 per pound). (See ref. 5, for example.) The data shown in this figure are based on an idealized column design. When typical metal end fittings are included, the real weight saving is not as large, more on the order of 25 percent.

The trend in structural weight saving that is expected to be achieved for complete airframes with a range of composite utilization is shown in figure 13. The band indicates a range of optimistic engineering designs, but weight savings are apparent throughout. The open data points are a few of the major military components (from table III) that have already been built and tested and are shown to lend credibility to the projection. The solid symbols are secondary structures, such as control surfaces. Because they generally are stiffness critical, they show a greater weight saving; but since they represent only a small percentage of the total airframe, their impact is not as great. Advancing technology will drive all these points downward slightly, but most of the weight saving has already been achieved. Gains will come through greater usage as material costs and manufacturing effort are reduced. As was indicated earlier, by 1981 it is anticipated that the industry will be building airframes which are 50 percent composite and which will have a 20-percent weight saving. This projection applies equally to subsonic and supersonic aircraft.

CONCLUDING REMARKS

The projections made on material usage for civil aircraft structures show that by 1981 the industry is expected to be building airframes that are composed of 50 percent composite materials and 50 percent metal alloys. This composite utilization will produce a 20-percent structural weight saving which is projected for either subsonic or supersonic aircraft. The 50 percent of the airframe that will be metal alloys will probably not be significantly stronger or lighter, but rather will seek to provide longer life and lower maintenance requirements through improved corrosion resistance and fracture toughness. In order to give the designer the confidence to take advantage of the 20-percent reduction in structural weight due to composites, programs should be continued and expanded toward providing adequate nondestructive inspection techniques, a history of satisfactory maintainability, and a demonstrated favorable response to long-time flight environments.

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TABLE I

MATERIAL CHARACTERISTICS WHICH LEAD TO PROBLEMS IN STRUCTURAL METALS FOR AIRCRAFT

METAL	ALUMINUM ALLOY		TITANIUM ALLOY		STEEL	
CHARACTERISTIC	2024-T3	7075-T6	6Al-4V	6Al-6V-2Sn	4340	300-M
STIFFNESS						
STRENGTH	X					
CORROSION		X	X	X	X	X
FRACTURE TOUGHNESS		X	X	X	X	X
HYDROGEN EMBRITTLEMENT					X	X
WELDABILITY	X	X		X	X	X

X PROBLEM

TABLE II

DEVELOPMENT STATUS OF VARIOUS COMPOSITE MATERIALS

CONSTITUENTS \ STATUS	ESTABLISHED	RESEARCH AND DEVELOPMENT HARDWARE	MATERIAL CHARACTERIZATION
FIBER	GLASS, BORON	GRAPHITE, PRD-49	CERAMICS, POLYMERS
MATRIX	EPOXY, POLYIMIDE, ALUMINUM	POLYQUINOXALINE, MODIFIED EPOXY	TITANIUM, NICKEL, HIGHER TEMP. ORGANICS
BOND	EPOXY ADHESIVE, POLYIMIDE ADHESIVE	METAL BRAZE, METAL DIFFUSION	RADIATION-ACTIVATED ADHESIVE

TABLE III

COMPONENT PROGRAMS IN APPLICATION OF COMPOSITES TO AIRCRAFT STRUCTURES

COMPLETED AND CURRENT

COMPONENT	NUMBER OF PROGRAMS	WEIGHT SAVING, PERCENT
WINGS	11	9 TO 15
FUSELAGES	5	19 TO 25
STABILIZERS AND STABILATORS	10	15 TO 25
FINS AND RUDDERS	5	20 TO 35
SLATS AND FLAPS	8	22 TO 47
SPEED BRAKES, FENCES, AND FAIRINGS	13	23 TO 32
LANDING-GEAR DOORS	5	29 TO 36
HELICOPTER BLADES	4	
HELICOPTER AND V/STOL SHAFTS AND HUBS	3	30 TO 43
MISCELLANEOUS	15	
TOTAL :	79	

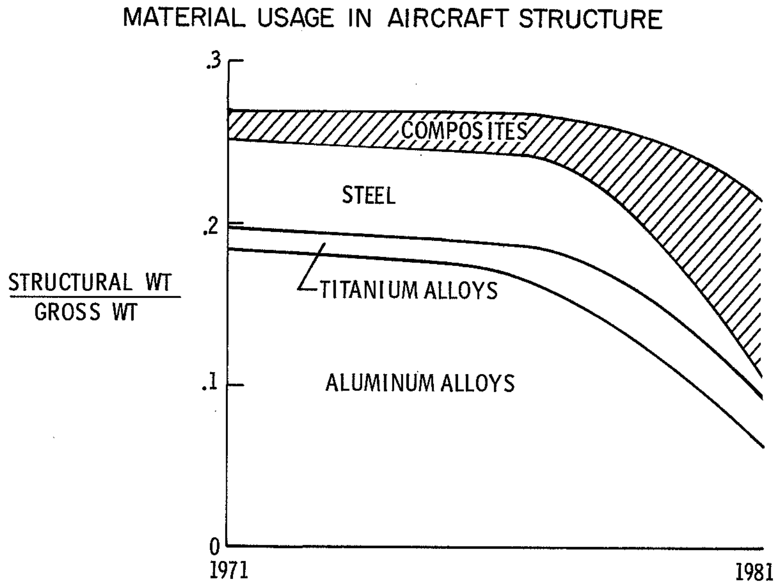

Figure 1

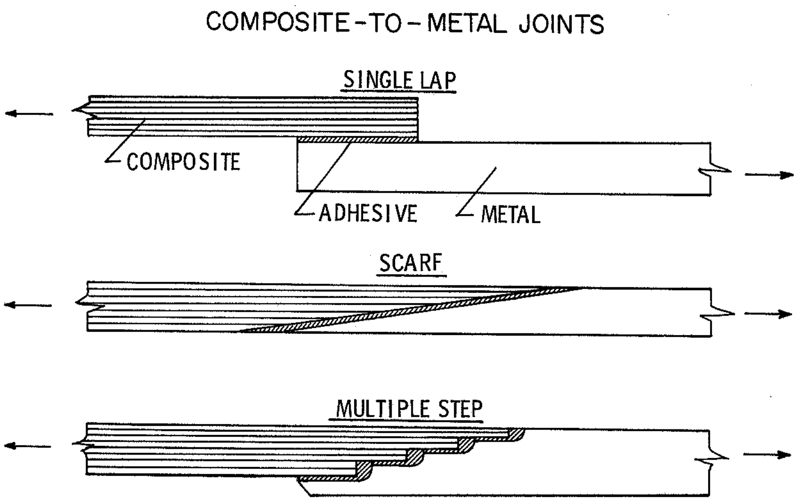

Figure 2

AREAS REQUIRING ADDITIONAL RESEARCH FOR COMPOSITE USAGE IN THE 1970's

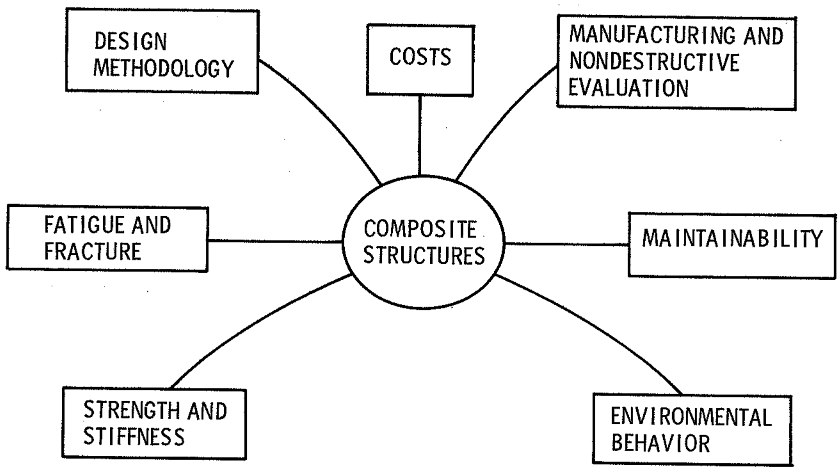

Figure 3

COMPOSITE FUSELAGE PANELS

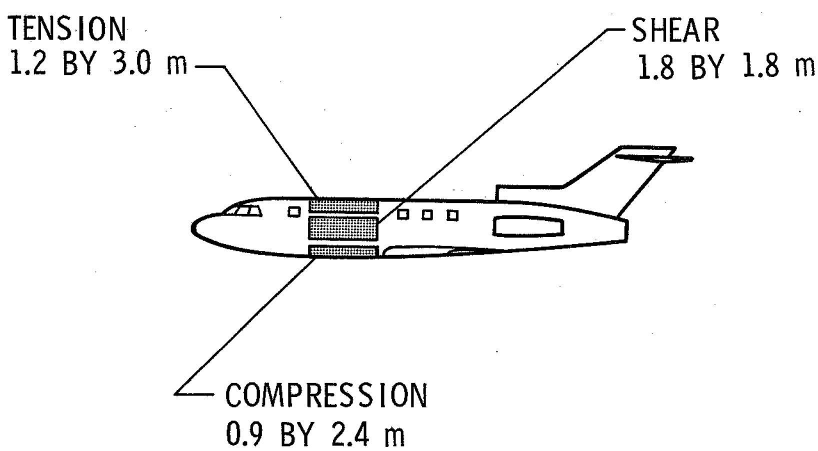

Figure 4

COMPOSITE-REINFORCED FUSELAGE PANEL FOR TENSILE (PRESSURIZATION) TEST

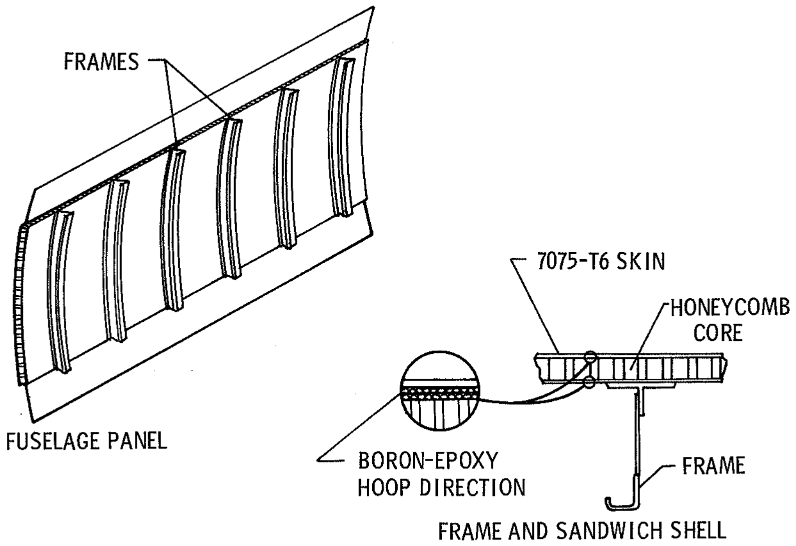

Figure 5

COMPOSITE-REINFORCED FUSELAGE PANEL FOR COMPRESSION TEST

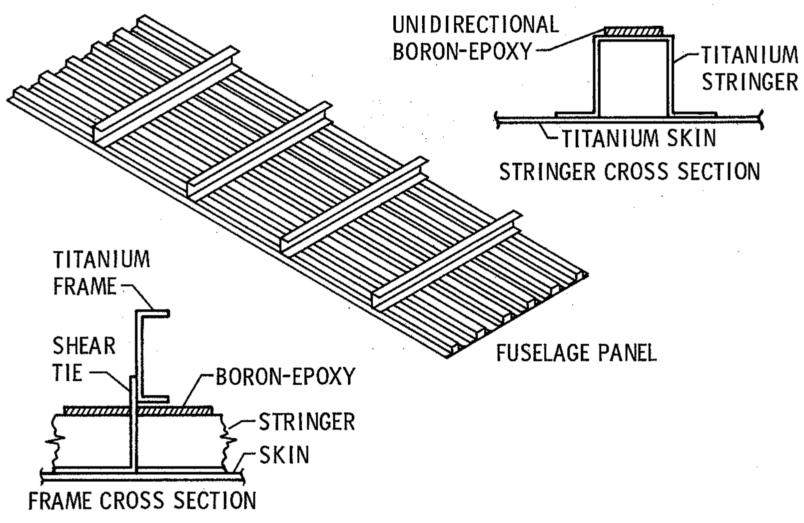

Figure 6

COMPOSITE-REINFORCED WINDOW-BELT PANEL FOR SHEAR TEST

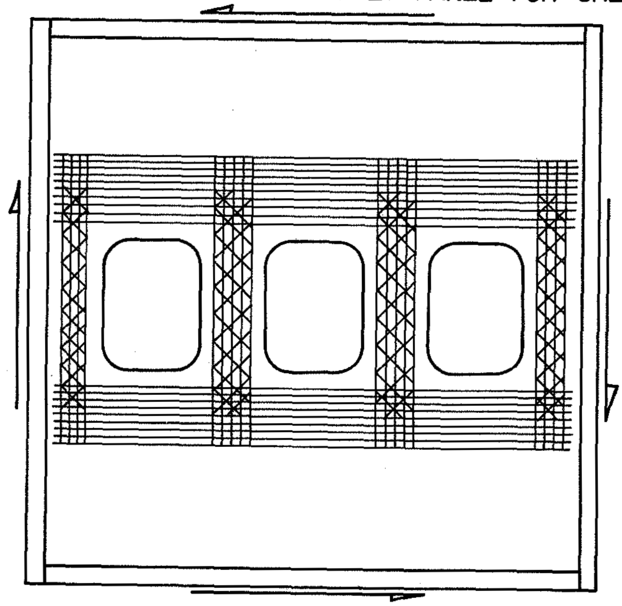

Figure 7

UH-1 TAIL-ROTOR DRIVE SHAFT

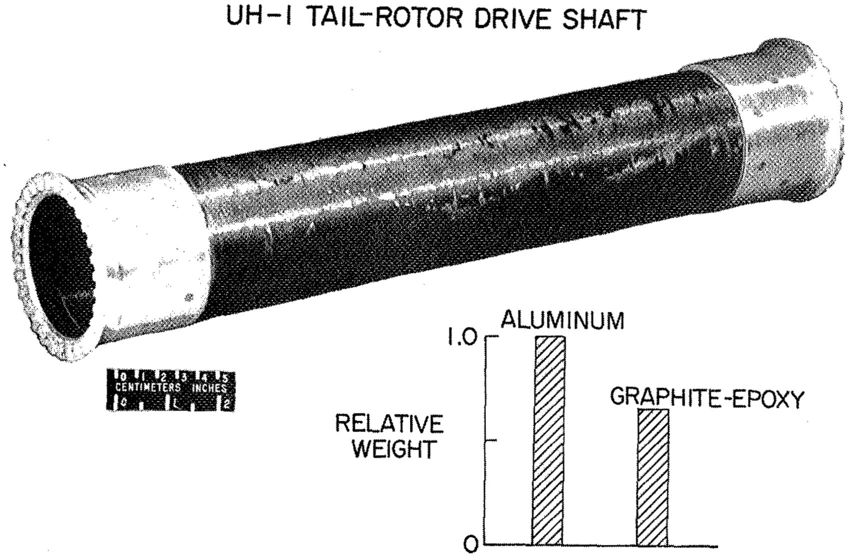

Figure 8

CH-54B HELICOPTER

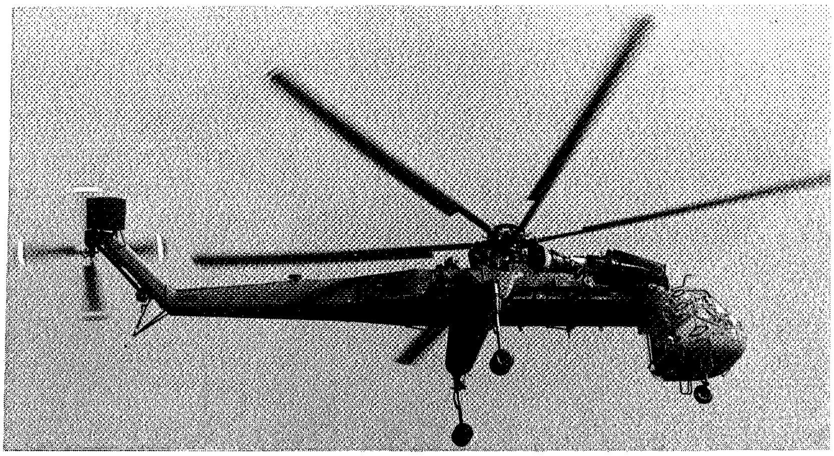

Figure 9

COMPOSITE APPLICATION IN CH-54B HELICOPTER

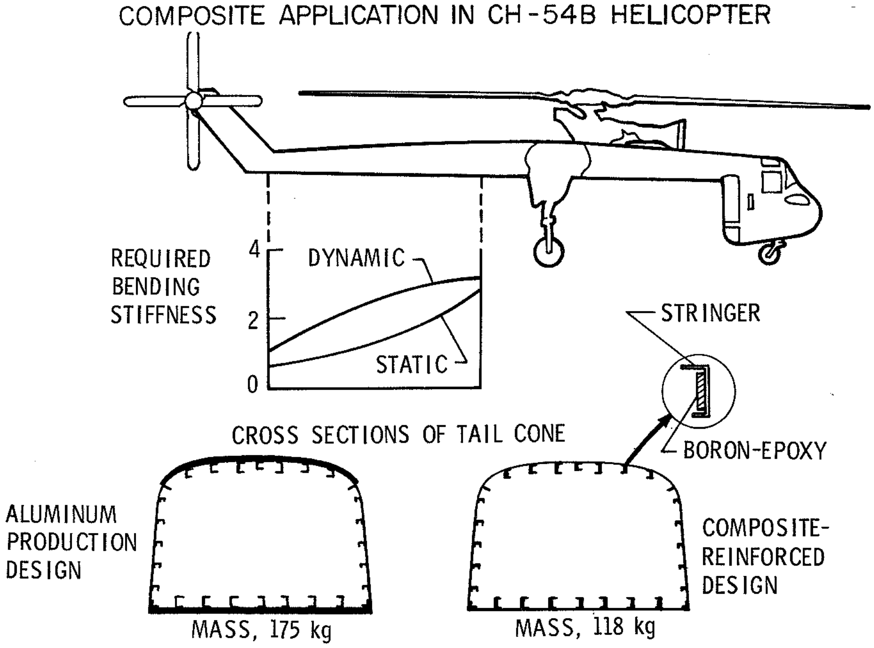

Figure 10

C-130 CENTER WING BOX

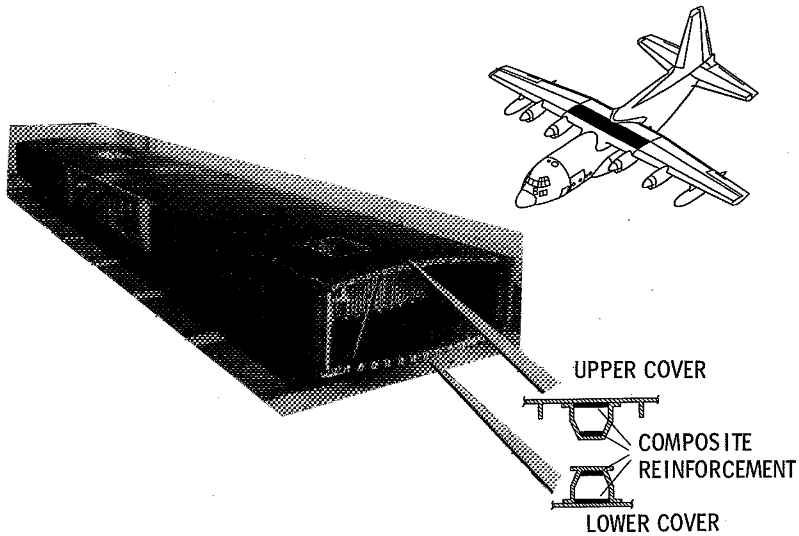

Figure 11

COST-WEIGHT COMPARISON OF COMPOSITE REINFORCED TUBULAR COLUMN

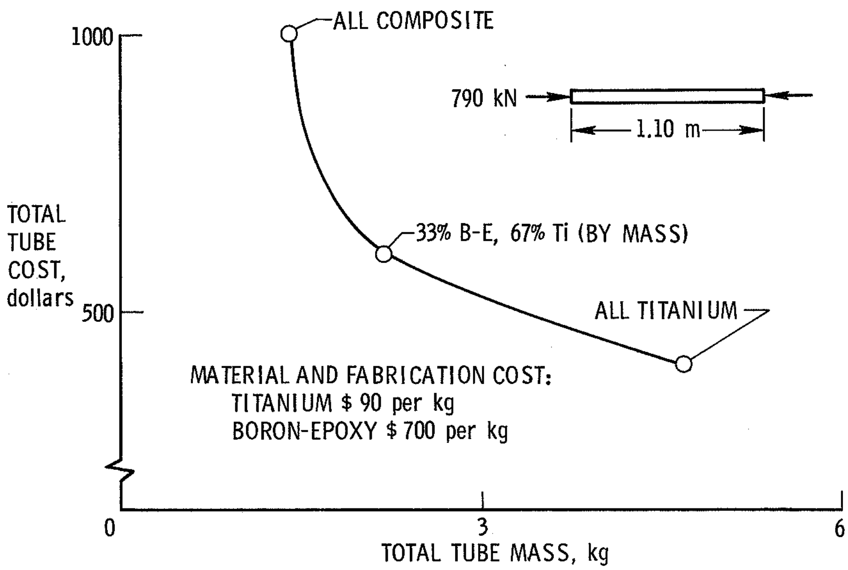

Figure 12

POTENTIAL WEIGHT SAVING FROM USE OF COMPOSITES

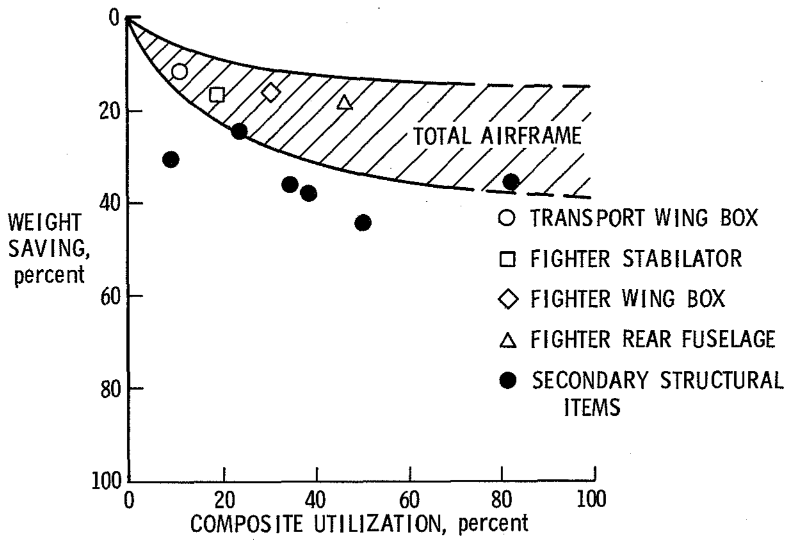

Figure 13

FATIGUE AND FRACTURE

By Herbert F. Hardrath
Langley Research Center

INTRODUCTION

Designing against fatigue and fracture in any vehicle subjected to repeated loadings requires consideration of a host of parameters that interact synergistically. Formal analysis procedures are far from adequate to accomplish the task by calculations alone. Furthermore, considerable scatter in the load experience in a given fleet of vehicles and in the fatigue response to a given set of load experiences, and the extreme complexity of a representative aircraft structure make the analytical task more formidable. Because aerospace vehicles must be made as light as possible for economic and performance reasons, the designer is forced to reduce margins of safety to the minimum.

The following table (ref. 1) shows U.S. civil fleet structural problems (exclusive of landing gear) represented by the number of airworthiness directives (AD's) issued for each type of aircraft:

Fleet	No. of AD's	Fleet	No. of AD's
707	15	Martin 202	5
720	9	Martin 404	2
727	1	A-26	2
BAC I-II	2	DC-3	3
Super 46	1	DC-4	1
F-27	9	DC-6	14
Convair 240	2	DC-7	2
Convair 340, 440	2	DC-8	1
Lodestar	1	Sabreliner	1
Constellation	12	Viscount	12
Electra	14	DH-104	1
Jetstar	4	DH-114	2

This table indicates that practically all aircraft types have experienced some form of fatigue deficiency, which ranges in significance from a maintenance nuisance to an occasional catastrophic incident. Fortunately, civil aircraft are usually constructed to be tolerant to damage, and scheduled air carriers have established very effective inspection and maintenance procedures that keep catastrophic incidents infrequent.

Clearly, an improvement in the state of the art could provide better design tools with which to combat fatigue failures. Conceivably, weight could be saved if one could confidently reduce margins where prudence now dictates conservatism to guarantee structural integrity.

The four major parameters that govern fatigue behavior are choice of materials (metals and composites), configuration, load experience, and environment (thermal and chemical). This paper concentrates on choice of materials, configuration, and load experience. Also included is a description of an NASA Langley Research Center program that should lead to improvements in the state of the art of fatigue analysis so that aircraft may be designed to have maximum efficiency consistent with high reliability.

SYMBOLS

a	crack length
a_c	critical crack length at stress S
a_0	initial crack length
Δa_n	incremental load factor
$(\Delta a_n)_{LLF}$	limit incremental load factor
C	resistance to crack propagation
K	stress intensity
K_T	theoretical stress concentration factor
N	number of cycles
n	crack growth exponent
R	ratio of minimum stress to maximum stress
S	applied stress
ρ	density

CHOICE OF MATERIALS

Metals

In a recently initiated study of how to trade material properties to satisfy a simple set of operating conditions for a specified life, the somewhat involved three-dimensional diagram in figure 1 was developed. Three other figures are needed in sequence to show the derivation of this diagram.

The static strength of three representative materials is shown on a vertical axis in figure 2. The particular materials chosen for the illustration and their pertinent mechanical properties are shown in the following table:

	Aluminum	Titanium	Steel
Alloy	2024-T3	Ti-6Al-4V	D6Ac
Tensile strength, MN/m^2	489	960	1700
Density, ρ, kg/m^3	2770	4440	7890
Fracture toughness, $MN\text{-}m^{-3/2}$	110	110	60
Crack growth resistance	1025	3043	23 000
Crack growth exponent, n	3.64	3.12	2.62

The high-strength steel and 2024-T3 aluminum alloy were chosen because they represent currently used materials with extremes in strength and toughness properties. A titanium alloy was included because of current interest in this class of material. Materials are selected mainly on the basis of their relative static-strength—density ratios. Because of the extreme premium on the weight of structures, during the past several decades, high-strength aluminum and titanium alloys and steels have replaced the modest-strength aluminum alloys.

Figure 3 shows the familiar S-N curve for unnotched specimens made for each of the three materials. Designers have selected materials, at least partly, on the basis of their fatigue behavior in most recent designs. From this point of view and for unnotched specimens, the three materials fall into the same sequence regardless of the life desired. Thus, the trend toward higher strength materials is still justified on this basis.

The left-hand vertical plane in figure 4 shows the reduction in static strength of the same three materials for various crack lengths. In recent years, the extreme sensitivity of high-strength materials to flaws and cracks has been recognized. However, stringent weight constraints force the use of such materials for highly loaded components, such as landing gears and actuators. The drastic reduction in static strength caused by even very short cracks in high-strength materials is, by itself, evidence that extremely sensitive inspection techniques must be employed to avert failures in structural parts made from these materials.

A three-dimensional surface was drawn through the S-N curve and the residual-static-strength curve in figure 4 to obtain figure 1. For clarity, only one surface, that for the 2024-T3 aluminum alloy, is shown. The surface represents the locus of combinations of operating stress (vertical axis) and initial crack length (left axis) that will cause failure in a given number of cycles (right axis). During operation a crack of a given size will grow at a steadily increasing rate with each successive application of a given level of stress. Finally, failure occurs when the residual strength of the part is reduced below the level of the next applied load.

The equation for the surface was developed by integration of the simple crack propagation formulas

$$K = S\sqrt{\pi a} \tag{1}$$

and

$$\frac{da}{dN} = \left(\frac{\Delta K}{C}\right)^n = \left(\frac{\pi^{n/2}\,\Delta S^n a^{n/2}}{C^n}\right) \tag{2}$$

If $R = 0$, $n \neq 1$, and ΔS is constant in amplitude, the integration leads to the relation

$$\pi^{n/2} N \left(\frac{\Delta S}{C}\right)^n a_0^{\frac{n-2}{2}} = \frac{2}{n-2}\left[1 - \left(\frac{a_0}{a_c}\right)^{\frac{n-2}{2}}\right] \tag{3}$$

If $a_0 << a_c$, equation (3) reduces to

$$N\left(\frac{\rho}{C}\right)^n \left(\frac{\Delta S}{\rho}\right)^n a_0^{\frac{n-2}{2}} = \frac{2}{n-2}\left(\frac{1}{\pi}\right)^{n/2} \tag{4}$$

From study of such a surface the following rationale may be developed for inspections: The crack-length axis is related to the required sensitivity of inspection procedures, and the life axis is related to the inspection interval. To an operator, the most desirable designs are those that can survive the longest lives and tolerate significant flaws. Even though this example has been developed only for very simple design cases, it can be used to assess the relative merits of the three materials discussed previously.

Figure 5 shows the portion of each of the three design surfaces that is higher than the other two for a given combination of initial crack length and required life. Thus, a part for the same service may be made lightest out of the particular material whose surface is shown. These surfaces should be recognized to depend critically on the assumptions made for this example. However, they may be regarded as qualitatively appropriate. From this comparison, the high-strength steel is superior for extremely short crack lengths only. Incidentally, the crack length shown includes the diameter of any hole or fastener from which it originates. Thus, for practical cases high-strength steels cannot tolerate significant flaws. The modest-strength aluminum alloy is superior over most practically identifiable crack sizes and inspection intervals. In most situations the relative resistance to fatigue crack propagation is a more significant characteristic of the material than are its static strength and toughness. On the basis of these considerations, the future development of structural materials might provide improved resistance to fatigue crack propagation. Modest reductions in static strength and fracture toughness may be tolerated, if needed.

In figure 6 the foregoing analysis is broken down into four constituents. It is postulated that a doubling of crack-propagation life (or inspection interval) for some design case is desired. The four sketches reveal what might be done in four areas to accomplish this goal. These indications should be regarded as rough rules of thumb.

For a given material and stress level, a doubling of crack-propagation life or inspection interval is available if the inspection sensitivity is improved so that cracks one-half as long as could be found previously can be found with high confidence. For a given inspection sensitivity and stress level, crack-propagation life may be doubled if a material is substituted with twice the resistance to fatigue crack propagation. For a given material and inspection sensitivity, the life may be doubled if the operating stress is reduced by about 20 percent; this last choice, of course, carries with it a 20-percent weight penalty. Conversely, a 20-percent reduction in weight will reduce crack-propagation life by one-half if no other changes are made.

During the next decade improvements in inspection techniques are expected. Cracks in very complex structures must be identified with high confidence. Improvements in material behavior are possible, but emphasis must be placed on improved resistance to the rate of crack propagation rather than on strength-density ratio. This requires reversal of a 40-year-old trend. Unless one of these improvements is made, some reduction in stress level must be expected, with attendant increase in weight, to improve the longevity of structures.

Filament-Reinforced Composites

The data shown in figures 7 and 8 are results of axial-load fatigue tests of unnotched specimens made of boron-epoxy composite and of Ti-6Al-4V titanium alloy. The basic fatigue properties of the two materials are shown in figure 7. The fact that the data for composite specimens lie well above those for unnotched metal has led to the widespread feeling that composite structures will have better fatigue resistance, even at a reduced weight, than present metal structures.

Boron-epoxy composite was bonded to stepped titanium alloy end fittings much as would be done for practical structural parts (fig. 8). The lower curve is the S-N curve for titanium alloy specimens containing a modest stress raiser tested at the same stress-density ratios. The advantage of the composite material is still clear but has been reduced significantly. A further reduction may be expected when the more practical assortment of cross plies is introduced instead of the uniaxial fibers present in this investigation. The modes of failure observed in these tests included a progressive debonding of the composite and cracking at one of the steps in the metal end fittings.

Research is underway to develop better, cheaper, and more reliable joints in structures made of composite materials. Such research deserves and will receive high priority to develop the full potential of these materials and to make their introduction into primary structure viable in the next decade. As indicated in paper no. 10 by Richard A. Pride, this need and building confidence in the long-time resistance to the operating environment are the major hurdles facing the adoption of composite materials.

EFFECT OF CONFIGURATION

The following illustrates qualitatively how various types of structures may be expected to behave in fatigue:

Feature	Advantage	Disadvantage
Monolithic	Few stress raisers Long life	Poor damage tolerance Poor maintainability
Skin-stringer	Control crack propagation	Many stress raisers
Planks	Fail safe	Poor control of crack propagation
Composites	Good fatigue resistance if axially loaded and no joints	Needs development High cost

Generally, each of the three types of metal construction has advantages and disadvantages; therefore, no distinct advantage is held by any one type. Plank construction has current appeal because it is a compromise between the older skin-stringer construction and the not-yet-feasible monolithic. In the author's view all three types of consturction will be in common use in the foreseeable future. Whenever possible, damage tolerance should be incorporated in structures as a distinct safety advantage. As discussed previously, the composites show promise for better behavior but require much development.

The relative merits of various fastener schemes are cited in general terms as follows:

Feature	Advantage	Disadvantage
Rivets	Easy manufacture Easy maintenance	Variable quality
Interference fasteners	Longer life	Higher cost Stress corrosion
Adhesive	Fewer stress raisers	Reliability degrades after long usage
Weld	Good repairability	Weld defects Unusable on many materials
Diffusion bond	Get sheet properties in bulky sections Fewer stress raisers	Needs development High cost
Weld bond	Easy manufacture Longer life	Needs development

As is well known, most fatigue difficulties arise because of the high stresses, fretting actions, and other complications introduced at fasteners. Here again, each system has advantages and disadvantages, with no distinct superiority for any one type. Fastening schemes may be expected to be improved during the decade. However, if a really superior fastener is developed so that, in principle, a weight saving could result, the previous discussion of inspection sensitivity and intervals must be considered. Any significant increase in stress carries with it a danger of more rapid crack propagation.

EFFECT OF LOADING

Generally, the loadings experienced by civil aircraft are fairly well understood and cataloged. However, figure 9 shows the cumulative frequency of exceeding a given load factor per nautical mile of flight for general-aviation aircraft in three kinds of service (ref. 2). The load factor is normalized by the limit load factor used in design. The extreme curves in the figure indicate that an aircraft used for crop dusting may experience 1000 times as many exceedances per nautical mile of a given load factor as are experienced by a personal aircraft. Considering that the same type of aircraft might be used in either service, the designer is confronted by a serious dilemma. Assuming all operators wish to fly a given number of miles, even the average experience provides only 3 percent of what the crop-duster operator may desire, whereas the personal owner pays for 30 times the life he may need! Purchasers of second-hand aircraft are afforded little useful guidance by consulting the log books that record only hours flown when aircraft are flown in various ways.

Langley Research Center has recently developed the instrument shown in figure 10 (ref. 3). The device receives signals from an ordinary strain gage attached to some critical spot on an aircraft structure. A comparator circuit senses when the strain experienced exceeds any of four preselected values, and an amplifier provides an electrical pulse to actuate the appropriate counter in the instrument. This unit weighs less than 0.9 kg and is capable of further miniaturization. Several such instruments are being flown in operational aircraft to evaluate their performance in service.

Such devices might be installed on aircraft that experience intense or frequent loadings to guide inspections or retirement of aircraft before a dangerous situation develops. Conceivably, as the understanding of fatigue increases, circuitry that would integrate the fatigue experience with proper weighting factors could be developed so that only one counter would be required. Its indication would provide a most useful guide for inspections. The present instrument contains approximately $1000 worth of parts when purchased in small lots. If such devices were mass produced, the manufactured price might be much lower.

A 10-YEAR PLAN

NASA Langley Research Center has recently started a systematic program to develop fatigue technology and to organize advances so that, in principle, the fatigue design process could be computerized during the next decade. Figure 11 shows how the basic needs can be identified and the necessary facets of the design problem can be fitted together. The four layers of the cube identify the four major parameters already discussed. Two main design philosophies, safe life and damage tolerant, are currently practiced to varying degrees in modern design and are depicted by the dual stacks of "bricks" in the diagram. The third axis of the cube is divided into three segments for the three obvious stages of a fatigue failure.

In the safe-life design philosophy, the three stages of the fatigue phenomenon are seldom, if ever, considered formally. Because of its simplicity, this type of analysis is expected to be continued for the foreseeable future; it is the method by which the total expected life of a structure is analyzed.

On the other hand, the damage-tolerant design philosophy has no formal way to determine how or when a crack might initiate. Quite frequently, a flaw is assumed to be introduced during manufacture. Remaining life is then established by deliberate analysis of the rate of crack propagation as a function of the four major parameters. Similarly, fracture is anticipated when residual strength is reduced below that required to resist expected or specified loads. This philosophy is the basis for certifying airworthiness of essentially all current civil transport aircraft.

From this simple picture, the design process can be broken down into manageable research challenges. Early stages of the Langley Research Center plan are expected to gather empirical data on fatigue, crack-propagation, and fracture properties of structural materials of current interest. Analytical expressions will be fitted to these data to provide the core of information that is required for design. Other phases of the Langley Research program will be devoted to developing better procedures for treating the effects of configuration, of complex load histories, and of environmental factors. Rational procedures will be favored because they will usually be more readily acceptable, but empirical procedures will be evaluated and recommended, as needed, to provide the overall design tool that is desired.

If the effort is successful, the vast amount of ad hoc testing that is a feature of any major design venture is expected to be reduced substantially. By soliciting and stimulating cooperation from other government, university, private, and industrial research and development organizations, a significant improvement in the state of the art may be accomplished with an investment that is only slightly higher than is now made to combat fatigue.

CONCLUDING REMARKS

The next decade of development in civil aviation should see improvements in structural integrity. Hopefully, these improvements can be made with minimum weight penalty. For fatigue-critical parts of structures, metals should be developed to have better resistance to fatigue crack propagation or selected on the basis of this characteristic rather than on static properties. Composite materials will be developed to exploit the improved performance suggested by current knowledge.

Damage-tolerant designs will and should be employed whenever practical to provide good safety. Improved fatigue-load monitoring devices should be employed, particularly in vehicles subjected to wide variations in load experience.

NASA Langley Research Center has started a 10-year plan of systematic research that should enhance the design of safe and efficient aircraft structures. The beneficiary will be the aircraft operator, who should become surer of the structural integrity of his fleet.

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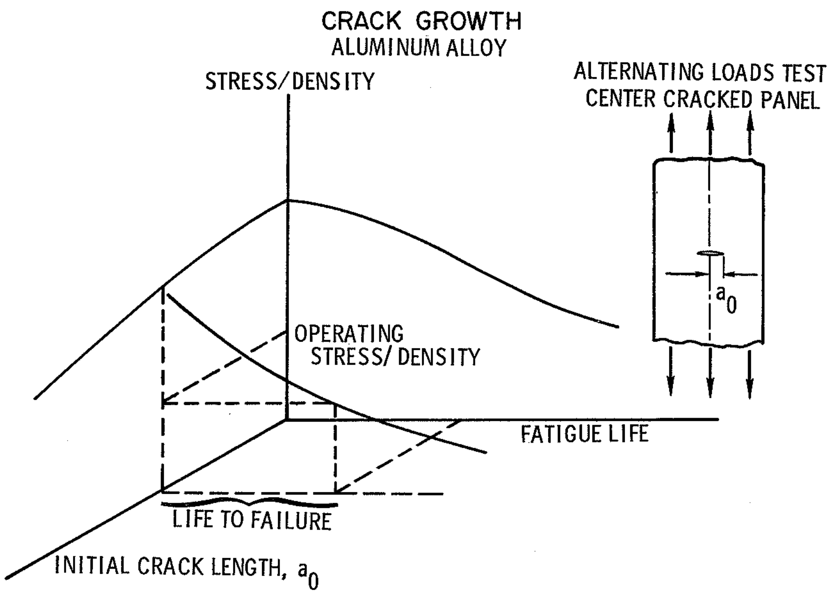

Figure 1

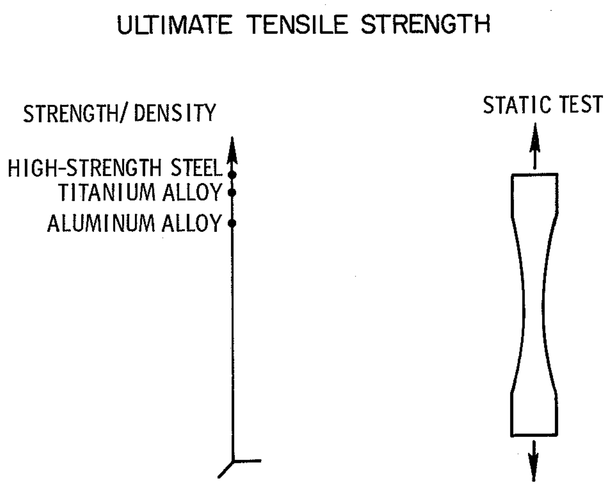

Figure 2

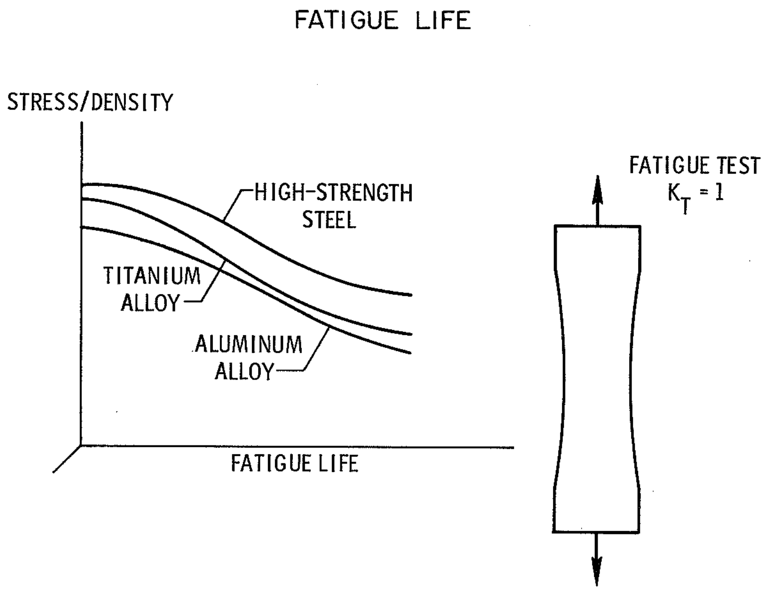

Figure 3

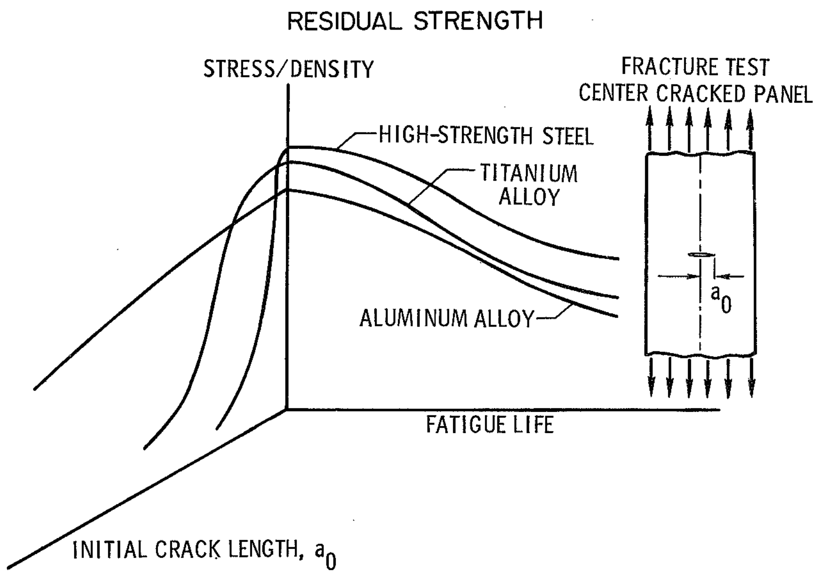

Figure 4

BEST CHOICE OF MATERIAL
FOR SPECIFIED LIFE AND INITIAL CRACK

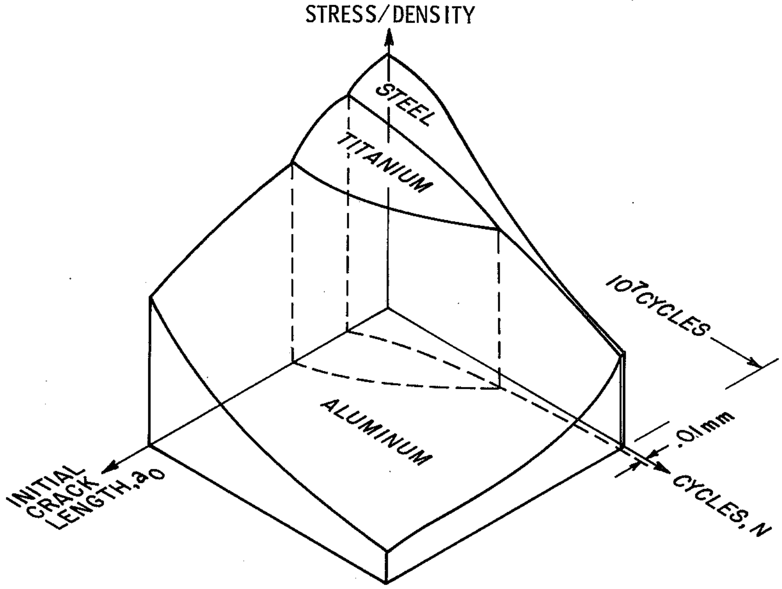

Figure 5

OPTIONS FOR DOUBLING CRACK PROPAGATION LIFE

(LOGARITHMIC SCALES)

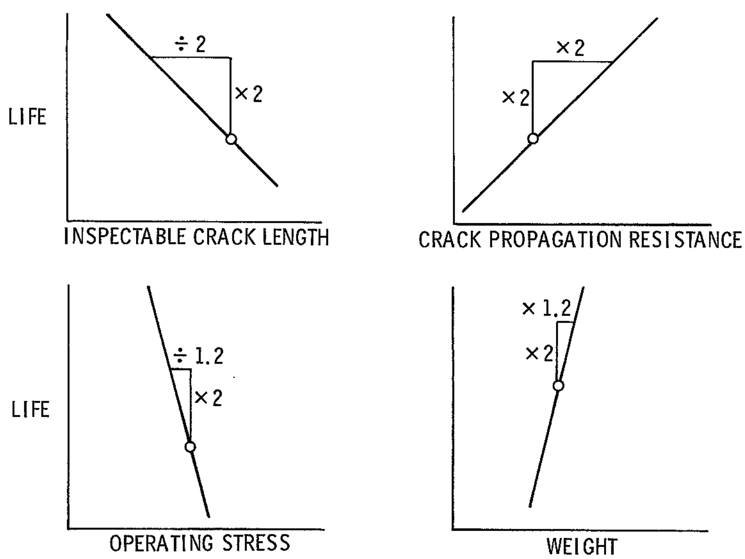

Figure 6

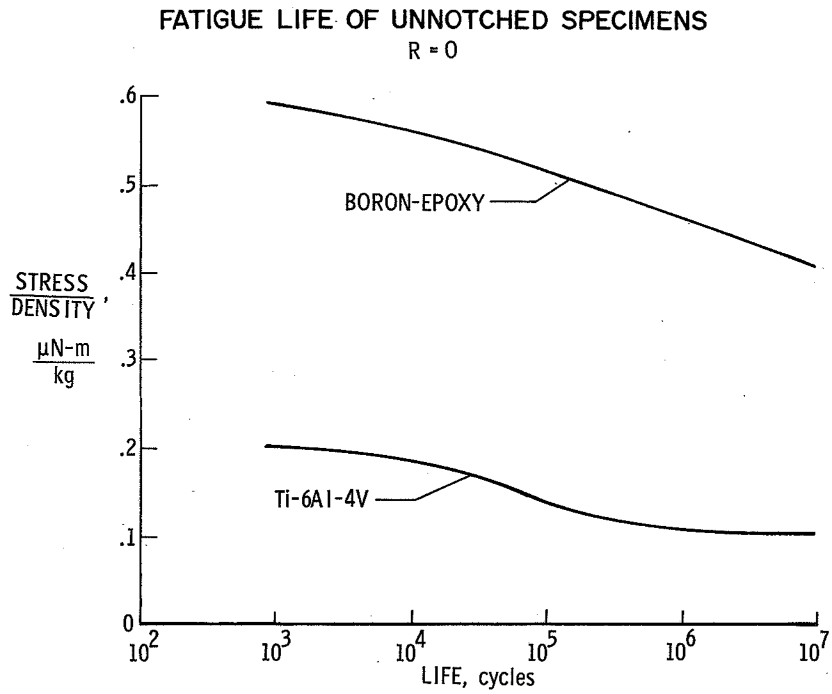

Figure 7

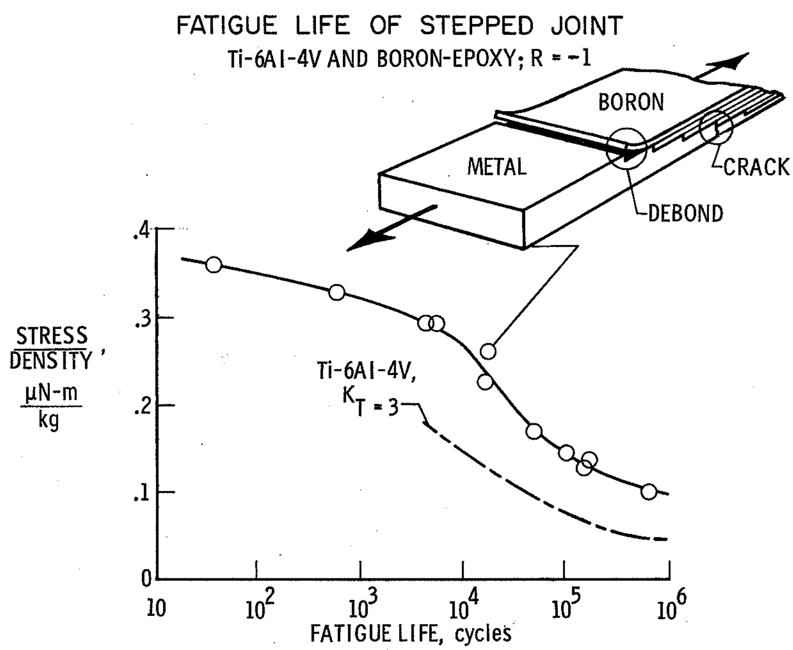

Figure 8

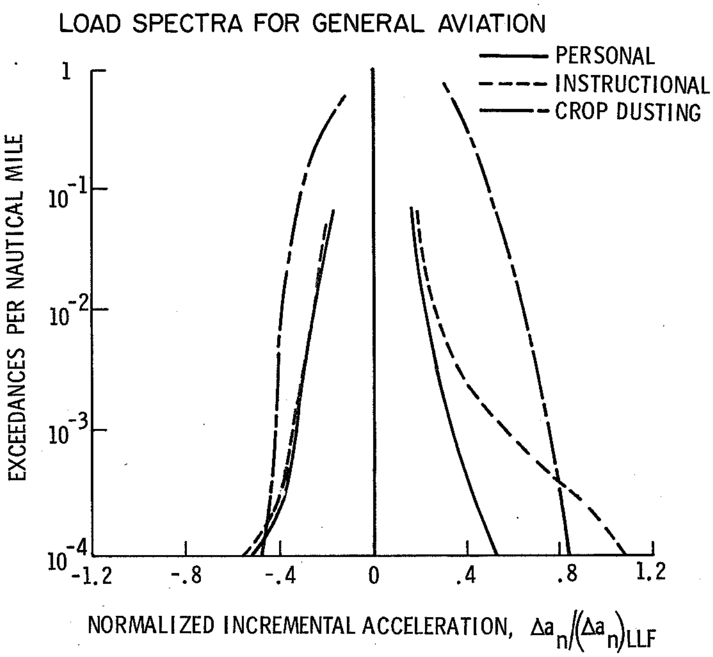

Figure 9

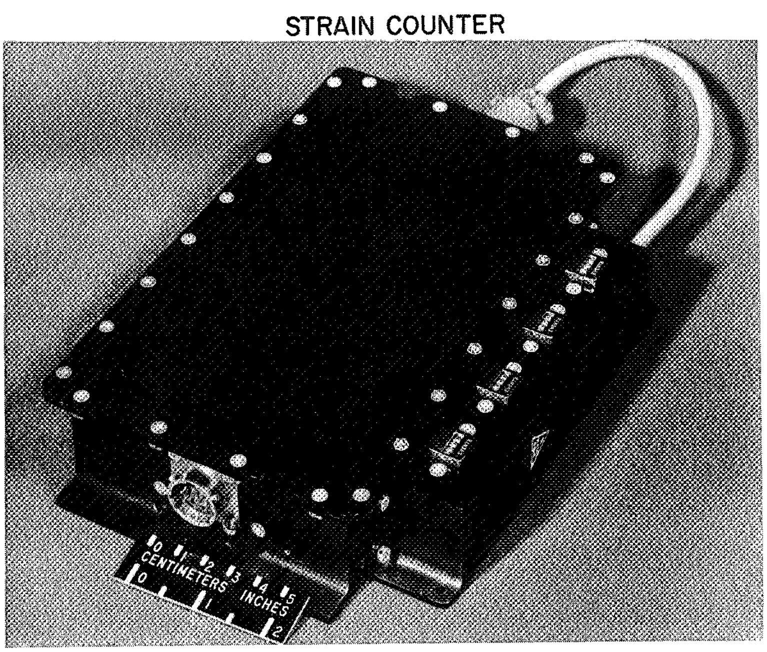

Figure 10

FATIGUE TECHNOLOGY

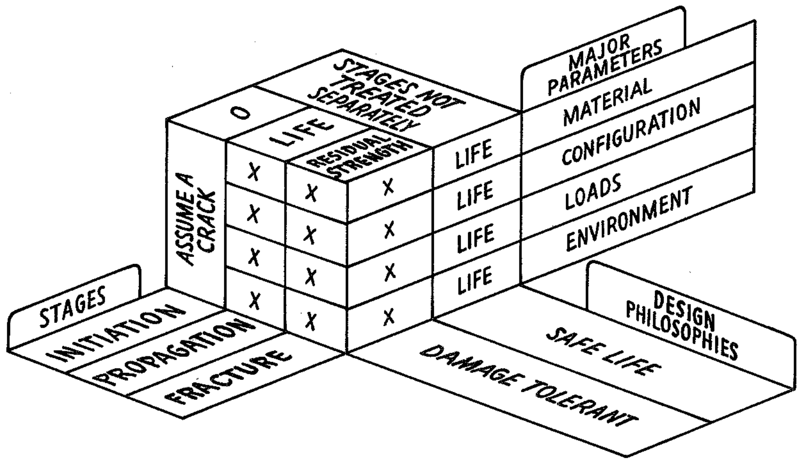

Figure 11

AUTOMATED DESIGN METHODS IN STRUCTURAL TECHNOLOGY

By Harvey G. McComb, Jr.
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INTRODUCTION

The structural design process for a typical aeronautical vehicle is very complex and often requires the largest "disciplinary" group on the vehicle design team. Because of the multitude of details to be examined, critical problems may not be revealed until late in the design process. Consequently, there is a great deal of interest among structural engineers in the potential for automating as much of this process as possible in the hope of reducing design time and costly errors. Even greater benefits should accrue if the entire design process (not just the structural design) were automated so that structural consequences of airplane configuration changes could be evaluated more accurately and rapidly in the early design phases. This paper discusses the impact of automation in three areas: structural analysis, structural design, and integrated design. The term "structural analysis" simply refers to the determination of stresses or internal loads and deflections in a given structure under given loading conditions (fig. 1(a)). Structural design, on the other hand, involves incorporation of the analysis into a resizing and reanalysis procedure in which structural allowables are taken into account and the process is cycled to a converged design (fig. 1(b)). Finally, the term "integrated design" is used to refer to interdisciplinary approaches where modules from several disciplines are integrated into unified systems for analysis and design (fig. 2).

STRUCTURAL ANALYSIS

Structural analysis is the core of the structural design process, and a great deal of progress has been made in the automation or computerization of this discipline in the past 15 years. The finite-element method illustrated in figure 3 is the primary analysis tool in use today. In this method an idealized model of a complex structure is conceived by using simple structural elements such as beams and plates. The computer can be programed to "assemble" these elements, satisfy equilibrium and continuity conditions, and calculate the stresses or internal loads and deflections under prescribed loading conditions. The properties of a wide variety of structural elements have been cataloged in the literature for use in constructing appropriate idealizations. The finite-element method is a highly flexible tool. For example, after completion of the analysis of a rather coarse model of the structure, the internal loads calculated for this model can be applied to a detailed model of a cutout or some other complexity as illustrated in figure 3 and detailed stress distributions can be obtained.

An example of a large general-purpose finite-element computer program is the NASTRAN system characterized in figure 4 and described in references 1 to 3. NASTRAN was developed under NASA sponsorship over the last five or six years. It is well documented and is available in the United States from the Computer Software Management Information Center (COSMIC), University of Georgia. NASTRAN is a unified program system capable of performing analyses of static stresses and deflections, buckling, vibrations, and transient response of complex structures. Once the structural model and the type of analysis desired is input to the computer, the NASTRAN executive routine automatically assembles the appropriate analysis module and the proper information from the data base to perform the desired analysis. The NASTRAN system contains plotting routines which are very useful for checking input data and for displaying output in readily understandable form. It is operational on three computers, the IBM 360, the CDC 6000 series, and the UNIVAC 1108, and was developed expressly to be easy to learn and use by structural engineers.

There are many finite-element computer programs in existence. Aerospace companies have their own which may have some of the same features as NASTRAN but are tailored to the company's individual requirements. A representative list of finite-element programs together with the company or agency responsible for their development is contained in table 1 (taken from ref. 4). This list is not exhaustive and the programs included have varying capabilities, but nearly all are routinely applied in aerospace structural design. The list gives an idea of the extensive existence and use of these tools. In spite of this highly computerized analysis capability, however, there has been a startling rise in the manpower resources required to design an airframe.

The resources required to accomplish the structural design task alone for large transport aircraft are illustrated in figure 5. This figure was based on information supplied by Roy Eudaily, Lockheed-Georgia Company (under NASA Contract NAS1-10701). The solid line shows man-hours required to design a unit mass of structure and the dashed line indicates computer hours per man-hour. It is clear, of course, that the highly computerized structural analysis tools have enabled designers to accomplish refinements and thoroughness in analysis that would simply be impossible without such tools. Nevertheless, the rate of increase in manpower required over the years is surprisingly high. It should be possible by proper automation of more of the design process to halt the increase and perhaps reduce manpower requirements in the next decade, as suggested by the dotted extension in figure 5.

STRUCTURAL DESIGN

Automation of the structural design process, rather than simply the structural analysis task, is characterized in figure 6. A schematic two-dimensional design-variable space is shown, where the term "design variable" (DV) simply refers to the sizes of structural elements, such as stiffener areas and skin thicknesses. On such a chart contours of structural weight and curves which represent constraints on the structure can be plotted. Two failure constraints are shown in figure 6. It is possible to program a computer to search systematically and automatically for the lightest weight structure, for example, which does not violate any of the constraints. Analysis routines must be available to define the constraint conditions and indicate when one of them has been violated in the search. Such a search is represented schematically by the arrows in figure 6.

Development and study of these search routines or algorithms has come to be termed "mathematical programing," and a general discussion of applications to structural problems is contained in reference 5. Many search algorithms are available, and it is not necessary to plot a substantial portion of the design-variable space, as is indicated in figure 6, to perform the search. In fact, for most practical design problems, there are many design variables so that the design-variable space is a hyperspace of many dimensions and impossible to show on a chart. To search through such a space with complete generality in design variables usually involves excessive computer storage and time. For practical problems, therefore, some artificial criteria are established which restrict the search to a subspace but still lead to an approximate optimum design. An example of this approach is the design of a stabilizer structure illustrated in figures 7 and 8.

Figure 7 shows an all-movable horizontal stabilizer for a supersonic aircraft. The stabilizer is pivoted at its root on a spindle which projects from the aircraft fuselage. The primary structure (without leading and trailing edges and tip) was designed by Grumman with the "fully stressed design" technique. (See ref. 6.) The structure is full-depth honeycomb with titanium cover skins. "Fully stressed design" means that elements in the structure are sized to carry a specified allowable stress under at least one of the loading conditions considered or are at their specified minimum sizes. The design was carried out automatically by cycles of analysis and resizing with the use of a finite-element model of the structure indicated in figure 8. The upper and lower covers were divided into 162 panels shown in the plan view in figure 8, but the total structural model included 890 elements and 1171 degrees of freedom. Eight loading conditions were considered, and the plot at right in figure 8 shows the skin thickness at section A-A after the first redesign cycle and after the ninth cycle. The curve at bottom (fig. 8) shows how the total structural mass is reduced with each redesign cycle. A total central processing unit (CPU) time of 46 minutes was required on the IBM 360/75 computer for the complete calculation including all nine redesign cycles.

Substantially larger structural models have been handled by the fully stressed design technique. An idealization of a space shuttle orbiter structure designed by McDonnell Douglas with a computer program based on this technique is shown in figure 9 (from ref. 7). The left half of the model (cut down the longitudinal center line) is shown in figure 9. It is a fairly refined model as indicated by the data in the figure, and it was sized for eight loading conditions with one circuit of analysis, redesign, and analysis requiring about 22 minutes central processing time on an IBM 360/85 computer.

INTEGRATED DESIGN

Efforts already have been made to integrate several disciplines into unified computer programs for analysis and design. There are a number of programs in existence which automatically size and configure an aircraft, primarily on the basis of empirical or statistical inputs from the various disciplines, especially from structural weights. Descriptions of some of these systems can be found in references 8 to 11. These programs can be very effective if a good statistical base is available, that is, if the vehicle being designed is not too different from existing vehicles.

Recent efforts have been aimed at incorporating more analysis into integrated systems to overcome the limitations inherent in the empirical approach. Some examples of these programs and the agencies responsible for their development are

ATLAS	The Boeing Company
IDEAS	Grumman Aerospace Company
DAWNS	NASA Langley Research Center
SAVES	NASA Langley Research Center
SWIFT	NASA Langley Research Center
IPAD	NASA Langley Research Center

The Boeing ATLAS system (ref. 12) integrates several disciplines, such as geometry, aerodynamics, loads, structures, and weights, and includes an executive routine which controls the sequence of analysis. The Grumman IDEAS system (ref. 13) does not have an executive routine, but includes a more extensive stable of programs (about 50) with carefully integrated input and output features so that results from one program can be immediately input to another. IDEAS is intended to incorporate all the programs necessary to generate external loads and perform the final calculation of the internal loads distribution for a complete aircraft. Grumman engineers credit the IDEAS system with shortening the final external-loads and structural analyses on the F-14 airplane sufficiently to make sound structural data available early enough in the design process to support necessary changes in design during the more creative stages (ref. 14). This work was accomplished within the 23 months between the contract go-ahead and the first flight – considered by Grumman to be an unusually short time for an aircraft of its type.

ATLAS has been exercised by Boeing engineers on a model of the U.S. supersonic transport (SST) prototype illustrated in figure 10 (from ref. 12). This model had earlier been analyzed by conventional methods which included many stages performed by hand. This was a model of the complete airplane and included the structural characteristics shown in the upper left in figure 10. The structural finite elements and the aerodynamic-lifting-surface panels do not coincide, and appropriate interpolation routines are included in ATLAS to make the layouts compatible.

The manpower and project flow time required to do the same analysis task to the same depth and accuracy on the SST model with the integrated program and by conventional means are shown in figure 11 (taken from ref. 12). The various markings on each bar are keyed to different parts of the analysis. On the flow-time chart the bars are overlapped because some of these tasks can be carried out simultaneously. Both the manpower and the flow time were cut in half by use of the integrated program. The tasks shown are only a small part of the total design process. Extensive use of integrated methods would result in substantially larger savings.

ATLAS and IDEAS are focused primarily on the analysis task, although they are integrated systems. The next three programs listed are not of industrial scale, but they are initial attempts to integrate disciplines into truly automatic design programs. DAWNS is described in reference 15 and represents integration of steady lifting-surface aerodynamics and a finite-element structural-analysis routine to perform automatically the fully stressed design of aircraft wing structures. SAVES, a program under development, is based on the DAWNS philosophy but extended to handle fuselage structures and fuselage-wing combinations. SWIFT (ref. 16) combines a very simple unsteady aerodynamic theory (piston theory) with a simple plate theory for the structural representation to perform a combined strength and flutter design of wing structures with the use of a battery of mathematical programing algorithms. The final program listed, IPAD, is discussed later on.

The DAWNS (Design of Aircraft Wing Structures) program has been exercised recently to design a series of delta wings for a hypersonic cruise vehicle. Pertinent characteristics of the wings are illustrated in figure 12. The wings had a double-wedge airfoil shape with a flat lower surface, a linear spanwise thickness variation to zero at the wing tip, and maximum thickness occurring at about two-thirds chord length aft of the leading edge. The aerodynamic planform and the plan view of the structural finite-element layout are shown in figure 12 along with a chordwise section illustrating the airfoil shape. The structural model contained a total of about 140 elements and 180 degrees of freedom. Calculations were made for a single loading case of a 2.0g pull-out maneuver at Mach 3 for three leading-edge sweep angles and three values of thickness-to-chord ratio. The aerodynamic-planform area (646 m^2 (6954 ft^2)) and the aircraft gross weight (236 000 kg (520 000 lb)) were maintained constant in all the calculations.

Fully stressed design results for wing structural mass for a range of leading-edge sweep angles and maximum thickness-to-chord ratios are shown in figure 13, where the mass is normalized to the case for the leading-edge sweep $\Lambda = 65^{\circ}$ and the maximum thickness-to-chord ratio $\frac{t}{c} = 0.05$. For these calculations it was assumed that the structure was cooled or protected so that it operates at 590 K (600° F). Thermal-stress effects were neglected since DAWNS does not contain thermal-stress capability. The cover panels were assumed to be honeycomb-sandwich with titanium skins, and the ribs and spars were assumed to have corrugated titanium shear webs. Minimum-gage limitations are included in DAWNS, and these considerations are very important for the higher sweep angles and thickness ratios. No "nonoptimum factors" are included in these results; therefore, they indicate trends for idealized "bare-bones" structure required to carry the loads. These calculations required a total of 45 minutes central processing time on a CDC 6600 computer for the nine cases. Results of this type are important structural inputs to vehicle-system trade-off studies. The DAWNS program includes a cathode ray tube (CRT) interactive graphics option. A photograph of the display console showing four wing designs for a fighter aircraft is shown in figure 14.

One approach for the organization of a large integrated system, named Integrated Programs for Aerospace-Vehicle Design (IPAD), is illustrated in figure 15. A manager program would supervise the activities of operational modules in the various disciplines which impact the design of an aerospace vehicle system. This manager program would control the strategy of calculations and manage the vast transfer of data by interfacing with a data base. The data base would be continuously updated so that all technical disciplines would have access to the latest design information. The refinement of the analysis idealizations would be kept balanced at each stage of the design process. Optimization could be carried out as loops within each operational module or by activating an optimization procedure contained within the manager program. The man-machine interface represents graphic displays and interactive graphics options. These features are very important in a system such as this because there will be significant portions of the design process which cannot be automated. Ways are needed for the engineers to monitor the design, introduce new ideas, and make design decisions which are beyond the capability of a computerized system.

Desirable characteristics of an IPAD system are

Open-ended: Capable of accepting operational modules with
- Preexisting codes
- Proprietary codes
- New technology developments

Designers have complete freedom to use company data and theory

Depth of analysis varies to match design stage

Designer can monitor progress of design

Designer can intervene to change
- Aerodynamic configuration
- Mission requirements
- Propulsion system
- Economic and production factors
- Structural layout and concepts
- Material selections

At this time, whether or not all these features are possible is not known, and studies should be undertaken to determine the feasibility of such a system. The system should be very flexible and open-ended. It should allow any of the disciplinary operational modules to be pulled out and replaced with one of the designer's choice. It should accept preexisting codes so that it can draw upon the many analysis programs in existence today. It should accept proprietary codes in such a way that the proprietary aspects of the codes would be protected. It should accept codes developed entirely independently of the IPAD system, and it should allow the designer to introduce new technology developments easily. It should be built in such a way that the designer has complete freedom to incorporate company data and theory into the IPAD system. It should be capable of performing analysis and design to whatever depth is suitable for the current stage of the design. The designer should be able to monitor the design and intervene to make such changes as are indicated at the bottom of the list of characteristics. The successful development of an IPAD system should provide an immensely useful tool for industry in generating future designs of a wide variety of aerospace vehicles and for government agencies in evaluating designs and conducting design studies germane to future national programs.

By 1985 certain benefits should accrue from the development and use of integrated aerospace vehicle design systems. In the first place, it should be possible to perform the required technology tasks to an adequate depth early in the design process to prevent many later design changes which escalate costs. In this way integrated systems should reduce costs of future large national programs. Based on experience to date, integrated systems should be able to reduce substantially the manpower required for design of a unit mass of structure (perhaps by 50 percent) and reduce project flow time by an order of magnitude. This capability can also be translated into overall cost savings. Finally, there are structural weight savings which can be attributed to improved design precision. Although these benefits are very difficult to estimate and are substantially smaller in magnitude than the previously mentioned benefits, it is believed that structural weight savings on the order of 10 percent may be possible.

CONCLUDING REMARKS

A rough summary of the state of design automation from the structures viewpoint is shown in figure 16. So far as analysis is concerned, very good idealizations of complete vehicles can be handled today. Aircraft such as the U.S. SST prototype, the McDonnell Douglas DC-10, and the Lockheed L-1011 have been analyzed in great detail by use of finite-element computer programs. Advances in analysis will come in the introduction of nonlinear effects, such as plasticity and large deflections. Also, very badly needed are techniques for discriminating the very detailed stress distributions about cracks and fasteners for fatigue and fracture calculations. In automated structural design, rather large components of prescribed external shape and prescribed loading are being handled with automatic resizing of the structural elements. By the late 1970's complete vehicles will be handled to the same refinement that is now being done with analysis alone. An important advance needed in this area is the incorporation of thermal-stress considerations in automated structural design procedures. In the area of integrated design, not just integrated analysis, the technology today is perhaps about equivalent to what it was for analysis alone 20 to 25 years ago. A thorough integrated design job will probably be possible on refined idealizations of complete vehicles sometime in the 1980's.

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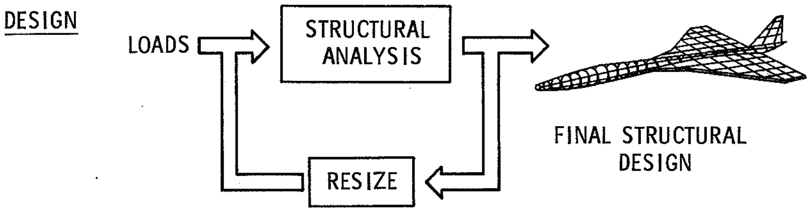
TABLE 1.- REPRESENTATIVE GENERAL-PURPOSE FINITE-ELEMENT STRUCTURAL-ANALYSIS COMPUTER PROGRAMS IN CURRENT USE IN THE AEROSPACE FIELD

Program name	Agency and/or industry responsible for development
ELAS	NASA-JPL
SAMIS	NASA-JPL/Philco-Ford
NASTRAN	NASA/Computer Sciences Corporation
FORMAT	AFFDL/McDonnell Douglas
MAGIC	AFFDL/Bell Aerosystems
ASTRA	Boeing
SAMECS, COSMOS	Boeing
REXBAT	Lockheed
FAMAS	Lockheed
SNAP	Lockheed
MESA	North American Rockwell
NARSAP	North American Rockwell
CLASP	North American Rockwell
STARDYNE	CDC/Mechanics Research Inc.
EASE	Engineering Analysis Corporation
ASTRAL	Grumman
BELL	Bell Aerosystems
STRESS	MIT
ASKA	North American Rockwell/Univ. of Stuttgart
ICES	MIT
MARC2	Brown University

(a) Analysis.



(b) Design.

Figure 1.- Automated structural methods.

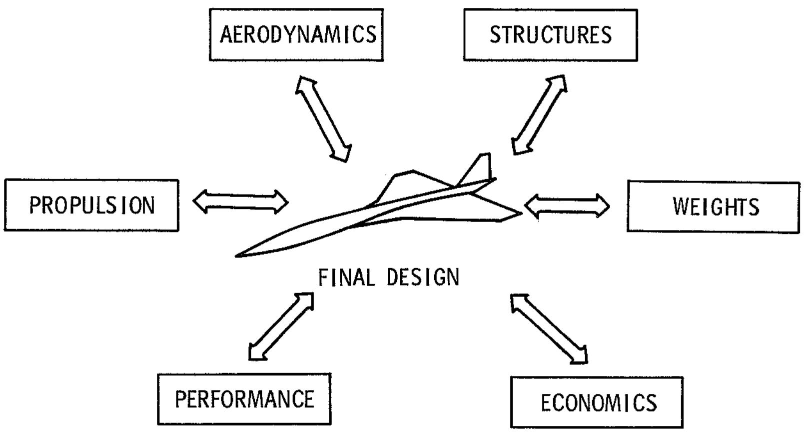


Figure 2.- Integrated design.

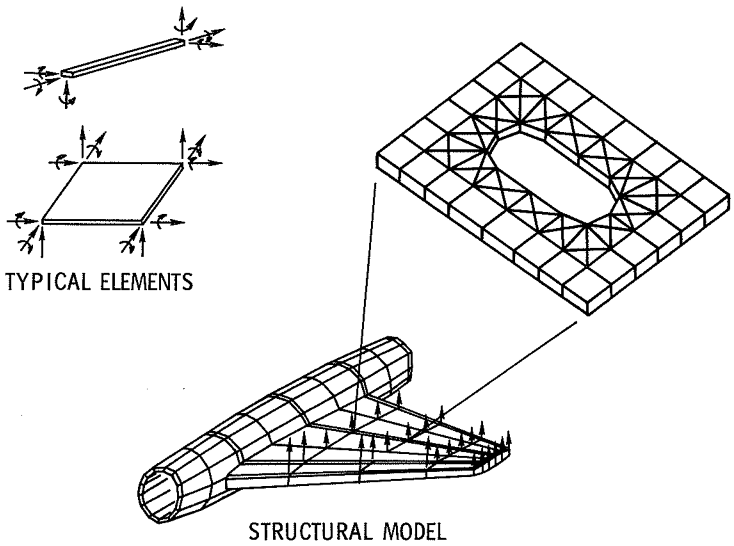


Figure 3.- Finite-element representation of complex structures.

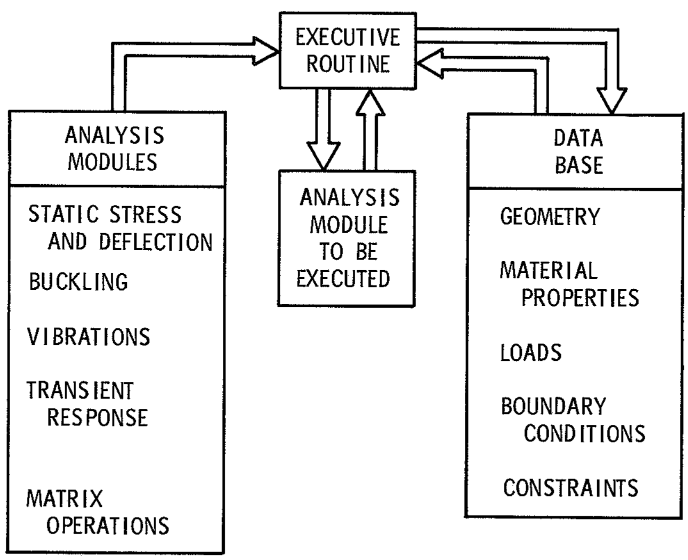


Figure 4.- NASTRAN finite-element structural analysis computer program system.

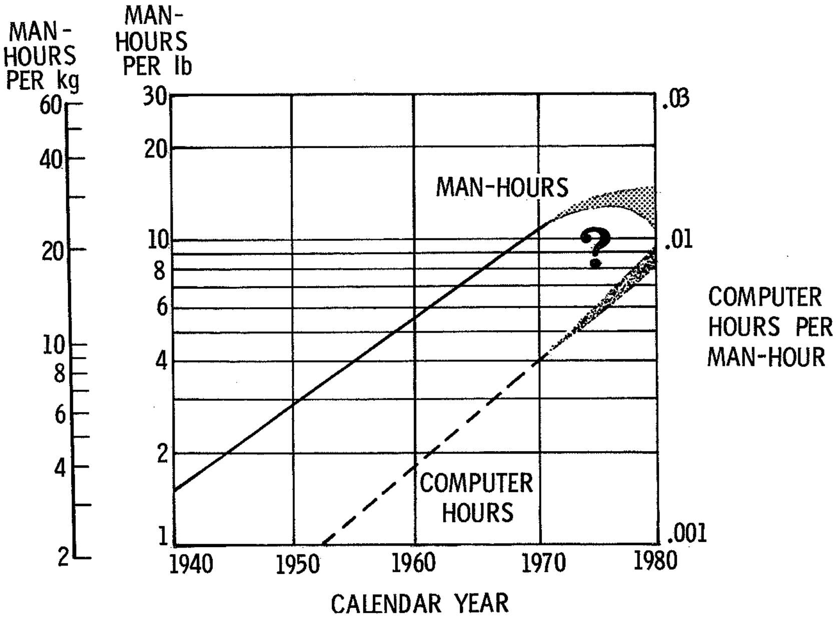


Figure 5.- Resources required for structural design of large transport aircraft (man-hours per unit mass of structure and computer hours per man-hour).

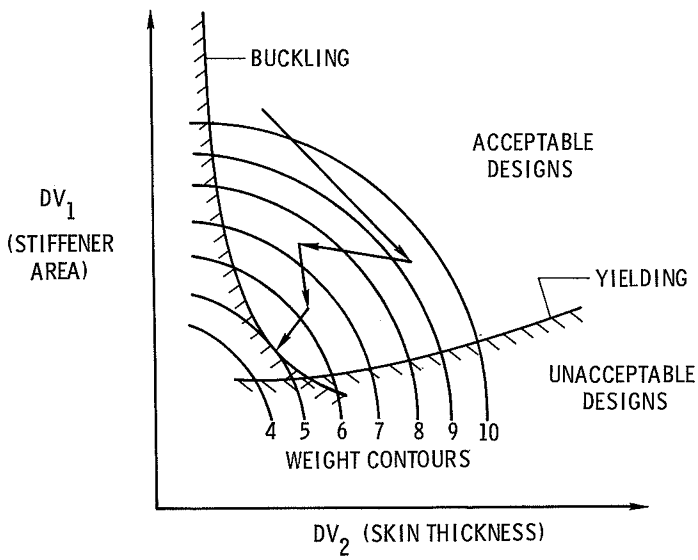


Figure 6.- Automated search procedures for minimum-weight design.

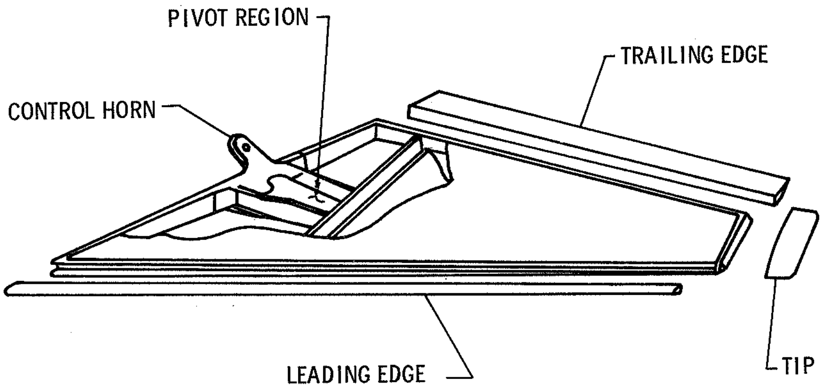


Figure 7.- Stabilizer structure for supersonic aircraft.

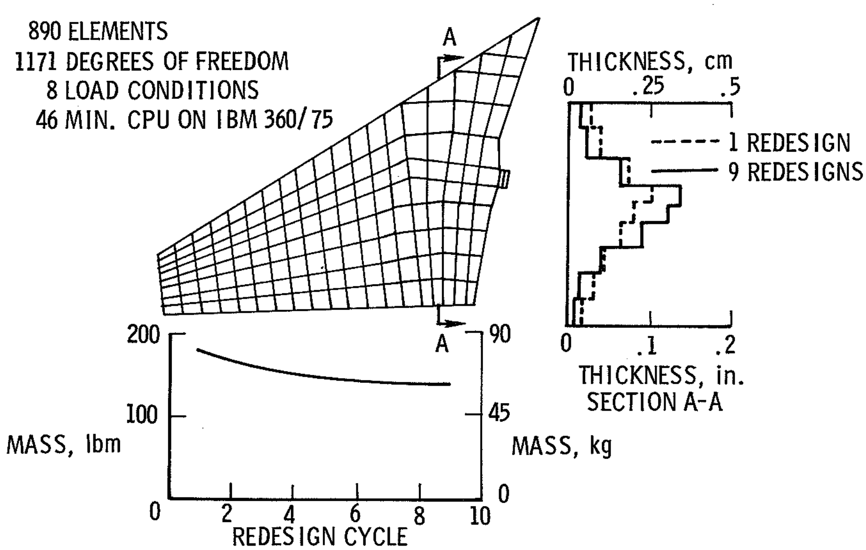


Figure 8.- Fully stressed stabilizer design.

2 776 JOINTS
5 324 BARS
2 710 PANELS
6 114 DEGREES OF FREEDOM (JOINTS)
10 814 DEGREES OF FREEDOM (TOTAL)
8 LOAD CONDITIONS
22 MIN CPU ON IBM 360/85 FOR ONE ANALYSIS-RESIZE-ANALYSIS CYCLE

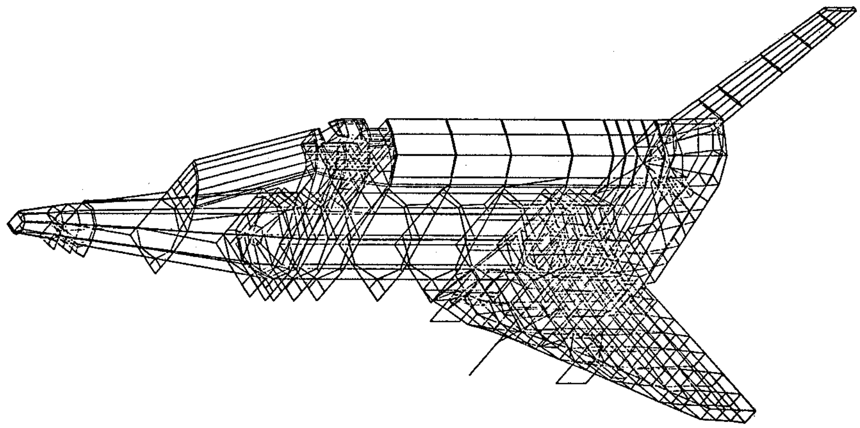


Figure 9.- Automated structural design capability – orbiter structural modeling.

1531 NODES
3512 ELEMENTS
6429 DEGREES OF FREEDOM
3 LOAD CONDITIONS
3 MACH NUMBERS

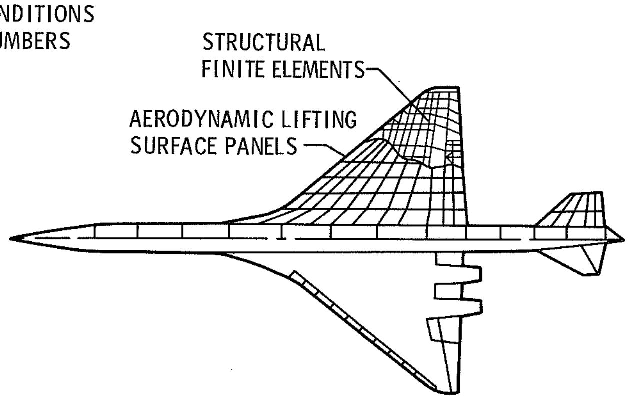


Figure 10.- SST model for Boeing integrated analysis evaluation.

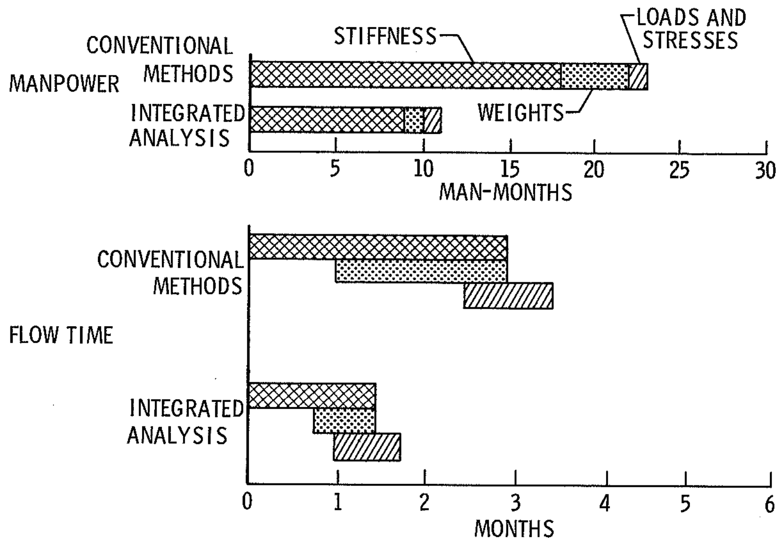


Figure 11.- Resources and time summary for Boeing integrated analysis evaluation.

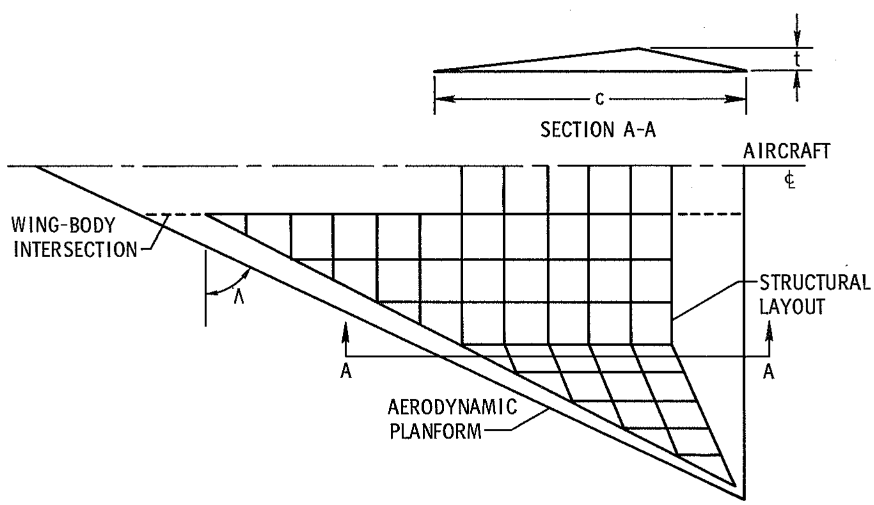


Figure 12.- Plan view of hypersonic-cruise-vehicle wing.

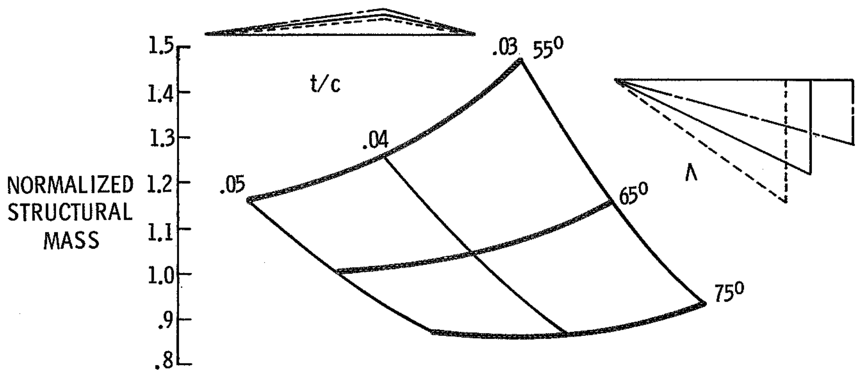


Figure 13.- Results from DAWNS for hypersonic-cruise-vehicle wing structure.

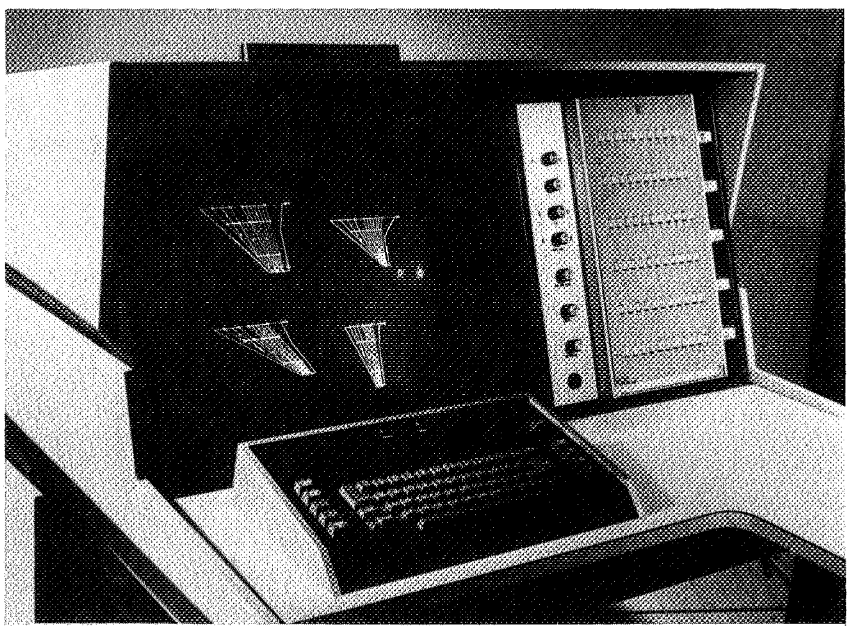


Figure 14.- CRT display console with four DAWNS wing designs.

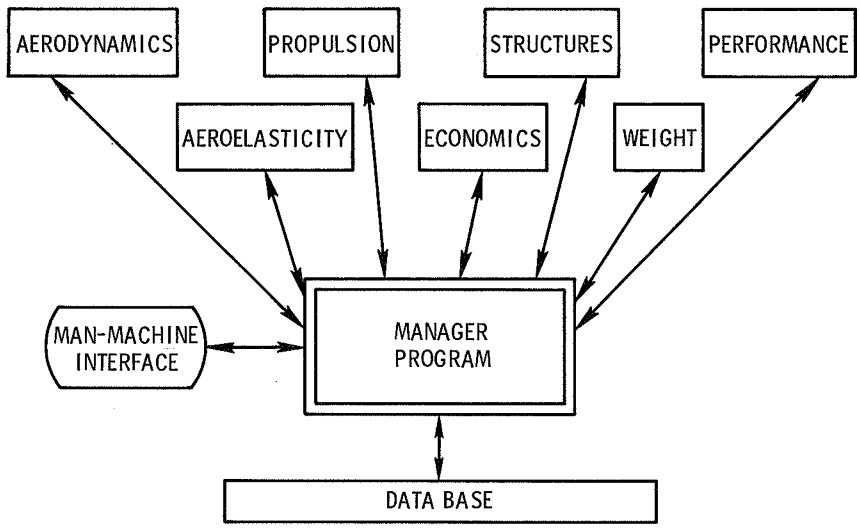


Figure 15.- Conceptual organization of integrated programs for aerospace vehicle design (IPAD).

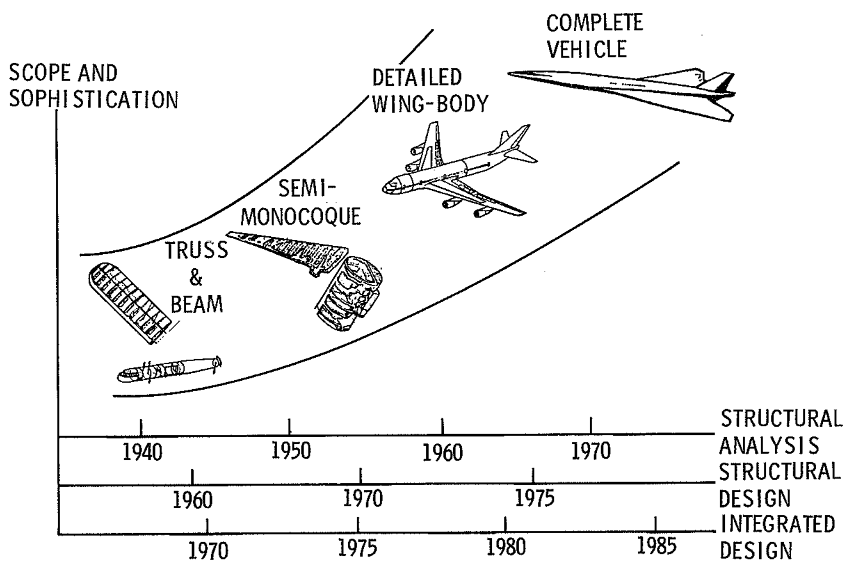


Figure 16.- Growth of automation in structural technology.

ADVANCED ACTIVE CONTROLS TECHNOLOGY

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SUMMARY

Advanced active control concepts are described and their potential for providing improved characteristics for aircraft, along with an indication of the status of the technology in relation to its readiness for application, is given. The concepts considered are relaxed aerodynamic stability, maneuver load control, flutter suppression, fatigue damage reduction, and ride quality control.

INTRODUCTION

The space program, research aircraft programs, and military advanced development programs have brought the development of modern controls technology to the point where substantial improvements in civil aircraft can be made in the near future. A quantitative indication of a potential payoff for the application of these advanced control concepts has recently been obtained by one of the advanced transport technology design study contractors. This preliminary result which was presented at a midterm review of the contract status is shown in figure 1. Figure 1 indicates, in a relative way, the return on investment (calculated by employing a particular set of assumptions and economic-estimating methods which cannot be detailed here) for fleets of two aircraft, one involving application of advanced composite structures, and the other involving application of advanced controls technology compared with the return on investment for an airplane with conventional controls and conventional structure. The improvement in return on investment for the advanced transport, which employed extensive use of composites in the wing and empennage structure, is indicated as being about 6 percent. The design which employed advanced controls for an aircraft with a conventional light-alloy structure is indicated as having an improvement in return on investment of about 9 percent. Thus, it appears that the potential payoff for application of advanced controls technology to transport designs is comparable to that of advanced composite structures. Of course, the payoff of almost any technology is dependent on the particular airplane and mission to which it is applied. In addition, the results shown, as mentioned, are of an interim nature and subject to change as the contract proceeds. On the other hand, the results are interpreted as indicating that advanced controls technology, along with other advanced technologies, should be pursued vigorously in the development of new and superior aircraft.

The advanced control concepts which were included in this economics study, and which will be discussed further in this paper, are briefly described in figure 2. The concept of relaxed aerodynamic stability is concerned with the freedom of configuration selection provided by the design approach which meets handling qualities and stability criteria by means of stability augmentation rather than by inherent aerodynamic stability. Maneuver load control involves the idea of redistributing maneuver loads in such a way as to reduce moments, and thus the amount of structural material. The concept of flutter suppression is concerned with the application of active closed-loop control systems to provide adequate margins of safety for flutter, rather than by the conventional methods of adding material stiffness, mass balancing, or dampers. Two concepts are concerned with the response of aircraft to gusts or turbulence; fatigue damage reduction pertains to reducing stresses in gust fatigue critical areas, whereas ride quality control is related to reducing aircraft motions in turbulence.

Each of these five concepts is discussed further in the subsequent sections; the discussion includes an indication of the manner in which the concept provides benefits and an indication of the state of readiness of each concept for application to civil transport design.

For a more detailed discussion of these concepts including quantitative results, the reader is referred to the bibliography which, although not intended to be comprehensive, lists most of the recent pertinent papers in this field.

RELAXED AERODYNAMIC STABILITY

This concept involves the incorporation of a highly reliable stability augmentation system which, even under emergency conditions, would provide the pilot with the same handling qualities that conventional inherent aerodynamic stability now provides. This concept permits greater freedom in selecting configuration arrangements and can lead to a smaller airplane to perform the same mission and to improvements in aerodynamic performance through reductions in drag.

Plans are underway to demonstrate this advanced control concept about 1973 by use of modified versions of the Boeing B-52 and McDonnell Douglas F-4 airplanes in the U.S. Air Force Flight Dynamics Laboratory Advanced Development Programs. This concept was incorporated in the Boeing supersonic transport (SST) design.

Some advantages of application of this control concept to a variable-sweep supersonic transport configuration have been described in paper no. 3 of this compilation by Edward C. Polhamus. Application of the concept to a fixed-wing SST design also has significant advantages, and these advantages can be discussed with the aid of figure 3. This figure shows the normal rearward shift of the aerodynamic center for a fixed-wing configuration as the Mach number is increased from subsonic to supersonic speeds.

Requirements for minimum drag or maximum lift-drag ratio $\left((L/D)_{max}\right)$ for both the wing and the horizontal tail can be met by placement of the center of gravity of the configuration at the target value indicated in figure 3. Normal fuel management can provide for the variation of the center of gravity indicated in the figure, which leads to a center-of-gravity location for subsonic flight farther aft than locations normally required for inherent aerodynamic static stability. The region of unstable aerodynamic damping would not have been noticeable to the pilot in that an ultra-reliable stability augmentation system was incorporated in the design and would have provided to the pilot the handling qualities normally associated with an inherently aerodynamically stable vehicle.

Incorporation of this hard stability augmentation system (SAS) concept in the SST provided several substantial benefits. If the configuration had been altered to meet normal aerodynamic stability criteria, the forward fuselage needed to be extended by 3.8 meters (150 inches) in order to place the center of gravity sufficiently far forward for static stability. The center of gravity would then have been at a nonoptimum position for supersonic cruise. It has been estimated (ref. 1) that incorporation of this advanced control concept in the SST permitted a weight saving of approximately 27 000 newtons (6000 pounds) with an accompanying reduction in cruise drag of about 2.5 percent. These improvements had the combined effect of giving the design an increase of 225 nautical miles in range.

MANEUVER LOAD CONTROL

The maneuver load control concept involves altering the wing lift distribution during maneuvers, shifting the load from the outboard section of the wing to the inboard section, and thus reducing wing root bending moments. This redistribution is accomplished by appropriate coordinated deflections of outboard ailerons and inboard flaps with coordinated elevator deflections to maintain trim. These controls would be deflected on command of an appropriately located accelerometer as the pilot maneuvers the aircraft. Normally, the controls would not become active until the normal acceleration exceeds some threshold value, for example, 1.5 g units; thus, for normal 1 g flight, the load distribution would not be altered from that desired for optimum lift-drag ratio. This control concept has been demonstrated in flight and is considered to be ready for fleet application in military service now. Application of the concept to civil transport designs requires the development of ultra-reliable control system components, just as in the case of the hard SAS discussed previously, in that this control system is used in lieu of structural material.

Current technology is adequate for providing control systems of sufficient reliability for safety of flight; however, the complexity of these systems leads to replacement or repair rates that probably would not be economically acceptable for airline operations.

The potential for digital systems to provide the needed jump in reliability is discussed in paper no. 16 of this compilation by George B. Graves, Jr.

Full realization of the potential of this control concept in many design situations is limited by consideration of the requirements for fatigue and flutter. These limitations can be discussed with the aid of figure 4.

A hypothetical wing structural design situation is illustrated in figure 4, which shows the variation with wing span of the amount of material required in the lower surface for three different design conditions. In the case illustrated, the critical case near the wing root is the static-strength requirement associated with the normal maneuver load. The benefit of a maneuver load control system in this design case would be limited by the requirement to provide sufficient material to meet the fatigue requirements which are just below those for static strength. Normally, a variety of loading conditions lead to the fatigue requirement which includes such factors as fatigue damage due to flight through gusts or turbulence, as well as fatigue damage due to other loadings, including the ground-air-ground cycle. Thus, a greater benefit might be provided by the maneuver load control system if an integrated design approach is adopted which also reduces the requirements for fatigue by means of a fatigue-damage-control reduction system. This concept will be discussed subsequently. However, in the case illustrated, the structural weight saving associated with these two concepts again would be limited unless a flutter suppression concept was incorporated to reduce the material requirements associated with providing an adequate margin of safety for flutter. The point of this figure is that realization of the full potential of active controls in saving structural weight requires a systems approach considering a variety of structural design conditions, with appropriate application of active controls to reduce the impact of these varied design conditions. Control systems intended to alleviate the structural requirements for some of these other design conditions are discussed herein.

REDUCED FATIGUE DAMAGE FROM GUSTS

Fatigue damage rates due to flight through turbulence and gusts can be reduced by means of moderately fast acting controls which are arranged to respond to appropriate accelerations and rates in such a way as to reduce dynamic stresses in fatigue-critical areas. In addition to the whole body motions controlled by the systems described, the fatigue damage control system is arranged to control stresses at frequencies associated with the lower structural modes as well. To the extent that structural weight would have to be added to a design, a fatigue damage control system can reduce structural weight associated with the gust or turbulence part of the added weight for fatigue damage.

This concept is the most fully developed of all the concepts discussed. A system of this type is in fleet operations in part of the B-52 fleet at present. About 200 airplanes in the B-52 fleet have accumulated over 40 000 hours of flight experience with this system. The only requirement for applying this control technology to civil transport designs is an improvement in system failure rates to meet airline operations goals. The system employed in the Boeing B-52 airplane and its performance are illustrated in figure 5.

The fatigue-damage-rate-reduction control system, which has been retrofitted to part of the B-52 fleet, employs the available control surfaces on the airplane. The rudder and ailerons are used in the lateral-directional system, and the elevators are used in the longitudinal system. These controls are actuated in response to appropriately located accelerometers and rate gyros to reduce the dynamic stresses due to flight through turbulence in the fatigue-critical areas of the airplane. The sketch to the left of the figure (from ref. 2) illustrates the effectiveness of the system in reducing the fatigue damage rate at the most critical fuselage station. The system provides at least a 50-percent reduction in the fatigue damage rate.

A closely related technology is the control of aircraft motions in gusts, as described in the next section.

RIDE QUALITY CONTROL IN GUSTS

A ride-control system operates in a manner similar to that of the fatigue damage control system, except that in this case the criterion for design of the system is that it should reduce accelerations and rotations rather than stresses. In some cases it may be possible to reduce both stresses and motions. Again, the need for an integrated system design approach is indicated to insure that the control system meets the desired criteria. The benefit of a ride-control system to a particular aircraft design is somewhat less quantitative than the other systems described in this paper in that the benefit would primarily involve such subjective matters as improved passenger acceptance and improved flight safety. A ride-control system would have a particularly desirable impact on passenger acceptance of STOL transports in light of the more frequent encounter with turbulence that the relatively low level operations of STOL aircraft will entail. The improved cockpit environment of a ride-control-equipped aircraft should provide for improved piloting performance in turbulence. Again, this benefit might be emphasized for STOL transports in landing and approach particularly when the ride-control system is integrated with an improved flight control system. This system would make reduced speed margins feasible as described by Alexander D. Hammond in paper no. 2 of this compilation.

Ride-quality-control systems are ready for application now, in the same sense that reduced-fatigue-damage-control systems are ready. A ride-control system is planned for the North American Rockwell B-1 bomber because of the low-level, high-speed severe environment in which this vehicle will operate. The system planned for the B-1 bomber is illustrated in figure 6.

The ride-control system for the B-1 bomber which is intended to provide an acceptable environment to the crew for long-duration low-level missions employs separate canard controls, as illustrated in the photograph of the model shown in figure 6. The relatively small canard surfaces are canted in such a way that differential operation of the controls provides lateral acceleration reductions. Collective operation of the canards significantly reduces the vertical accelerations in the crew compartment.

FLUTTER SUPPRESSION

In order to suppress the flutter condition, controls must react at frequencies associated with higher elastic modes involved in flutter motion. Flutter suppression systems are designed to increase the velocity at which a flutter instability would occur by decoupling elastic modes that would otherwise tend to couple and by increasing the damping of the elastic system. Again, a flutter-suppression-control system must have reliability equivalent to that of structural or mechanical systems which the control system would replace in the design. For designs which would otherwise be flutter critical, a reduction in structural weight is provided by a flutter-suppression system.

The needed improvement in reliability and maintainability of control systems suitable for flutter suppression probably can be provided by the development of digital systems such as described in paper no. 16 by George B. Graves, Jr., in this compilation. In addition to the improved reliability of systems, other advanced developments are needed for flutter suppression to become available for civil transport applications. Flight demonstration of a flutter-suppression system is planned on a modified B-52 as part of the U.S. Air Force Flight Dynamics Laboratory Controls Configured Vehicle (CCV) program. In addition, conceptual and wind-tunnel studies are underway. A photograph of a model employed in a flutter-suppression wind-tunnel study is shown in figure 7.

The model shown in figure 7 is being used in a study of the leading-edge trailing-edge flutter-suppression concept (ref. 3). Flutter can normally be thought of as involving a combination of two types of motion, that is, pitch and plunge. In general, effective control of the two motions requires two independent controllers – thus, a leading-edge trailing-edge control concept. Analytical studies of the control system illustrated indicates that it is capable of increasing the flutter speed by about 28 percent.

Technology for flutter suppression is the least well developed of all the advanced active control technologies discussed. In some cases the potential payoff for flutter suppression is sufficiently great that this technology should be pursued, and it is felt that flutter suppression could be available for civil transport applications in the 1980's.

CONCLUSIONS

Consideration of the potential payoff of applications of advanced active control systems indicates that the needed further development is worthy of substantial support. For the most part the advanced active control systems considered represent relatively modest extensions of current stability augmentation systems in use in civil transport operations today. These controls are being applied in military operations now. Realization of the full potential of these systems requires an integrated systems approach and the payoff for the systems will be mission and airplane oriented. Full application of these concepts to civil designs will require improvements in reliability and maintainability relative to today's systems. Reliability from the point of view of safety can be provided through redundancy; however, improvements in failure rates are required in order to make airline operations of such systems economically feasible.

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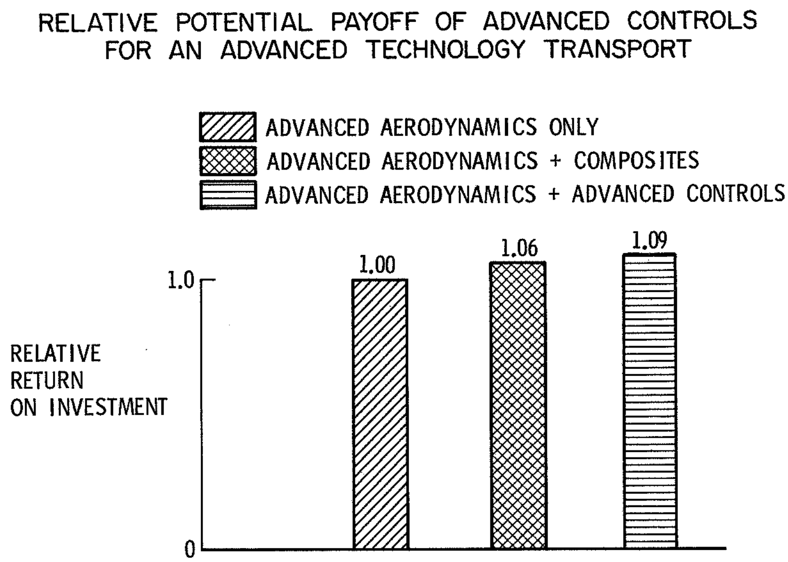

Figure 1

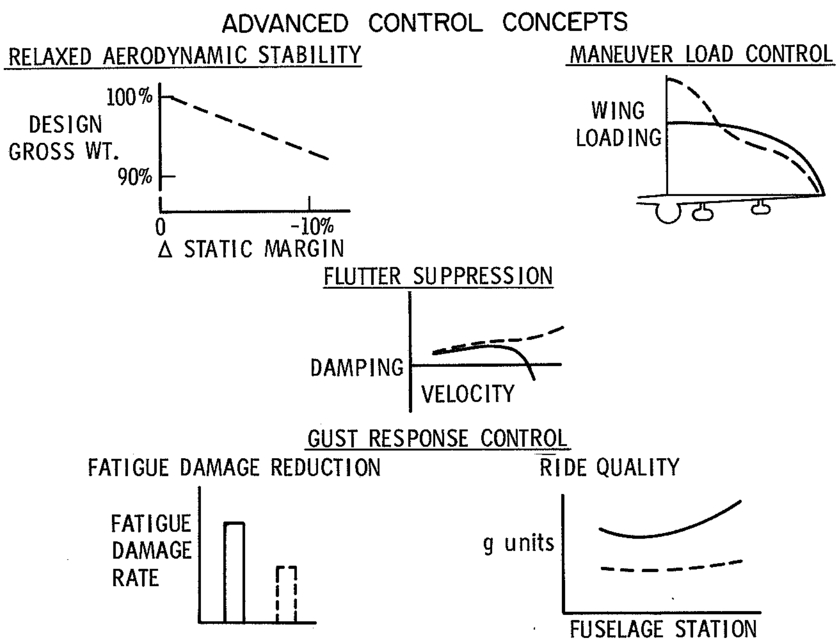

Figure 2

CENTER-OF-GRAVITY CONSIDERATIONS FOR AN SST

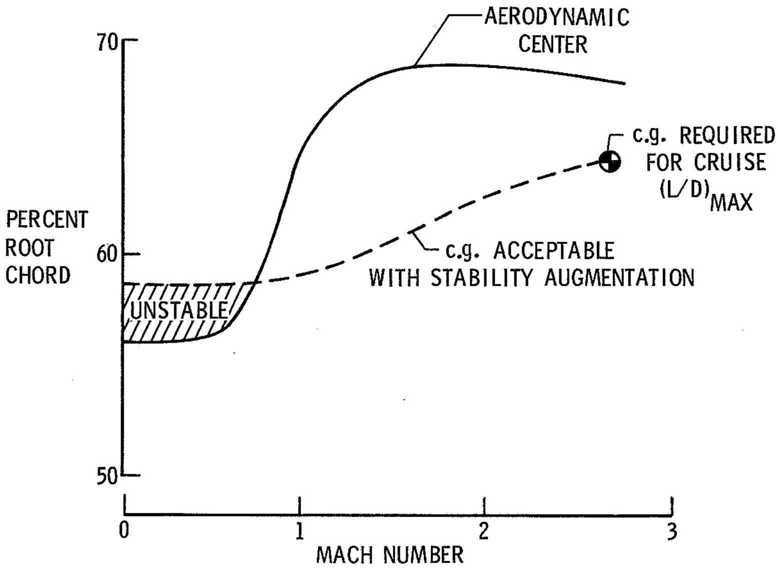

Figure 3

NEED FOR INTEGRATED SYSTEM DESIGN

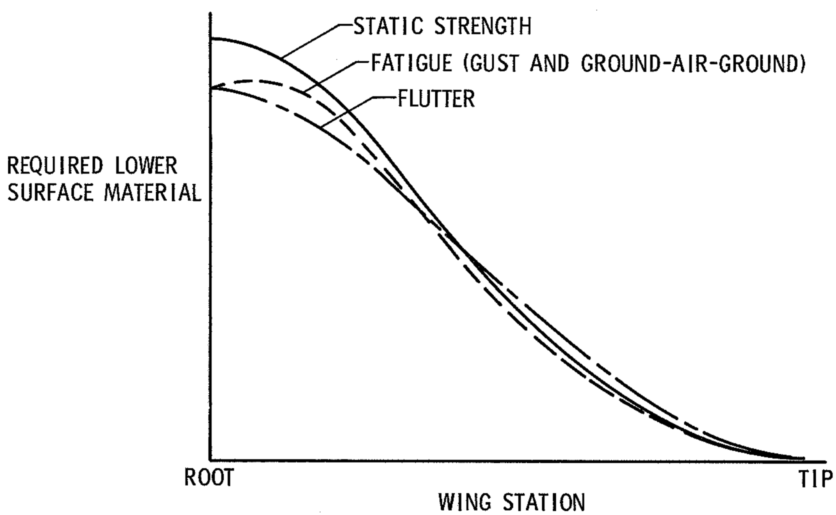

Figure 4

FATIGUE-DAMAGE-RATE REDUCTION CONTROL SYSTEM ON B-52 FLEET

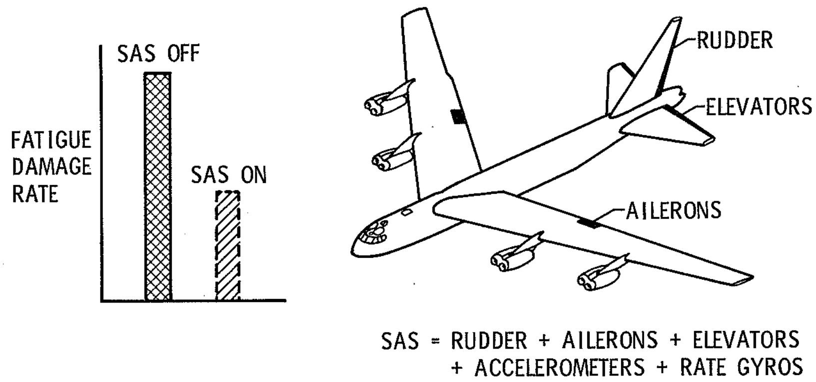

Figure 5

B-1 MODEL

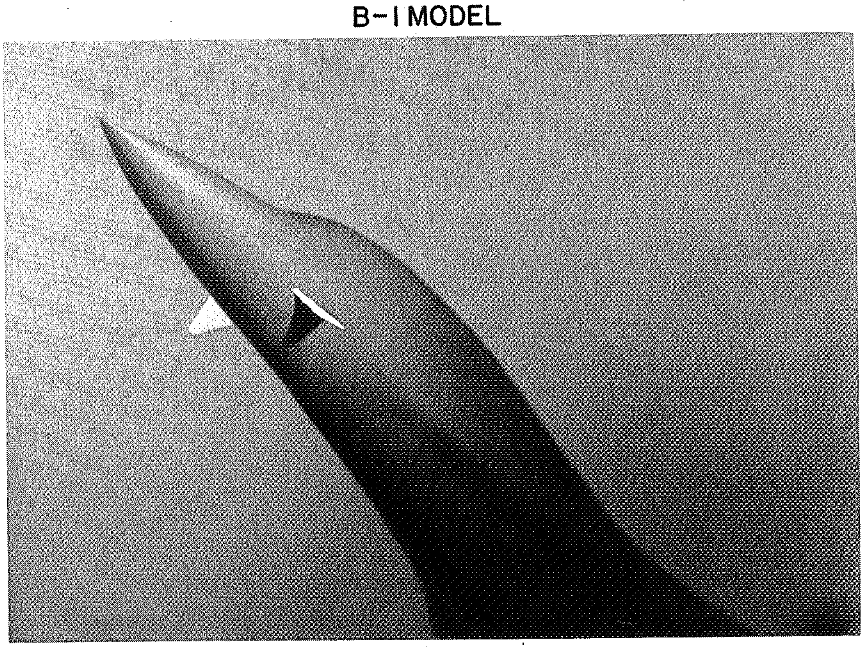

Figure 6

FLUTTER-SUPPRESSION MODEL

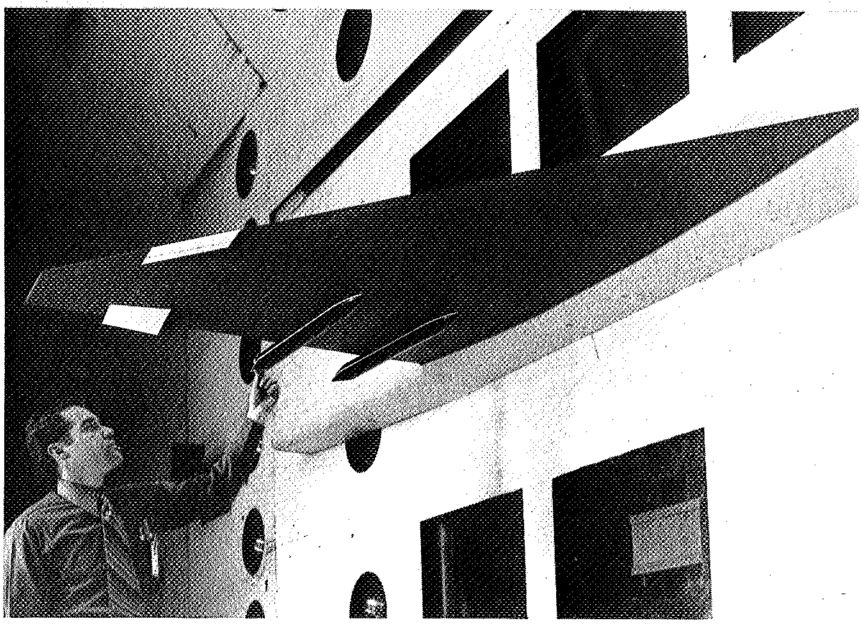

Figure 7

THE AIRPORT-AIRPLANE INTERFACE

THE SEVENTIES AND BEYOND

By John P. Reeder
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INTRODUCTION

This paper is a projection of current technologies relating the airplane and the airport for an advanced transportation system. It will deal primarily with transport aircraft in high-density traffic. The order of discussion is as follows:

(1) Air transportation situation

(2) Technical status of the operating system

(3) Airport-airplane interface

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DISCUSSION

Air Transportation Situation

Several major airports in the United States are operating at maximum capacity today and air traffic delays are common. Means for alleviating such delays now and for the future are urgently needed if growth of air travel is to be uninhibited. Although more airports and runways are obviously needed, increased utilization of existing facilities is of first priority, particularly during instrument-flight weather. The latter need will bring increasing pressures on the air traffic system and the aircraft operators to expedite traffic flow in high-density areas by time sequenced, steep, continuous descents into shorter, decelerating, final approaches to the runways. Dual-lane runways will permit take-offs alongside continuous landing operations. Application of advanced avionics and automatic control will be a necessity for full participation in the system.

Aircraft types to be considered in the decades ahead are CTOL (conventional take-off and landing, fixed wing), STOL (short take-off and landing, powered lift), and V/STOL (vertical and/or short take-off and landing). By far, the largest segment will be conventional aircraft (CTOL). STOL aircraft using powered lift will have been introduced in commercial operations by the mid-1980's. Initially, they will be applied to independent city-to-city routes from dispersed airfields to relieve air and ground congestion in the high-density areas, although connecting flights to major terminals will be provided. Inevitably, the expansion of air travel will open up vast new market demands, many requiring the use of V/STOL aircraft other than helicopters. Such aircraft will enter commercial operations in limited numbers following successful STOL operations, perhaps in the 1990's.

Today, except for one French airline, there are no commercial aircraft operations in category III weather, defined generally as a runway visual range $\leqq 200$ meters ($\leqq 700$ ft), and only very limited operations with decision heights less than 60 meters (200 ft). In the future, short-haul operations, particularly, must operate essentially independent of weather and with a schedule reliability comparable to that of local bus and train service if they are to be accepted by the public as a routine mass transportation system. Consequently, all short-haul airfields as well as major terminals must be equipped with precision landing guidance suitable for operations in category III visibilities. The discussion in the remainder of this paper will be based on operations in this type of weather.

Furthermore, scheduled operations of all types must achieve lower accident rates than currently prevail. Use of automatic control is eventually expected to improve flight safety appreciably, particularly during landing.

Technical Status of the Operating System

The technical effort for all aircraft types will have been directed toward reduced noise output and toward developing approach path capability for minimizing airspace, noise, and time during approach and landing. Figure 1 shows qualitatively the approach paths to be expected. The airplane approach path will be steepened with respect to current instrument landing systems (ILS) except for flattening of the path to reduce the rate of descent and to restabilize flight prior to the final flare. STOL approach paths of similar steepness do not have to be flattened prior to the flare because of lower initial rates of descent. The V/STOL approach path will tend toward vertical descent near the ground for noise reduction and obstacle clearance. Steeper approach paths than indicated, where the aircraft attitude corresponds to the approach path, are probably not practical for passenger comfort. The advanced microwave instrument landing system is expected to provide accurate guidance for these descent profiles except, perhaps, for V/STOL aircraft should the need for near-vertical descents develop. However, the ability to fly the aircraft along such paths during instrument flight, particularly if decelerating simultaneously, will depend on further development of flight systems, pilot displays, and automatic control.

V/STOL and CTOL instrument approach techniques are compared in figure 2. In addition to wing flaps and landing-gear extension, auxiliary lift systems must be brought into operation for the V/STOL, the aircraft attitude readjusted, and the thrust vector angle positioned for speed control at selected times. The starting of auxiliary lift systems must be judiciously timed for minimizing approach time and fuel consumption. Inasmuch as the pilot is generally working near his maximum capacity in a precision-instrument low approach in a conventional airplane today, it can be seen that the V/STOL types will require a high degree of automation to reduce the workload to manageable levels.

Proving of automatic control for advanced flight profiles in category III visibilities and establishing full pilot acceptance will require a lengthy trial period before line operations are possible, unless improved and accelerated testing can be developed. Acceptance of automatic landing and monitoring displays for V/STOL if sustained, near-vertical descents are required will be the most difficult of all because the pilot's ability to verify his position visually during landing approaches will be severely limited.

It is believed that the advanced air traffic system will require distributed management between pilot and controller (ref. 1) because it is felt that system capacity, efficiency, and safety can best be achieved this way. Distributed management means that, where judgment and safety are involved, the pilot will have command decision responsibility.

Airport-Airplane Interface

The airport-airplane interface begins where the descent from cruise flight to the terminal area is initiated. At jet speeds and altitudes, this represents perhaps 20 minutes flight time or 125 to 300 nautical miles from the airport. The influence of sequencing may, however, extend farther out.

A simplified terminal area is shown in figure 3 to illustrate future traffic flow. Aircraft are envisaged as flying on multiple-parallel trunk routes at standardized speeds, dependent on altitude. Slower aircraft will be constrained to lower altitudes. Flights will also be constrained to specific vertical paths, most generally in descent. Because of increasing congestion of flight paths in many terminal areas, sequencing is going to have to be handled through speed control combined with only minor lateral maneuvering. Note that in the figure, V/STOL approach paths lie over those for CTOL when close in, but under those for CTOL when farther out.

It is probable that in the future, transport aircraft will be designed and certified on the basis of specific maneuvering, approach, and landing speeds. This implies segregation in landing on the basis of speed categories at high-density-traffic airports.

The necessity for long, straight-in, final approaches will have been alleviated for all aircraft. The advanced control, display, and guidance systems will permit shorter, curved, and steeper paths, perhaps depending on passenger acceptance in the case of STOL and V/STOL. Pilot acceptance, on the other hand, may initially require constant speed and constant configuration for about the last 150 meters (500 ft) of descent for wingborne aircraft. Also, pilot acceptance may require that the approach track straighten out in alinement with the runway some distance prior to the threshold and the landing flare. Rates of descent must be less than $4\frac{1}{2}$ m/s (900 ft/min) below about 60 meters (200 ft). (See ref. 2.)

Automatic control of transport aircraft will eventually become routine in all phases of flight. It is assumed that the steeper, curved descent paths desirable for noise alleviation and time saving can be readily achieved automatically. The effectiveness of these paths for noise alleviation and time reduction can be further enhanced if the aircraft can be allowed to decelerate through the final approach to the threshold of the runway. (See ref. 2.) This maneuver appears feasible only if performed automatically, at least for wingborne aircraft. Figure 4 illustrates possible noise reduction for a decelerating two-segment approach. In this example, speed is reduced with automatic positioning of the flaps and throttle. The thrust remains lower than on a constant-speed 3° approach throughout. However, the problem of pilot monitoring and takeover during such nonsteady operation is greater than for an operation with constant configuration, speed, and thrust. Therefore, considerable operational experience will be required for acceptance.

Inasmuch as pilot acceptance of curved, steep, and decelerating approaches to low-visibility landings will take a lengthy time period, it is imperative that efforts at noise alleviation by quiet-engine development and nacelle treatment, as well as by the adjustment of the ground track, be continued.

The trailing-vortex hazard can be minimized for closely spaced aircraft on approach by constraining them to a given planar surface. There is hope that research now in progress on the mechanism of trailing-vortex decay will result in a means for modifying the wing flow characteristics to accelerate vortex breakup. A major part of the energy must be dissipated within about 40 seconds to be wholly effective, however, as this amount of time is the interval between aircraft anticipated for landing, all else considered. If vortex breakup can be accelerated, the rate of vortex dissipation will be greater in the presence of the ground than in free air. Vortex-detection systems will monitor key locations in the approach, landing, and take-off paths for all runways, however, as a safeguard against unusual vortex persistance. Of course, if such situations do occur, traffic will suffer delays.

If early vortex breakup can be induced, a vertical separation of only 150 meters (500 ft) for crossing paths in the terminal area may be all that is required. However, allowance must be made for abort paths of aircraft underneath.

Take-off and landing operations in cross winds have always created problems, but community-noise considerations are now forcing the use of preferential runways with reduced consideration of cross-wind conditions. Landings in strong cross winds can cause difficulties inasmuch as control effectiveness may be marginal, and acceptable bank angles to counter drift at touchdown are limited in large aircraft. Staying on the runway after landing during heavy braking or on wet or icy runways may be even more difficult inasmuch as the cornering capability of tires is reduced in these circumstances. As illustrated in figure 5, the STOL airplane is more critically affected by cross winds than conventional airplanes because of its lower landing and take-off speeds. The crab angles shown in the illustration must be reduced or eliminated before touchdown with conventional landing gear. Take-off performance of all aircraft with conventional gear in cross winds is adversely affected by the drag due to lateral drift and scrubbing of the tires, as well as the drag due to the use of controls, such as lateral-control spoilers, which may be required during the take-off roll. These effects have never been defined in terms of take-off performance. In the future, aircraft design for satisfactory operations must consider at least 30-knot cross winds, both for coping with preferential runway constraints and for STOL operations from single-runway airports. Suitable cross-wind landing-gear configurations will be required for many STOL aircraft applications in particular, and probably for conventional transports as well. Studies are being made of castering wheel arrangements suitable for application to a STOL aircraft with tricycle gear.

The air-cushion landing-gear concept has a potential for alleviating cross-wind landing and take-off problems. A cushion of air escaping from underneath an elongated doughnut-shaped ring, when inflated, supports the aircraft without ground contact. Thus, the aircraft is free to assume any crab angle necessary over the runway. Independently inflatable pads on the bottom of the ring make contact with the ground to provide braking.

The U.S. Air Force and the Canadian government are now conducting a joint development program for flight testing the air-cushion landing system on a modified De Havilland C-8 transport aircraft. Landing tests with a 1/10-scale dynamic model of this airplane are being conducted by the Langley Research Center in support of this program.

Some potential advantages and some potential problems to be investigated for an air-cushion landing system are as follows:

Advantages

Cross-wind landing and take-off characteristics

Ability to negotiate rough terrain and water

Reduced landing-gear weight for very large aircraft

Reduced unit ground loads

Problems

Braking

Cornering forces for ground control

Retracting for cruise

Maintenance

Debris circulation

Noise

Whether landing manually or automatically, conventional aircraft will require a means of direct lift control for positive touchdown positioning. Also, for all aircraft, quick engine response, even at low thrust, and automatic speed control will be required. Automatic landing for all aircraft types will require inertially smoothed and quickened autopilot coupling to the guidance signals. Improved displays for monitoring the approach driven by sensors independent of the ILS will be required. The displays must be good enough to accomplish a manually controlled landing if necessary. Braking and reverse thrust will normally be applied automatically at ground contact to minimize the landing run.

Most major airports today have evolved well beyond their initial planning; and their runways, turnoffs, taxiways, and ramps are not configured for maximum movements. A part of the plan of a major new airport now under construction (Dallas-Fort Worth Regional Airport Land Use Plan-2001, Aug. 1969), which applied some of the latest concepts for increasing traffic flow rates, is shown in figure 6. One of the airport's two major parallel runways is shown. (Its take-off and landing sequencing are independent of those on the other major runway which is not shown.) It is a dual-lane runway for achieving a maximum rate of mixed operations. The lanes are separated by about 300 meters (1000 ft). Take-offs are permitted on one lane dependent to some degree on longitudinal separation from closely sequenced landings on the other. This arrangement can increase capacity for total operations by about 40 percent. (See ref. 3.)

High-speed turnoffs usable from either direction are shown for the major runway lanes. Uniformity of landing speed and accurate control of the touchdown spot should make it possible to use high-speed turnoffs effectively if properly located and to reduce runway occupancy time to about 25 seconds from the threshold. Traffic spacing at such an interval, however, will not allow decision and action time for an abort by following aircraft. If a 15-second safety margin is included, a suitable threshold interval might be about 40 seconds. The ability to maintain such an interval within ±5 seconds promises to increase capacity about 70 percent. (See ref. 3.) Runway center-line guidance during landing runout will be required, as well as guidance into the high-speed turnoffs, if speed permits. If not, an abort has to be commanded for the following aircraft if the spacing is near minimum.

Taxiways will have to remain uncongested if high-frequency landing intervals are to be used. Multiple taxiways and an expansive ramp, as shown in figure 6, are required. Note also that dual taxiways parallel to the runways permit inbound and outbound movements simultaneously. However, important keys to control of congestion on the ground are realistic scheduling of flights and elimination of air traffic delays.

To expedite take-offs, high-speed entrances to the runways should be provided similar to the high-speed turnoffs for landings. These entrances are provided to the dual-lane runway, as illustrated at the right of figure 6. These would permit entering the runway at speeds comparable to those for high-speed turnoffs. Runway center-line guidance will also be required during the take-off roll.

Noise-abatement power cutbacks combined with maneuvers after take-off have a built-in accident potential. The pilot should be expected to follow simple procedures only, such as maintaining fixed power, heading, lift configuration, and speed to about 300 meters (1000 ft). Higher thrust of future aircraft should improve the situation by making steeper climb paths and higher rates of climb possible. Better displays of a situation type with flight-path prediction capability would then make outbound tracking much easier and safer.

CONCLUDING REMARKS

Marked advances in aircraft, flight control, and traffic management are going to occur in the next decade, with a promise of perhaps doubling runway capacity for mixed take-off and landing operations. Demonstration of advanced flight profiles and techniques with automatic control could occur within 5 years with an appropriate expenditure of effort. However, the implementation, the proving, and the acceptance of the results of the demonstration in viable airline operations will probably take an additional 5 to 10 years for CTOL and STOL aircraft types, depending on the level of national support. V/STOL aircraft will take somewhat longer to reach a similar status, depending on the demand for V/STOL service and the added complexities of the required aircraft systems.

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GUIDED LANDING PATHS

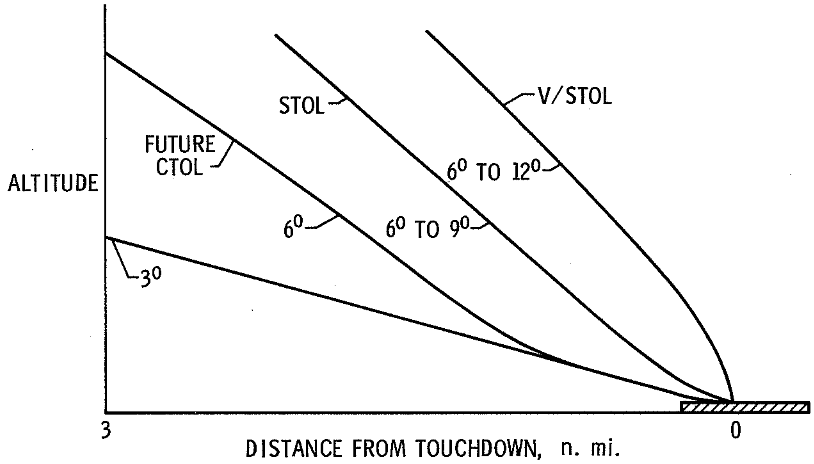

Figure 1

APPROACH TECHNIQUES

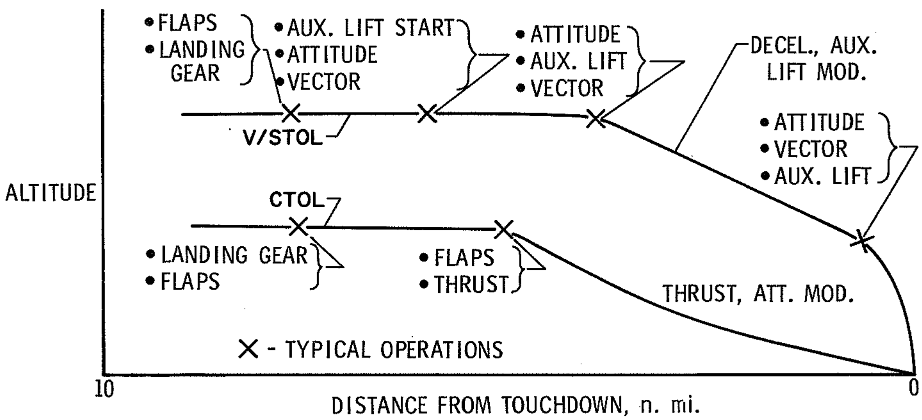

Figure 2

TRAFFIC FLOW AT MAJOR TERMINAL

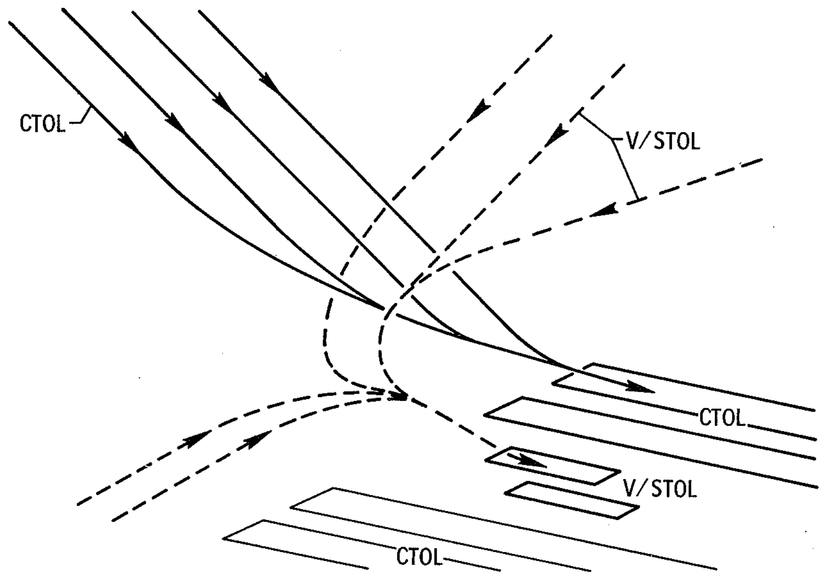

Figure 3

DECELERATING 6° TO 3° APPROACH

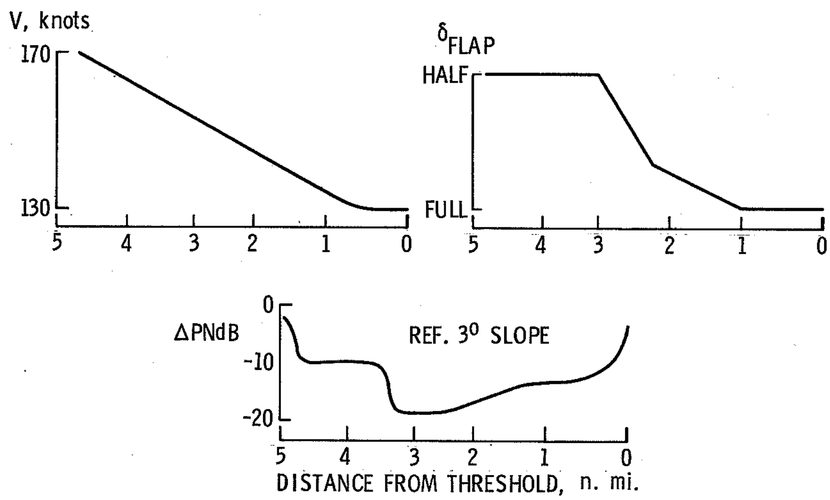

Figure 4

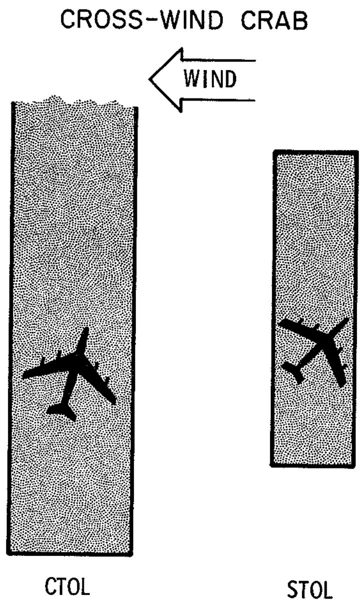

Figure 5

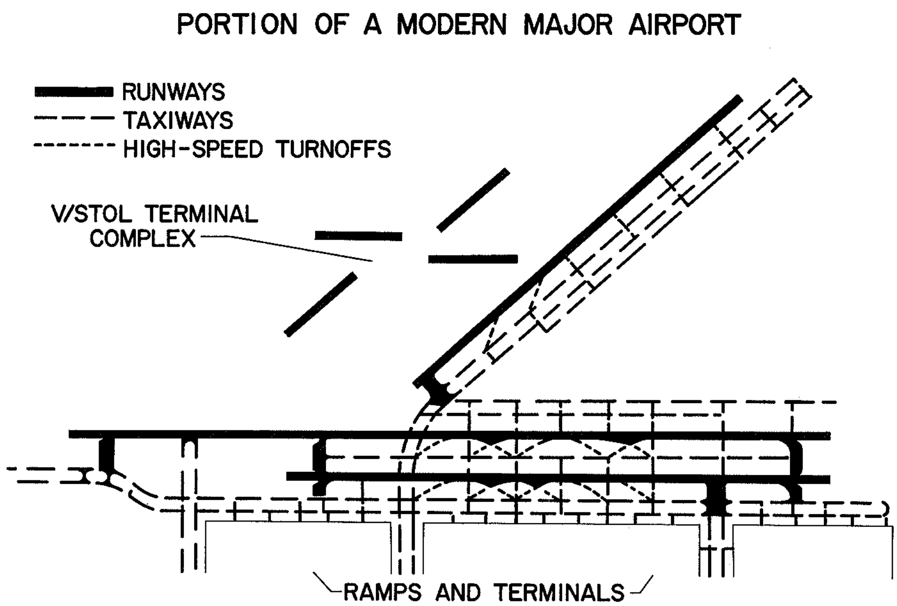

Figure 6

THE PILOT-AIRCRAFT INTERFACE

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INTRODUCTION

The pilot-aircraft interface is illustrated in figure 1. It naturally centers in the cockpit but is specifically represented by the means through which the pilot receives his information and through which he, in turn, controls or communicates with the aircraft and the environment. In any aircraft each of the pilot's channels of sensory perception is utilized in one way or another. The predominate input channel is visual; this channel includes both information external to the cockpit and that derived from instrument displays within the cockpit. Other input information is derived from hearing, certain kinesthetic cues and disturbances, and other sensory cues, such as touch of the controls and selectors and smell. Primary output command is exercised through conventional controls and selectors. This interface could apply to any pilot-aircraft combination; however, the remainder of the discussion will be directed at the civil transport because technology application for business and other general aviation aircraft generally follows the transport. Some of the pilots of these aircraft, however, may not have the assistance provided by a second or third crew member as the pilot of the transport has; therefore, pilot-aircraft interface requirements are affected.

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First an attempt to define the nature of the problem is made; then the state of technology in this area is briefly considered. Next, the requirements for applying this technology as well as that for assessing promising technology for application to the pilot-aircraft interface are examined. Then a review is made of the most important elements of technology which should be used during the 1970's and which will have a significant impact on the civil transport cockpit of the 1980's.

DEFINING THE PROBLEM

Controls, selectors, and dial and needle instruments which were in use over 30 years ago are still common in the majority of the civil aircraft presently in use. By comparing the cockpit of a 30-year-old three-engine transport (fig. 2) with that of a current four-engine jet transport (fig. 3), this similarity can be seen. However, the cockpit of the jet transport has become much more complicated than that of the older transport because of the evolutionary process of adding information by more instruments, controls, and selectors to provide increased capability or to overcome deficiencies. This trend toward complexity in the cockpit can be attributed to the use of more complex aircraft systems and the desire to extend the aircraft operating conditions to overcome limitations due to environmental constraints of weather (e.g., poor visibility, low ceiling, etc.) and of congested air traffic. System complexity arises from adding more propulsion units, stability and control augmentation, control automation, sophisticated guidance and navigation systems, and a means for monitoring the status of various aircraft systems.

The flight-control panel of a simulator equipped with several advanced display concepts is shown in figure 4. (See refs. 1 and 2.) The master command and attitude display, or electronic attitude director indicator (EADI), is located directly in front of the pilot and the horizontal situation and navigation display is located directly underneath the EADI. To the right of the EADI is a third electronic display which can be used for supplemental information in whatever detail the pilot needs. In the example shown in figure 4, additional vertical situation information is provided. These three displays provide the pilot with simplified, easy-to-interpret information; they combine and integrate functionally related information and thus reduce the clutter of many individual instruments which otherwise tend to fragment the pilot's attention. With such displays, only information required at a given time need be displayed; however, the overall situation would still remain apparent to the pilot. By time-sharing displays, scaling, content, and sensitivity can be adjusted for specific flight conditions.

The primary flight controls have been redesigned and relocated to avoid interference with the primary displays and the computer keyboard. To improve the primary controls, cross coupling and interaction between controls need to be eliminated, more uniform feel and predictable response characteristics are necessary, and a better means for interfacing with the automatic control systems is desirable. For many aircraft, particularly V/STOL, pilots need integrated propulsion controls to simplify handling of the complex mixture of propulsive and aerodynamic lift as well as some form of attitude stabilization either through attitude command or a rate command with attitude hold.

STATE OF TECHNOLOGY

Figure 5 depicts some of the available technology that is desirable in the cockpit; however, space is a limiting factor. Incorporation of Data Link (an automatic means of transmitting information between aircraft and the ground), for example, could significantly reduce pilot workload in communication and data handling. Records are available which show a single crew making 98 frequency changes and over 500 radio contacts in a normal working day. (See ref. 3.) The head up display of flight-path information has only recently achieved a functional design, an objective evaluation, and inclusion in a civil aircraft cockpit.

The trend toward clutter and complexity in cockpits caused by sequential additions must be overcome. That progress is being made in this direction is apparent from a study of the latest wide-body jet cockpits, but even these do not supply the fully integrated and functionally oriented cockpit that pilots desire (ref. 4). The need is not only to find a way to get more information into the cockpit, but to do it in a manner which neither compromises the existing pilot-aircraft performance nor increases pilot workload. There is a need to provide even better performance combined with greater capability, such as operation to lower minimums or under increasingly severe environmental conditions. As noted in reference 5, a fully integrated cockpit is needed; otherwise, many of the advantages of simplification through easy-to-interpret display and increased use of automation in control and data handling may be lost. Examination of the existing concepts and hardware that could be used for improving pilot-aircraft interface leads to the conclusion that lack of technology is not the primary problem, but lack of application is.

APPLICATION OF TECHNOLOGY

The important problem is determining how existing technology can be applied more effectively to the pilot-aircraft interface and thereby improve operational performance and safety. To obtain this goal, (1) the pilot's needs must be evaluated, (2) the pilot's role must be reexamined, (3) certain conflicts and trade-offs involving pilot-aircraft interface must be resolved, and (4) promising technology must be used in a cockpit, systematically evaluated in a realistic task, and modified to insure that it meets the pilot's needs as well as the mission requirements.

Pilot's Needs

One method used for determining what the pilot actually needs is studying accidents and incidents. This method has produced some penetrating insight into design-induced pilot errors and the "combinations" of circumstances that result in overloading the pilot. (See ref. 6.) Determination of the pilot's needs often can be made directly by asking, or at least by paying attention to, the pilot's definition of the problem and his solutions. (See refs. 4 and 5.) The pilot's definition is often important because of actual operational experience involving the full-task elements. It is also possible to evaluate pilots' needs from their collective response during simulation and flight training.

Pilot's Role

In discussing the pilot's role, examination of the simplified block diagram shown in figure 6 is helpful. This figure shows the principal elements in the pilot-aircraft interface loop which influence handling qualities. (See ref. 7.) The pilot-aircraft interface loop shown here consists of an inner and outer loop. Different concepts of the role of the pilot from controller to manager are developed depending on how these loops are closed. In the inner loop, the pilot acts as a controller and he personally is involved in each element of aircraft response. Information is needed to complete the loop. In the outer loop, the pilot acts as a manager, directing by command the completion of this loop. Each role results in some level of task performance and workload. Because of preoccupation with manual control in the past, it has been a general practice to associate handling qualities primarily with the aircraft stability and control characteristics. Actually, handling qualities encompass not only the aircraft stability and control but the total of the pilot-aircraft interface features as well, and all these features are, in turn, subject to the forces of the environment and the stress on the pilot. In the past, the pilot has been so involved with inner loop control dictated largely by stability and control characteristics that his primary purpose of closing the outer loop, which involves total test performance, may be forgotten. Only by reducing the workload involving the inner loop, can he find time to devote the necessary attention to outer loop closure and completion of the primary mission.

Regardless of whether the pilot serves as a controller or exercises command functions only, he still must have the necessary information to complete the outer loop. The important question then becomes, What information should the pilot be provided with to enable him to do this adequately? Experienced pilots, in general, agree that cockpit displays have not received the attention they deserve. Most feel that improved information displays in the cockpit can provide a major contribution to civil aviation and are essential to the selective use of automation. Pilots see their role as changing progressively toward one of management rather than physical involvement but with the pilot still providing a fail-operational link in the system. Pilots see a requirement to remain in the loop as the interface between the machine and the environment with automation, simulation, and data handling having important roles to play in maintaining and developing this interface. All this must be done while recognizing human limitations and requirements. (See ref. 5.)

Conflicts and Trade-Offs in the Pilot-Aircraft Interface

Many conflicts and trade-offs in the pilot-aircraft interface are involved which preclude designing the cockpit solely from ideas such as those presented by the pilots; therefore, additional considerations become necessary. Pilots want simplicity rather than complexity – that is, they want more information, but fewer instruments to look at; precision of control without undue workload; and a semblance of standardization among aircraft without stifling innovation and the incorporation of new and better ideas. In addition, the manufacturer and operator are faced with maintaining a balance between improvements to enhance safety and economics. Cost is always an important consideration; thus some certainty that the cockpit hardware being added will actually be accepted and used in operational service is required.

Assessing New Technology

At some point the projected interface technology must be installed in the cockpit where pilot assessment involving the important task elements can be obtained. Without an aircraft development program or major flight programs, it is almost impossible to get major changes and improvements in the cockpit. Systematic evaluation using flight simulation techniques can however provide an increasingly useful tool for expediting application of pilot-aircraft interface technology by assuring in advance that pilot acceptance and mission requirements will be obtained.

TECHNOLOGY FOR THE 1970's

In discussing the technology for the 1970's, the technology which can be used to meet the pilot's needs and validation of its application through systematic evaluation will be examined. The important technology that is required to meet the pilot-aircraft interface needs is computer displays for information, digital systems for control, computer displays for command and management, and flight simulators for pilot assessment. Much of this technology is dependent upon onboard digital computers, the advantages of which are generally well known. By using and applying computer displays, digital flight-control systems, and advanced flight simulators during the 1970's, the transports of the 1980's, that is, advanced short take-off and landing aircraft for short-haul transportation and control-configured vehicles having extremely high aerodynamic efficiency, could be provided with major improvements in the pilot-aircraft interface.

The digital computer through its extreme accuracy, easy-to-modify programing methods, and packaging (miniaturization) offers many potential applications. Its inherent reliability through self-testing and exact repeatability combined with a large information storage capacity and high-speed computations make it ideally suited for the varied requirements which have just been discussed. It provides a flexibility so great that many of the ways it could improve the pilot-aircraft interface are not yet known.

Digital primary flight-control systems can provide improved flying qualities and additional automatic control and stabilization modes. They offer the first real hope for achievement of a reliable fly-by-wire control system having extreme flexibility and adaptability. Computer flexibility in computation tasks is such that complex interactions can be uncoupled and control modes provided which are more in line with longer term objectives. Standardization in computer hardware appears practical without compromising flexibility; the same computer can support different aircraft and missions.

As the basis for advanced electronic displays, the inherent flexibility of the digital computer is important in making available a wide selection of information through the use of time-shared and multifunction displays and the large storage capacity. An almost unlimited amount of information can be displayed in almost any manner desired. As a result, a potential exists for simplification through integration of the information needed at a given time into simple, easy-to-interpret displays. Functions related to any aspect of the pilot-aircraft interface can be displayed; it remains only to devise the displays and their information content in order to provide the pilot a wide selection of information. Standardization is also possible in computer hardware as well as in the display elements.

Much available technology is never applied because it does not receive the necessary systematic evaluation and objective pilot assessment to assure that it will meet the pilot's needs and fulfill mission requirements. Both the research and development simulator (figs. 7 and 8) with its often unique capabilities and the advanced training simulator (fig. 9) with its full-task capability represent the tools for determining how the digital computer technology can best be used to meet the needs in the pilot-aircraft interface.

Simulators of varying capability, as shown in figure 10, can and should be used during research, design, and development to provide an initial feedback regarding pilot acceptance and to guide development. These simulators provide the means whereby an iterative process of test, evaluate, and modify can be applied in advance of the aircraft. During the flight-test and the certification, or acceptance, phases, the simulator not only provides the means for preparation of flight-test plans and certification procedures in advance but also affords the opportunity for detailed examination of the alternate procedures for using the computer flexibility as well as the problems uncovered during flight tests.

By the time an aircraft enters the training and operation phase, the full-task mission training simulator is available and enables the assessment of the new cockpit features under more realistic task and workload conditions. Here the pilots and other crew members learn to function in the new cockpit environment and to efficiently use the controls, selectors, and displays provided.

With the continuing trend toward more automatic control and more instrumentation and information in the cockpit, pilots will still require recurrent training in how to use manual controls and the emergency aspects of degraded levels of stabilization control and information. In this training phase, the pilots have their first opportunity to examine and evaluate the aircraft and especially the pilot-aircraft interface features in the context of the full and continuous mission requirements. By incorporating the actual computer hardware into the simulation programs, as with "iron bird" mock-ups using control system hardware, much can be gained in establishing reliability of the man-machine interface in such computer applications. Not only is reliability of the computer hardware important, but the total system reliability wherein the system must fail operational with the pilot still in the loop must be considered. Upon completion of the initial training period, operational service with the aircraft begins. As operational experience accumulates, knowledge from incidents and accidents are fed back into the system by a process which ranges from slow osmosis to instant reaction. There is a need, however, for improvement in the feedback of lessons learned from operational experience, particularly those lessons which involve the pilot-aircraft interface. This improvement can be made by providing "operational" feedback (fig. 11) from experience gained during new crew training or by crew-performance studies conducted in advance or in parallel with normal training programs. Essential to such operational evaluation studies is the full-task mission advanced training simulator, or advanced research simulator, which more closely approximates the operational workload situation. By utilizing this advanced training simulator to obtain "operational" feedback pertaining to simulated emergencies requiring onboard decisions and design-induced pilot errors, corrective action could be determined quicker and with greater confidence because of the more realistic operational conditions.

APPLICATION FOR THE 1980's

A major advance during the 1970's must be the development of more effective means for systematically evaluating the available technology for improving the pilot-aircraft interface if major innovations in the cockpit are to be obtained during the 1980's. Figure 12 provides a pictorial illustration of a few of the innovations which should be possible.

With the bulky control columns removed from in front of the pilot, important cockpit space would be made available to provide better access to controls, selectors, and displays. Control and flight management could be simplified by provision of hand or finger-tip controls through which the pilot would have better interface with the automatic systems and could control or command the aircraft to follow functions, such as flight-path angle or gradient, ground track, ground-speed hold, rate of climb or descent, and so forth, more in line with the pilot's objectives.

A series of electronic displays will probably be used not only to provide redundancy but to enable logical association of related information. Simplified, easy-to-interpret displays will be available to the pilot and will provide the information needed for a given phase of flight. With time sharing of individual display units, it will be possible to eliminate the clutter in any given display.

The pilot will be able to use optimum sensitivity, scaling, and content for any given phase of flight. He will be able to change displays as desired or to call up additional information as needed on another display, and he should be able to command and control the aircraft with precision unthought of a few years ago. With a vast supply of information available for callup by the pilot in whatever detail is desired, there need be no doubt or confusion with respect to the existing or developing situation.

CONCLUDING REMARKS

In this paper, it has been noted that it is not a lack of technology that is preventing a major redesign of the cockpit and incorporation of desirable pilot-aircraft interface features, rather it is the lack of application of existing technology. A major advance during the 1970's must be the development of a more effective means for systematically evaluating the promising technology for improving the pilot-aircraft interface.

The important technology most likely to influence major change and improvement in the civil aircraft cockpit of the 1980's and beyond is based upon the onboard digital computer. This technology will include computer displays for information, digital systems for control, computer displays for command and management, and simulators for pilot assessment. If applied, this technology could result in simplified control and flight management, better interface with automatic systems, and improved flying qualities. A series of electronic displays would probably be used to provide redundancy and to enable logical association of related information. They also would provide simplified, easy-to-interpret displays which would enable the pilot to call up information as needed with the desired sensitivity, scaling, and content for any given phase of flight. By applying this technology, the pilot would be able to command and control the aircraft with ease and precision unthought of a few years ago.

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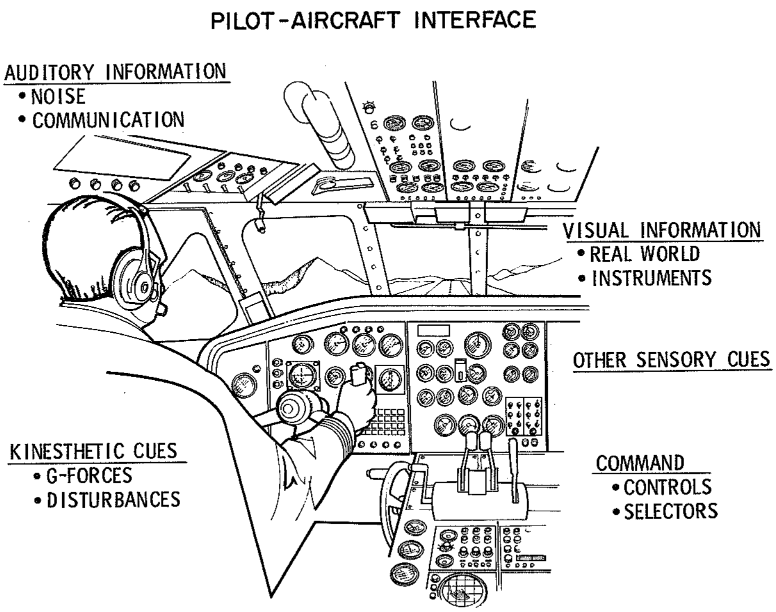

Figure 1

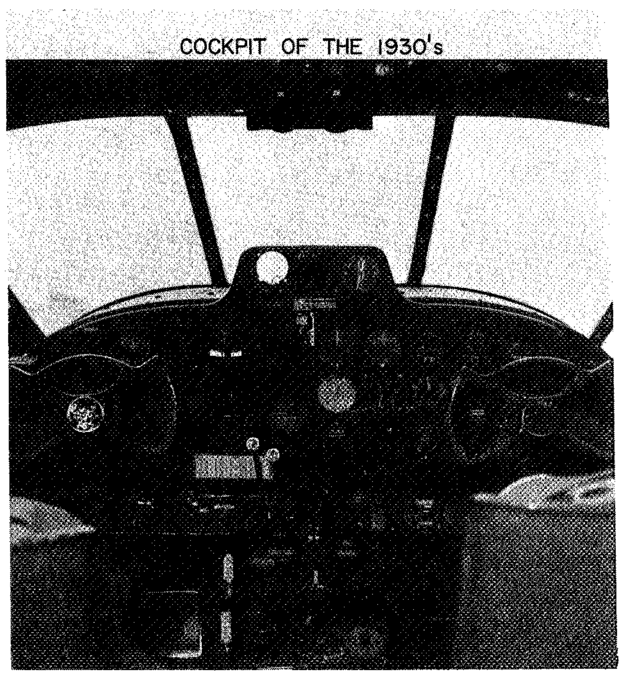

Figure 2

COCKPIT OF THE 1960's

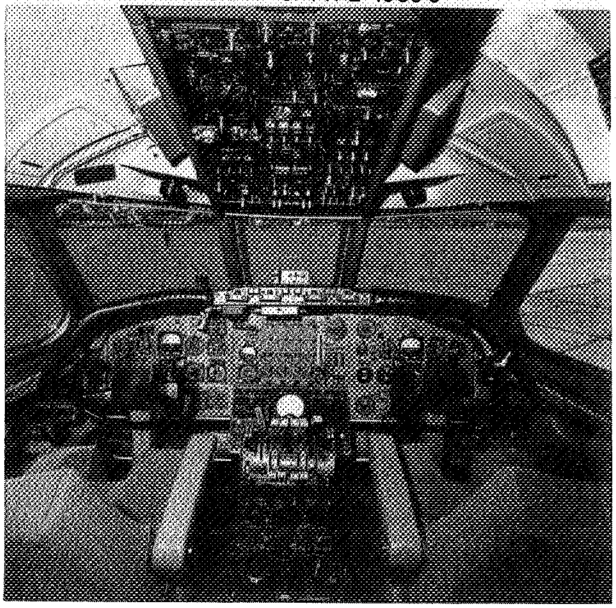

Figure 3

SIMULATOR WITH ADVANCED DISPLAY CONCEPTS

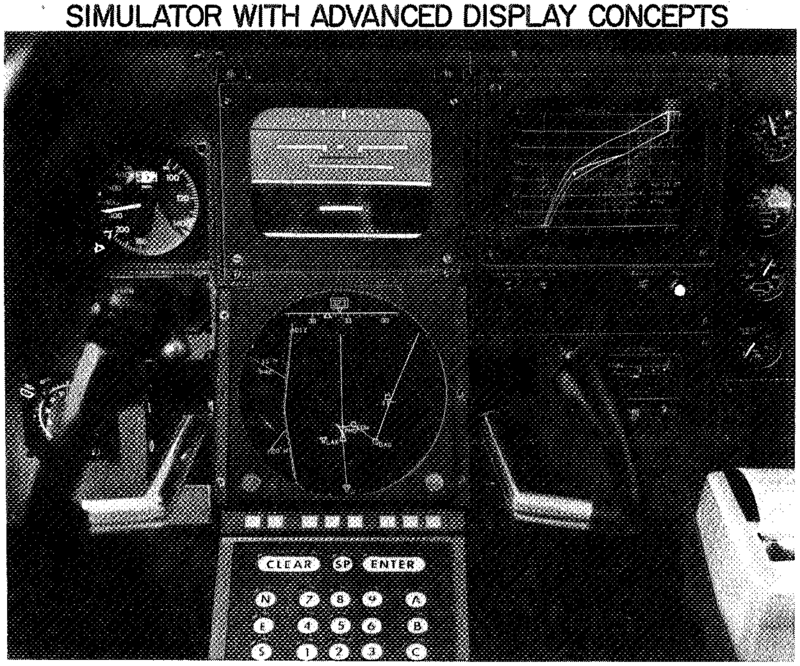

Figure 4

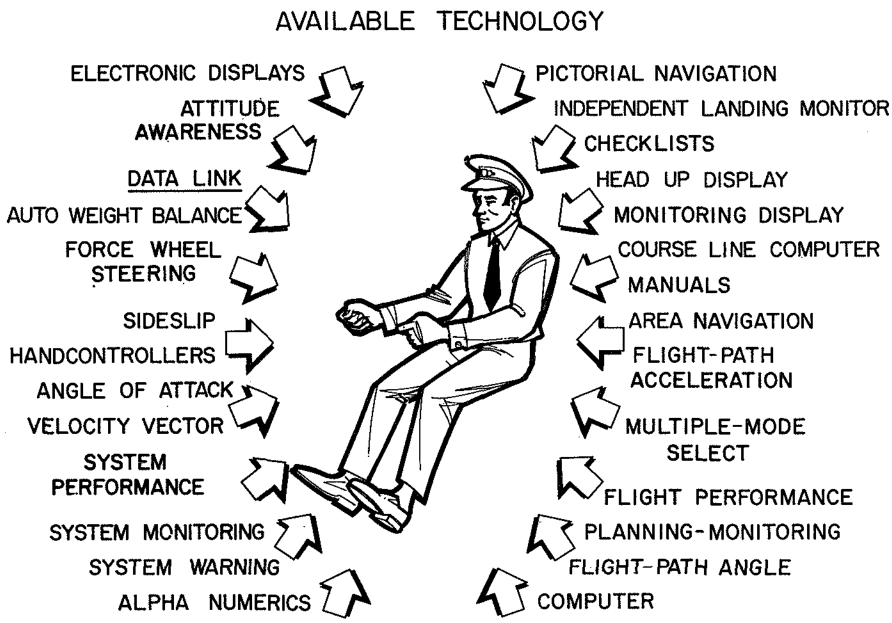

Figure 5

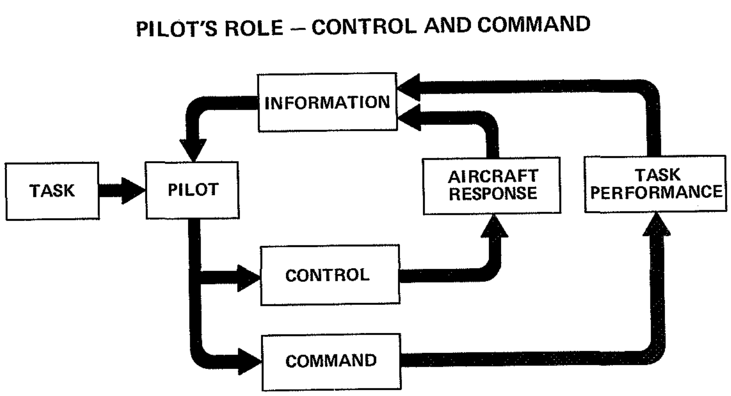

Figure 6

ADVANCED RESEARCH SIMULATOR

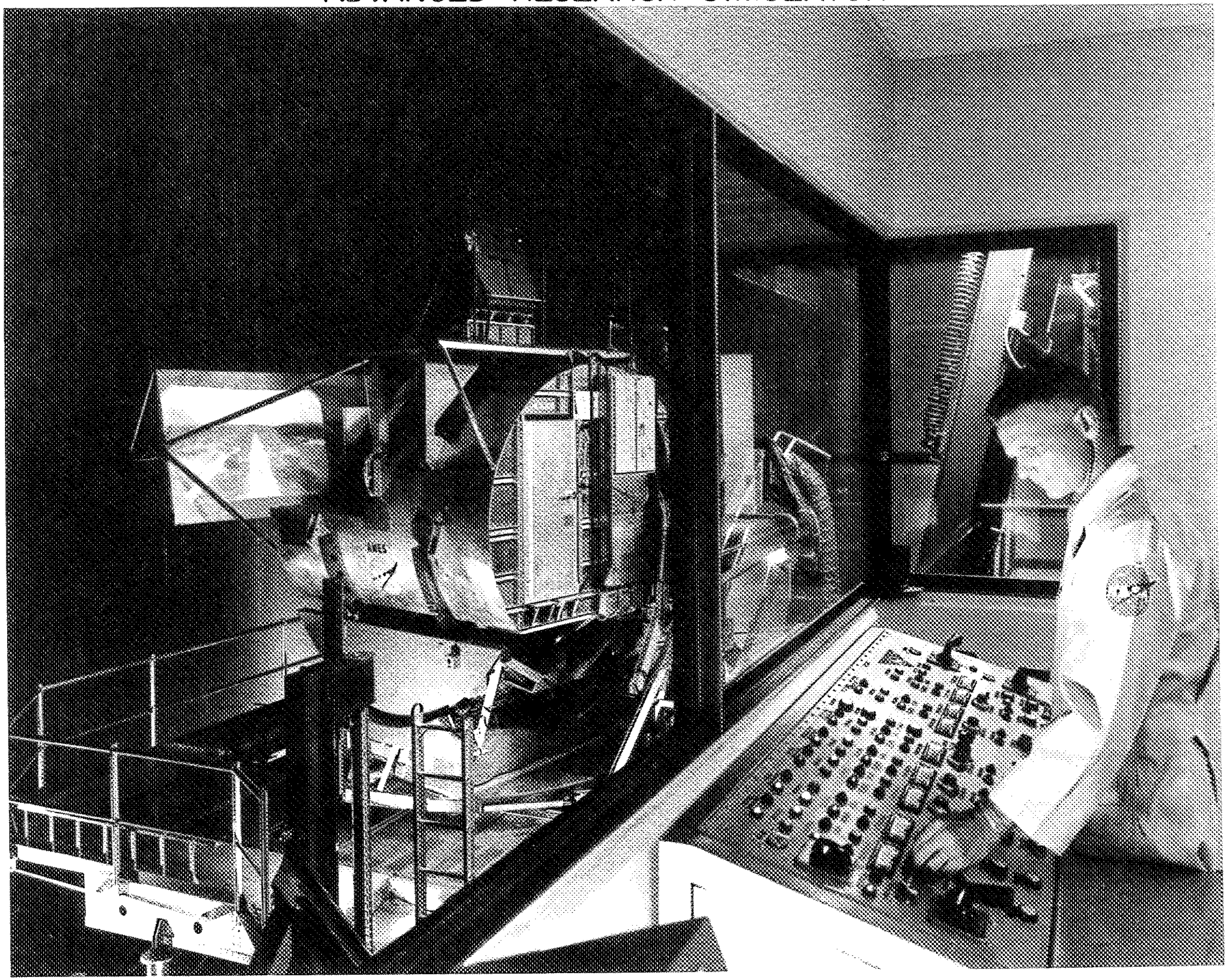

Figure 7

ADVANCED RESEARCH SIMULATOR (INTERIOR VIEW)

Figure 8

ADVANCED TRAINING SIMULATOR

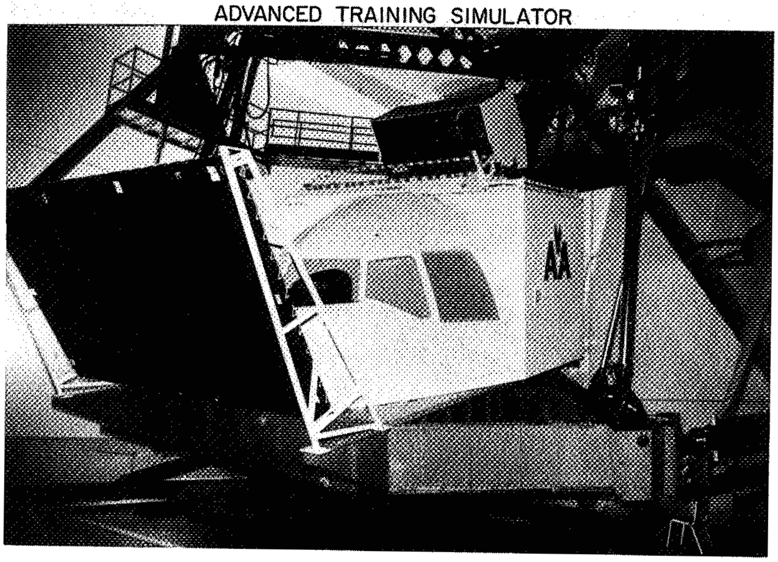

Figure 9

FLIGHT SIMULATION
A TOOL FOR GETTING TECHNOLOGY INTO THE COCKPIT

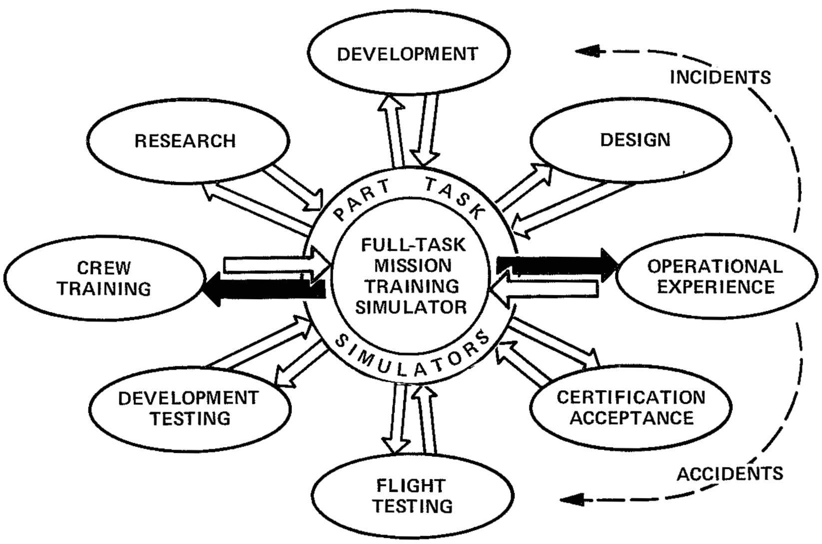

Figure 10

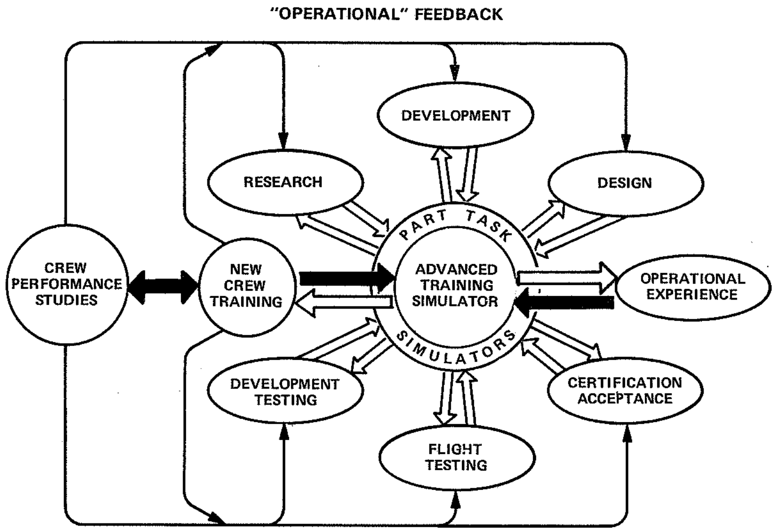

Figure 11

SIMPLIFIED CONTROL AND FLIGHT MANAGEMENT FOR THE 1980's

Figure 12

ADVANCED AVIONIC SYSTEMS

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INTRODUCTION

The development of avionics is illustrated by the sequence shown in figure 1, where four of the most recent items are underlined. Stability augmentation systems (SAS) and automatic landing equipment are incorporated in the latest series of transports to obtain the desired handling qualities and to improve operations during approach and landing. Area navigation equipment is available to provide flexibility in using the existing network of ground-based navigation stations. Also, a firm national commitment has been made for a new microwave landing system which will greatly improve the accuracy of landing guidance signals and permit maneuvering during approach. Expected future developments include the three items indicated at the top of figure 1:

(a) Computer-centered digital systems for both flight management and advanced control applications

(b) Automated communications

(c) Systems for wide-area navigation and surveillance

These developments will permit the application of advanced control methods which offer significant gains in vehicle performance. Also, they will be important in meeting the needs for automation in the future air-traffic environment, as well as in achieving efficient crew performance with increasingly complex airborne systems. This paper will present a brief assessment of the potential for these advances which appear so essential for the future.

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COMPUTER-CENTERED SYSTEMS

In considering future systems from the piloting or flight-management viewpoint, it is clear that computer-centered systems will be essential in performing a number of functions. Some of the more important items are the following:

CTOL, STOL

- Precise route-time flight profiles
- Flight director with path angle and acceleration
- Climb and descent envelopes
- Noise abatement approaches
- Pictorial navigation display
- Air traffic situation data
- Status of complex systems

VTOL

Time-critical, complex control during approach
Minimize powered-lift energy
Precision, low-altitude navigation

In the case of conventional- and short-take-off-and-landing vehicles (CTOL, STOL) many of the functions result from the need to fly precision flight paths – either to meet air traffic control requirements in a congested environment, to remain within desired performance envelopes during climb and descent, or to control approach paths to minimize noise. Of course, computation is important to derive flight-director information with both the path angle and acceleration data needed to fly precision paths. Also, computer-driven displays are needed for pictorial navigation displays, air traffic situation data, and effective monitoring of complex systems.

Vertical-take-off-and-landing (VTOL) aircraft present even more demanding requirements. Here, manual methods are simply not adequate for the time-critical, complex control task during approach and landing under instrument conditions. Automatic systems also are needed to minimize powered-lift energy for operating economy and to perform precision low-altitude navigation.

One possible arrangement of flight-management displays which was studied recently by The Boeing Company is shown as figure 2. The displays are largely computer driven with direct crew control of the system. Most observers agree that at least two displays are needed, each having performance equivalent to present cathode ray tubes. The advantages include effective display integration, the ability to use pictorial and graphic formats, mode selection suited to the phase of flight at hand, and high accuracy. Also, the computer provides flexibility for changing operating procedures should this be required.

An example of a typical display mode is shown in figure 3. This figure shows a terminal-area situation, including adjacent traffic with its identity, airspeed, and altitude; the relative position of the destination airport; the projected flight path of the aircraft; and the time error from the desired position. Of course, such a display would probably require processing and data transmission from cooperating ground equipment. It could, however, be of considerable value in achieving efficient operation along with high levels of safety in the future.

The next subject for consideration is computer-centered systems for use in electronic flight controls. The potential for advanced active-control systems is well recognized, since these appear to offer significant gains in vehicle performance. Included are controls for stability augmentation systems, maneuver-loads alleviation, flutter control and modal suppression, improved handling qualities, and ride smoothing.

A detailed discussion of control applications is not within the scope of this paper; however, it is recognized that the use of electronic controls for these applications would involve sensors, actuators, and computing equipment with much commonality. When this situation is considered along with the needs imposed by future flight management concepts, one general conclusion can be reached: Digital techniques appear to be the best approach for both advanced electronic controls and computer-centered flight management. This is not to say that all implementation must be accomplished by using digital techniques, since there may be cases where this is not desirable. Rather, digital techniques permit the use of computing methods which can provide the required power, precision, and reliability. Also, digital methods are more suitable than analog techniques for accurate data transfer in multiple-loop control systems.

ADVANCED DIGITAL TECHNIQUES

Digital techniques are being used in an increasing number of aircraft applications, and sound progress has been made toward reliable systems; however, much additional work will be necessary before systems can be designed to meet future requirements.

First, it is necessary to apply advanced control theory to the digital design problem, rather than simply using an analog design and applying digital techniques by means of high sampling rates. Thus, methods for sampling-time analysis, model following, and parameter identification which are suited to digital implementation must be developed. It will be important to simplify the type and number of computer operations wherever possible so that reliability problems remain tractable.

Sensors and actuators must be developed to interface efficiently with digital processing elements that employ redundancy and self-checking features. Also, the processing system, or logic configuration and its software, must be designed to provide the reliability associated with accident-free flight times of a million hours, or longer. Digital component developments will also be important; however, advances have been dramatic in the past and it is not expected that this will be a critical factor.

An advanced system concept is indicated in figure 4. The key element is a fault-tolerant computer which has access to information from almost all the sensors on the aircraft while permitting the crew inputs and displays which are needed for each phase of the flight. Much of the routine communications with the ground would be handled by digital data link. With a full digital approach, it should be possible to reduce the number of sensors and other input elements from that which would be required in conventional designs with high levels of redundancy. This could be accomplished with isolation methods which permit sharing of raw sensor data among several redundant paths through the computer. As indicated in figure 4, some form of distributed processing would be required for efficient error checking and redundancy interfacing at the sensors and actuators.

Designing such a system with distributed processing and a fault-tolerant computer implies that the system would remain operational even with multiple component failures. Extensive status monitoring would be provided for both the hardware and software and, in addition, a separate form of monitoring would be used for certain critical functions. This is referred to as "independent function monitoring" and takes the form of a parallel, but completely independent, assessment of system performance to verify that the system is providing the responses which would normally be expected from the disturbance inputs which exist.

The approach with regard to ultrareliable computer configurations and software can be further outlined as follows:

Multiprocessors and distributed arrays for redundancy and self-reorganization

Error detecting and correcting codes for arithmetic functions

Language redundancy and semantic checks

Dynamic error control and fault masking

Multiprocessors and distributed arrays would be used for one level of redundancy and self-reorganization in the event of component failures. Error detection and correction codes would be used for arithmetic functions as well as in data transfers. Language redundancy and semantic checks would be provided in the software. The system would be designed for dynamic error control and fault masking. The intent is to achieve a high degree of parallelism with rigorous diagnostics and "inspection-type" status monitoring which provides continuous assessment of the "health" of the system. The objective is to achieve system designs which are essentially free from the effects of failures.

The digital components which are needed will be considered next. Recent advances have been dramatic and much of the basic solid state technology is now in hand, although further improvements will undoubtedly be important. Some of the component advances that can be expected are as follows:

Programmable monolithic arrays

Cellular designs for self-reorganization
Integral diagnostics, processor, input/output, control, memory

Memories, bulk storage

1000-bit semiconductor chips
300-nsec cores
Plated wire, laminated ferrites
Bubble domain (10^2 bits/mm^2)
Holographic pages (10^4 bits/mm^2)

Interface elements

LED multicolor arrays
Beam penetration color CRT
Liquid crystal displays

Programmable monolithic arrays will permit designs with self-reorganization capability and a high level of part reliability. Both random-access memories and bulk-storage devices with the proper characteristics will become available. These include solid-state elements with a large scale of integration, high-speed cores, and either plated wire or laminated ferrites with low power requirements. Magnetic bubble domain units or holographic "page" systems will provide bulk storage which is extremely compact and reliable. Also, the interface elements for display and information transfer will be available. These include light-emitting diode (LED) arrays, beam penetration color cathode ray tubes (CRT), and liquid crystal devices. In addition, it is expected that variable intensity, plasma cell displays will be available which should interface easily with digital systems.

Research progress leaves little doubt that most of these digital component advances will be achieved very soon. For example, figure 5 illustrates recent advances in electronic devices. On the left is an example of monolithic solid-state logic using a large scale of integration. It is now possible to integrate almost all the functions needed for programmable logic arrays into a single element. This results in much greater reliability than in equivalent designs based on discrete components or a smaller scale of integration. Shown on the right in figure 5 is a portion of a magnetic bubble domain storage element which might be used to provide extremely compact mass memory systems to replace tape and disk units.

Figure 6 presents, in summary form, the history of computer-component characteristics during the past few years. It can be seen that cost, power, and volume have been reduced significantly while reliability has improved by more than an order of magnitude. If these trends continue, even at reduced rates, there should be no question regarding the future of onboard computers as an integral part of avionic systems in civil aircraft.

As a final step in considering the reliability of digital systems, figure 7 illustrates the potential of advanced approaches to system reliability. This figure indicates the estimated complexity of advanced designs as compared with conventional designs using straightforward redundancy and voting logic, and large numbers of discrete parts as the level of redundancy is increased. The advantage of advanced designs results from fault-tolerant parallel logic with a large scale of integration on each monolithic element. This reduces the complexity as far as parts are concerned and permits redundancy to be employed at more "levels" in the system design.

While much work lies ahead, especially in ultrareliable designs and the application of modern control theory in these designs, it is apparent that digital techniques offer the potential needed in the future. It should be possible to achieve practical implementation of the flight management and advanced electronic controls that are required.

COMMUNICATIONS

It is generally recognized that new methods must be employed to remove the limitations inherent in the present voice system. Digital data links are expected to handle routine air traffic control and operational messages, with voice communications being used only infrequently. The requirements include variable data rates to accommodate needs both en route and in terminal areas. Short access time, error detection, and wide coverage are necessary. It is estimated that the system in the United States will have to be designed to handle communications with a total data rate of about 10 million bits per second. Of course, this design problem must be handled by the Department of Transportation and FAA.

A number of design approaches are possible; however, it is likely that a system will be evolved to use both satellite- and ground-based elements, as illustrated in figure 8. In this example, communications for oceanic traffic and most long-range flights over the United States would be handled by satellite-based links. The following communications technology advances should make this feasible:

Satellite transmitter powers of 2 to 5 kW

Directive multibeam satellite antennas (25 to 30 dB)

High power L-band aircraft transmitters (100 W)

Directive aircraft antennas (5 to 10 dB)

Economic digital decoding and error detection

Developments are underway which should provide satellite transmitter powers of 2 to 5 kilowatts, and directive multibeam antennas can be designed. In the aircraft portion of the system, L-band transmitters at power levels of approximately 100 watts are needed, as well as directive antennas. These should become available along with economic methods for decoding and error detection.

It is believed that systems can be designed to handle 1000 to 5000 aircraft from a satellite with good reliability and freedom from multipath effects. The development of automated operational concepts is more of a systems design problem than one limited by component technology.

WIDE-AREA NAVIGATION AND SURVEILLANCE

While the present VOR area navigation equipment and the upgraded radar systems will provide a measure of improved performance in the immediate future, it is clear that further advances are needed. Again, this is a matter that involves the future airspace system and responsibilities of the Department of Transportation.

The capacity is needed to handle on the order of 50 000 aircraft simultaneously over the United States with accuracies of about 300 meters (1000 feet) in plan position. Update times of 10 to 100 seconds are needed during en-route flight, depending on aircraft speed and the method of using onboard inertial systems. Higher response (2 to 10 seconds) is needed in the terminal area. Some of the techniques for advanced position-measurement systems are as follows:

- Advanced radar networks
- Multifunction one-way "time-frequency" measurements
- Multilateration with two or more stations
- Satellite ranging and interferometry
- Ground-based hyperbolic grid systems
- Hybrid approaches

It is not within the scope of this paper to discuss all these methods; however, the ground-based hyperbolic-grid technique can be considered as an example of what may be possible. Figure 9 indicates the coverage which results from a very-low-frequency (VLF) hyperbolic grid with long baselines such as those used in the Navy OMEGA system. It can be seen that a favorable grid structure results over a large geographic area. For example, the insert shows an area about 100 nautical miles square. In this area the line crossings are nearly orthogonal, which permits accurate position determination. Some of the key features of such a system are the following:

Continuous navigation without frequency changes or station transitions

Coverage at all altitudes

Differential techniques for accuracies of 200 to 300 m (600 to 1000 ft)

Simple course line computation

With such a system, continuous navigation would be possible without frequency changes or station transitions, and the ground wave propagation mode would provide coverage at all altitudes. This is an important advantage as compared to the present VOR system, which requires many stations and provides only limited coverage at low altitudes because of high terrain and other obstructions. Differential techniques should permit accuracies on the order of 200 to 300 meters (600 to 1000 feet), and simple course-line computation is an inherent advantage. However, it should be recognized that there are a number of unresolved questions about such a system. Research efforts are underway on the following major items:

Interference, precipitation noise

Aircraft antenna design

Automatic signal acquisition

Digital phase-locked loops

One additional comment is that the hyperbolic-grid approach appears well suited for direct use by slower aircraft (100 to 200 knots). However, onboard inertial equipment may be necessary when such a system is used by high-speed aircraft because of response-time limitations resulting from the processing of extremely narrow band, low-frequency signals.

CONCLUDING REMARKS

This paper has been intentionally limited to a broad assessment of the future advances in avionic systems, and it has been necessary to omit discussion of a number of important points. Perhaps the most important point which has received no mention is the fact that future concepts must be compatible with <u>all</u> the aircraft that form the air traffic environment. This includes the air carriers, the military, and the entire spectrum of general aviation aircraft. System concepts must provide the high performance needed by the air carriers while permitting compatible designs which are economically feasible for general aviation and suitable for military use. This is especially important in the design of future airborne collision-warning systems and navigation, communications, and surveillance systems.

In regard to the avionic systems advances which have been discussed, it is apparent that future design efforts must properly consider the aircraft configuration and structure, as well as the entire air traffic control and communications system. The development of the proper systems concepts will be of utmost importance. To accomplish this it will be necessary to maintain an effective working relationship between the aircraft industry, the users of civil aircraft, and the Government agencies with responsibilities in research, development, and implementation.

With the development of advanced digital techniques, it will be possible to design a new class of flight-management and electronic control systems. Thus, two general conclusions are reached:

1. An effective systems approach is essential
2. Computer-centered digital systems can yield:
   (a) Improved vehicle performance and operation
   (b) Automation for efficient air-traffic management

As a final remark, it should be noted that the pacing items for equipment design are the development of advanced control theory and highly reliable digital configurations and software. From an overall standpoint, timing will be greatly influenced by the schedule for improvements to the national airspace system.

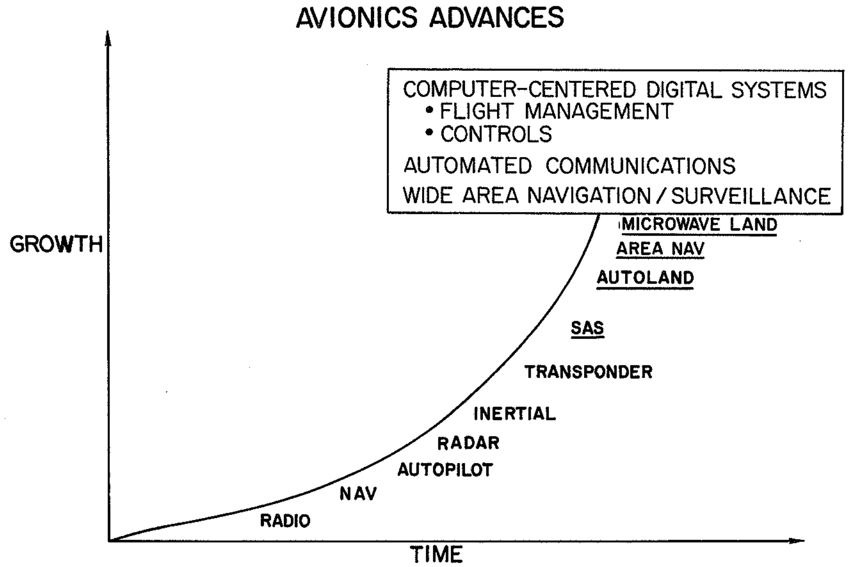

Figure 1

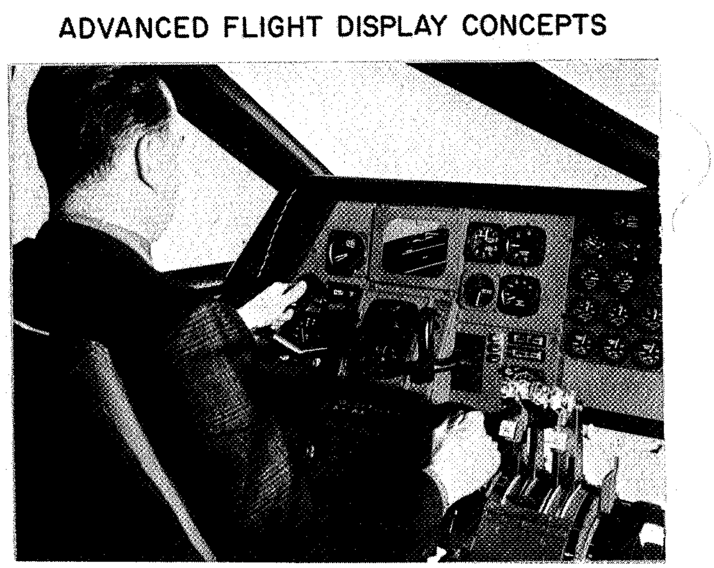

Figure 2

TERMINAL AREA SITUATION DISPLAY

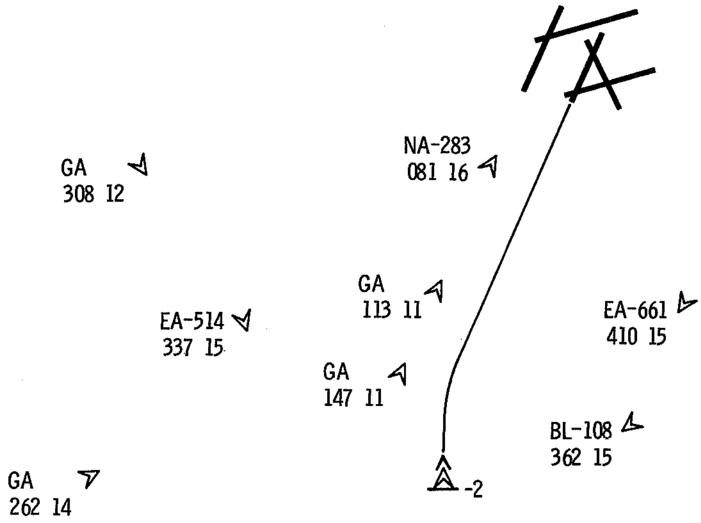

Figure 3

ADVANCED SYSTEM CONCEPT

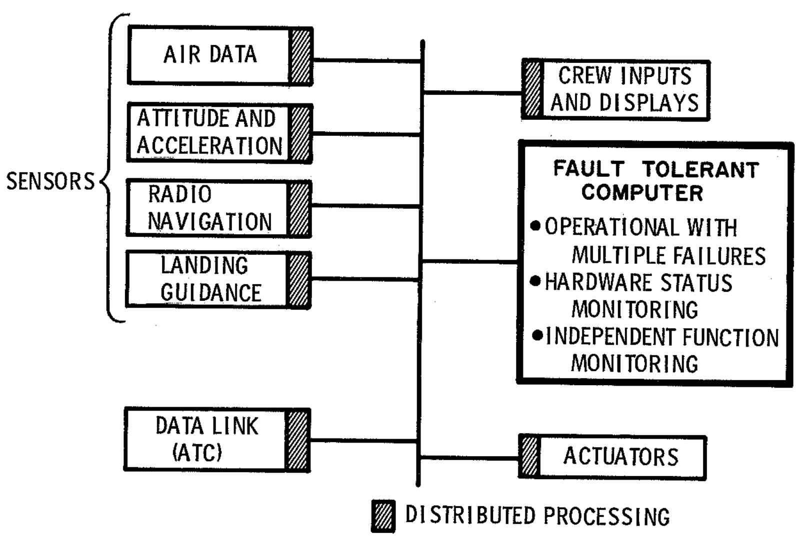

Figure 4

EXAMPLES OF ADVANCED COMPONENTS

PROGRAMMABLE LOGIC ARRAY

BUBBLE DOMAIN MASS MEMORY

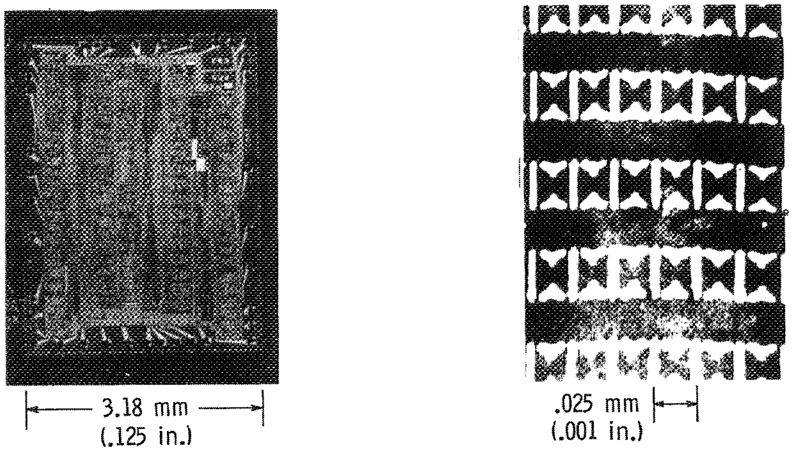

Figure 5

TRENDS IN COMPUTER CHARACTERISTICS

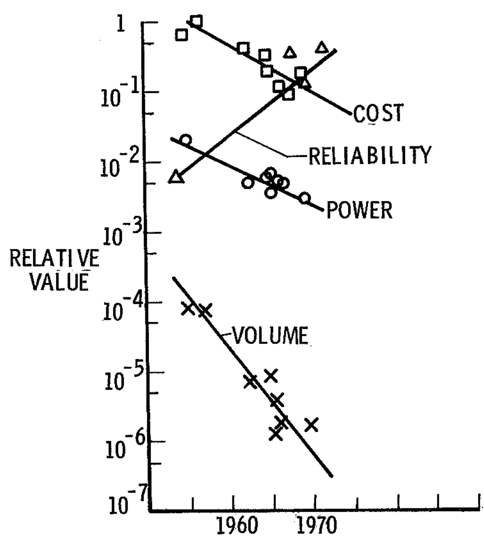

Figure 6

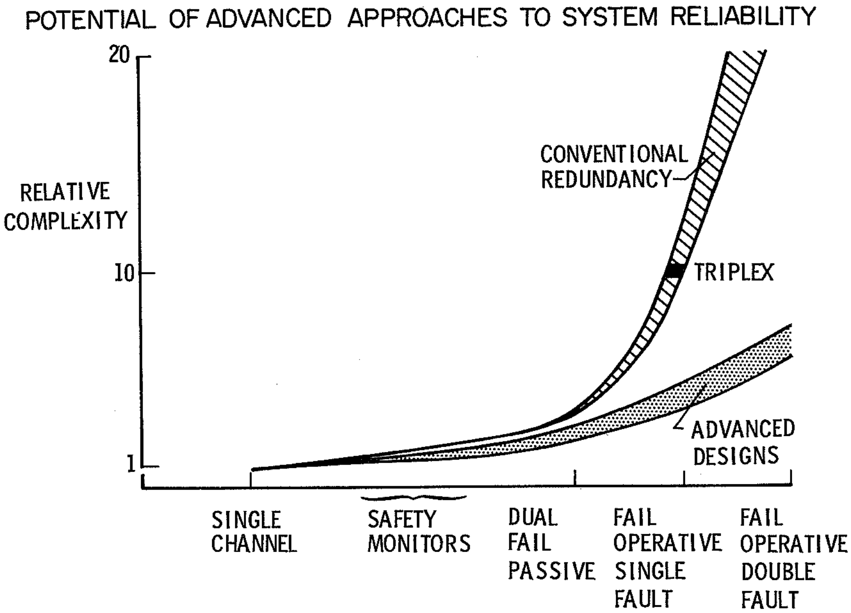

Figure 7

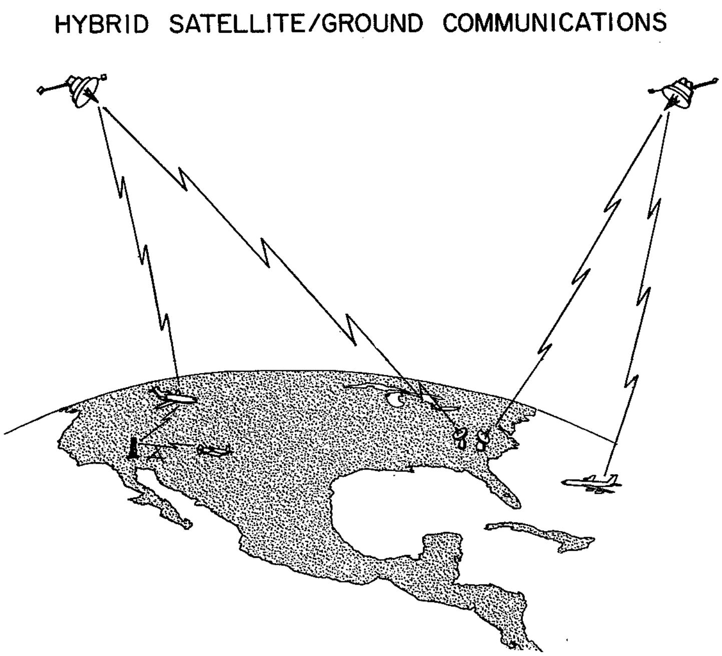

Figure 8

VLF HYPERBOLIC GRID COVERAGE

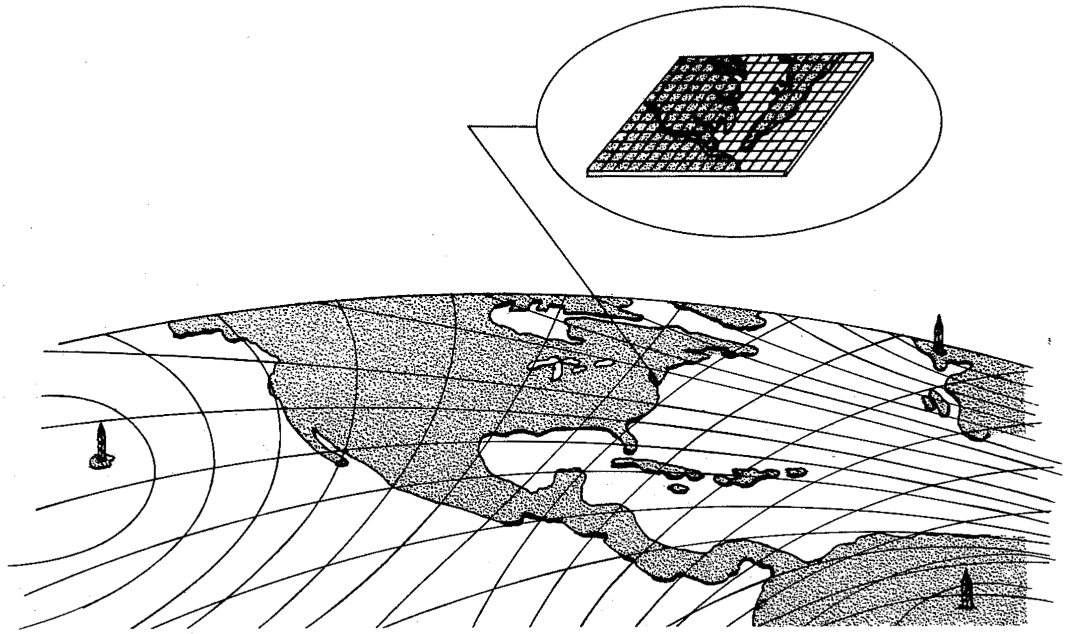

Figure 9

TRENDS IN AIRCRAFT NOISE ALLEVIATION

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INTRODUCTION

Noise produced by aircraft has become a major concern of the public, government, and industry. The recent joint DOT-NASA Civil Aviation Research and Development Policy Study (ref. 1) highlighted aircraft noise abatement as the national problem deserving the highest priority for solution. The elements of the problem are displayed in figure 1. Noise is a pressure fluctuation, emanating from a source, that is transmitted through the atmosphere or a structure to a receiver. The principal sources of aircraft noise are engines, rotors and propellers, boundary layers, and the sonic boom. Elimination or reduction of noise at the source is the most obvious approach to noise abatement, and other papers in this compilation report on some of those efforts. Still other papers report on new operational procedures, such as the steep approach, to reduce noise exposure. Such procedures reduce engine noise by operating at reduced power while also lengthening transmission paths of the noise through the atmosphere so that less noise reaches airport communities. Flight or ground structures can also be used to shield passengers or the public from direct impingement by the noise that cannot be eliminated at the source. In any event, a problem exists when noise reaching a receiver causes annoyance (people) or degraded performance (people or structures). This paper will concentrate on the effect of noise on people and structures and will also touch on noise reduction from rotary-wing vehicles.

COMMUNITY NOISE

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The first problem discussed is that associated with aircraft noise in the airport community. This problem has been forcefully brought to everyone's attention in recent years and has impacted all phases of civil aviation. Figure 2, based on data from reference 2 and related research, shows the extent of the community annoyance problem. The percent of people in the airport community that are highly annoyed by aircraft noise is plotted against the percent of the total airport community population that have actively complained to local authority. These data were collected for NASA in airport neighborhoods of seven large cities and two smaller cities by statistical techniques that involved several thousand interviews. They show that a lot of people are highly annoyed before complaints become numerous. For example, for those interviewed, over 25 percent of the total airport neighborhood population is upset over the noise before 5 percent begin to complain officially. In some of the larger cities, nearly everyone in the airport community is already highly annoyed. The message is simple – the actual situation in the community is worse than indicated by the number of complaints! The real problem is to alleviate the annoyance of a large part of the community – not to stop the complaints of a relatively smaller group of people.

The physical and psychological effects of noise on people has only recently begun to be understood. Research into the effects of noise on learning, on doing jobs of various complexities, on sleep interference, and on mental health is now underway and will lead to greatly increased understanding of community noise effects in the next decade. This research can be expected to lead to more definitive noise abatement criteria and to more stringent noise abatement requirements. Figure 3 illustrates the present FAR 36 certification requirement (ref. 3) for sideline, approach, and take-off noise for civil aircraft of different gross weights. Due to the severity and anticipated persistence of community noise problems, probable noise goals for civil aircraft designed to operate in the 1980's are expected to be 10 to 15 EPNdB below the present requirements for certification, as suggested in reference 1. This goal is consistent with present projections of noise reductions that advanced technology can deliver over the same time period.

Reductions in noise exposure always come at some cost. Therefore, every approach to community noise abatement must be exploited to its maximum extent in order to meet these goals with minimum economic impact. The system approach to community noise alleviation is illustrated in figure 4. The technology development to reduce noise sources on the airplane has already been mentioned and will continue to provide the biggest payoffs. Land use planning to keep residential communities out of the intense noise exposure area is a trade between economics, service to the passenger, and tolerance by the community. Operational procedures that reduce community noise exposure by flight-path control have been in use for several years but are limited in use by safety constraints. New operational approaches such as steeper descents, as well as safety considerations, are discussed elsewhere in this compilation.

Figure 5 illustrates a recent innovation in operating procedures called the dynamic preferential runway system that is now in limited use. The idea is to spread the noise exposure in the airport community by changing operating runways and thus minimizing annoyance. The system can provide short-term community noise alleviation and consists of a computer-aided guide for runway utilization that accounts for community noise exposure conditions. (The inset curves show data from ref. 2 indicating that annoyance can be determined as a function of noise exposure and demonstrate that criteria for noise exposure can be established for specific communities.) The factors considered in the computer prediction of noise exposure include the types of operation (take-off or landing) and local community characteristics such as population density, weather, time, and day of the week. Preliminary experience with this system indicates a significant reduction in complaints received.

Overall, the community noise problem can be expected to remain as a severe constraint on civil aviation. Every possible approach – reducing the noise produced, new operating procedures, and increased understanding of the response of people to noise – must be exploited to help the system.

PASSENGER ENVIRONMENT

Airline passengers make up the second largest group of people exposed to aircraft-generated noise. In general, the passenger environment has been easier to control than community noise. The trend of interior overall noise levels for cruise flight is shown in figure 6 and is based on data from sources such as references 4 to 7. A great deal of judgment goes with this collection of data since exceptions can be found for every case, depending upon the specific aircraft interior design, location in the passenger compartment, and so forth. However, the generalizations are believed to be valid. Note that onboard equipment noise, such as comes from the air-conditioning system, is well below the overall interior noise level, an indication that boundary-layer and power-plant noise predominate in determining the passenger environment. The highest interior noise levels are associated with general aviation aircraft, about which more will be said later, and generally comes from propeller and engine-exhaust noise. The same sources (propeller and exhaust) predominated in the propeller era of civil aviation. As is always the case in noise alleviation, when one noise source is suppressed, another source becomes dominant. In jet transports, boundary-layer and engine noise from turbomachinery and the jet became dominant. Indications are that the SST passenger noise environment initially will be higher than that in current subsonic jets, due to the character of noise transmission through a typical structure excited by a supersonic boundary layer (ref. 8). However, the goal must be to bring these noise levels down to present levels by applying better design approaches. Although not shown in figure 6, STOL vehicle passenger environment is expected to be equivalent to that of present-day jets in cruise flight but will have the interaction between the propulsion system and the structure as a new noise source during low-speed flight.

The penalty for acoustic insulation to control interior noise in commercial aircraft is illustrated in figure 7. The weight of acoustic treatment, as a fraction of take-off weight, has been approximately constant with time. Again, considerable judgment went into establishing this trend due to the difficulty in determining just what should be charged to acoustic insulation. Continuation of current practice will require this parameter to increase to control SST interior noise during cruise and the low-frequency flow-interaction noise expected from STOL vehicles during powered-lift flight. The goal of research, however, is to cut the weight penalty in half through better understanding of the noise source – subsonic and supersonic boundary layers and flow interaction – and by applying improved and optimization techniques to reduce the noise transmitted through the structures such as discussed in reference 9.

The effectiveness to be expected of better noise control design approaches is illustrated in figure 8. These are one-third-octave-band sound pressure levels inside a commercial airplane cruising at a Mach number M of 0.85 at an altitude of 9450 meters (31 000 feet) reported in reference 10. The noise transmitted through the untreated skin is seen to be markedly reduced when treated with damping tape or rubber wedges. Damping tape, much like friction tape but heavier, can be applied directly to skin panels and reduces noise transmission by damping panel response. The wedges are shown installed in figure 9 and reduce noise transmission through two mechanisms – changing panel boundary conditions and adding damping. The wedges are especially effective in the audible frequency range and reduce overall sound pressure levels by about 10 dB.

GENERAL AVIATION INTERIOR NOISE

The highest interior noise levels in civil aircraft occur in general aviation aircraft. Figure 10 presents one-third-octave-band sound pressure levels from 11 light twin-engine aircraft with reciprocating engines in cruise flight at approximately 75 percent power from data of reference 7 to illustrate the seriousness of this problem. The band represents the spread of maximum noise levels that are generated principally by the propeller and engine exhaust. Also shown is a damage-risk-criterion curve for a 2-hour exposure from reference 11.

This criterion states that sound pressure levels less than this curve can be tolerated without hearing damage for exposures of not more than 2 hours per day for 5 days. However, exposures at any frequency at higher levels or for longer times can possibly lead to hearing damage. Obviously, some general aviation aircraft do not provide adequate acoustic protection. Further evidence of the seriousness of this problem is that 50 percent of all general aviation pilots have measurable threshold hearing shifts after 10 years of flying, as reported in reference 12. Since there are over half a million general aviation pilots and passengers in the United States, the magnitude of the problem is also evident. Technology can reduce these interior noise levels to more reasonable values, as has been done in commercial aircraft, but the economic incentive has been lacking. However, the shift of public attitudes for consumer protection and environmental control will probably lead to legislation requiring acoustic protection in general aviation aircraft if the industry does not adopt adequate standards.

An example of improvement in acoustic performance that can be achieved through the application of available technology is illustrated in figures 11 and 12. An experimental muffler was designed and tested on a general aviation helicopter. The noise from this particular vehicle is dominated by exhaust noise from an unmuffled reciprocating engine.

This nonoptimum installation cost about 20 kilograms (45 pounds) of weight and about 3 to 4 percent loss in horsepower. The immediate result was a reduction of cockpit noise to a level permitting direct voice communication between pilots. In addition, flyover noise was reduced by about 8 EPNdB. The frequency spectra measured during hover show the muffler to be very effective in reducing exhaust noise, as indicated by the disappearance of the large spikes in the spectrum, and to uniformly reduce the higher frequency noise. However, the low-frequency rotor noise now predominates. In addition, a high-frequency tone associated with the engine cooling fan becomes very objectionable. Therefore, further noise alleviation on this helicopter can come only by reducing noise from these two new sources.

ROTARY-WING NOISE REDUCTION

The rotor noise just mentioned leads into the subject of noise generated by rotary-wing vehicles. Figure 13 illustrates the trend of noise produced by these vehicles and is based on material such as references 13 and 14. Noise has been increasing about 10 dB per decade due to increased size and performance of vehicles. The increase comes from designing without noise constraints and would continue unless noise requirements are included in future designs. However, available technology can reduce rotary-wing vehicle noise by about 15 dB. These noise reductions can be achieved by cutting tail and main-rotor noise through decreases in rotation speed and by increasing the number of blades to reduce blade loading. Increased performance demands will still continue to drive noise upward. Therefore, the goal of advanced technology must be to reduce rotary-wing vehicle noise further by 10 dB per decade to hold overall noise at presently achievable levels. Current research on the fundamentals of blade noise generation, effects of tip shape, rotorcraft operating procedures, and rotor vibrations promise to deliver this kind of noise reduction in the future.

AIRCRAFT NOISE REDUCTION LABORATORY

Increased understanding of the fundamental aspects of aircraft noise generation, transmission, and response must be developed if the advanced technology goals and projection discussed in this compilation are to be achieved. Figure 14 is an artist's concept of a new facility for basic and applied research on aircraft noise reduction that is now under construction at Langley Research Center. It will be fully operational in less than 2 years and will provide major new capability for research. Table I illustrates the capabilities of the laboratory. Across the top are acoustics problem areas, starting with noise sources, continuing with noise propagation, and ending with the human and structural response areas. Along the side is indicated the vehicle end item to which the technology is applicable. Solid circles indicate a prime new capability of the laboratory, partially filled circles show capability to supplement other facilities, and open circles indicate limited capability. Of course, some areas exist for which the laboratory will have no capability. The large number of blocks filled in, however, shows that the laboratory will have great capability for research on rotor, jet, and flow-interaction noise and on human and structural response problems that will apply to most of the vehicles of concern. One wing of the laboratory will be devoted to simulating noise environments for human factors and acoustic physics studies. The other wing contains an applications area that has quiet flow facilities combined with an anechoic chamber and a reverberation chamber. This laboratory is intended to be the focal point for noise research and development within the government and is expected to support many joint NASA-University-Industry programs. Its product will be the technical base for optimum acoustic design.

CONCLUDING REMARKS

The theme of this paper has been that all aspects of noise reduction technology must be exploited to achieve the noise control that society is now demanding of civil aviation. A great amount of new technology development is required in order to continue to reduce aircraft noise. A major acoustics technology capability exists and is being expanded to meet these future needs.

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TABLE I.- AIRCRAFT NOISE REDUCTION LABORATORY

● Provides prime capability
◑ Will supplement other facilities
○ Has limited capability

Vehicles \ Research areas	Sources								Propagation				Reaction			
	Low-velocity jet	High-velocity jet	Rotating blades	Reciprocating-engine exhaust	Integrated-lift propulsion	Boundary layer	Equipment components	Sonic boom	Duct propagation	Duct acoustic materials	Flow choking	Atmospheric propagation	Complex structures	People outdoors	People indoors	People in flight
Conventional take-off and landing	●		●						●	●	●	◑	◑	●	●	
Short take-off and landing	●	◑	●		●				●	●	●	◑	◑	●	●	
Vertical take-off and landing	●		●	●					●	●	●	◑	◑	●	●	◑
Advanced transports	●		●			○			●	●	●	◑	◑	●	●	◑
Supersonic transports		◑	●			○		○	●	●	●	◑	◑	●	●	◑
General aviation			●	●								◑		●	●	
Military aircraft	●	◑	●	●	●				●	●		◑	◑	●		◑
Shuttle		○				○	●	○					◑	●	●	
High-speed ground	●		●			○			●	●	●	◑		●	●	

THE AVIATION NOISE PROBLEM

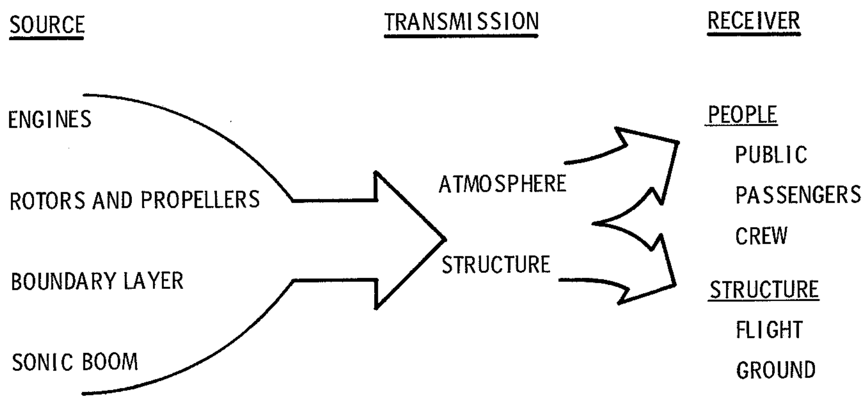


Figure 1

COMPLAINTS DUE TO ANNOYANCE

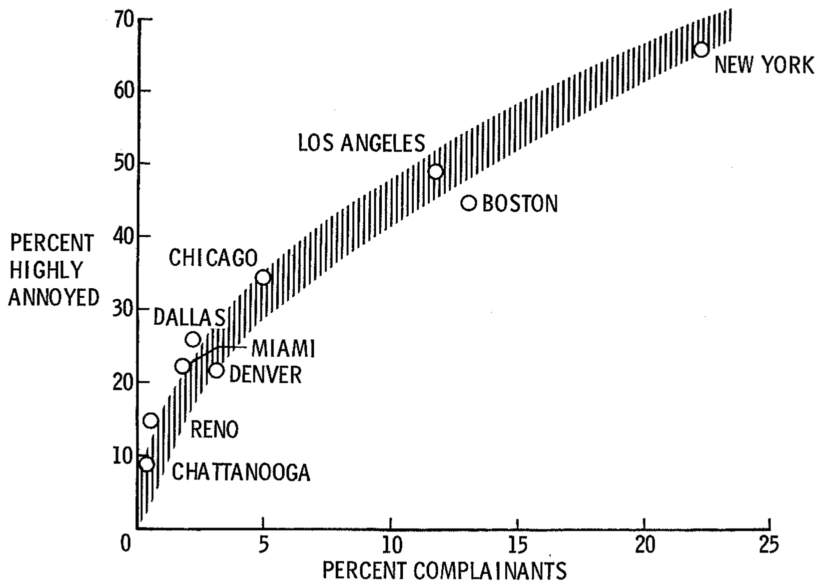


Figure 2

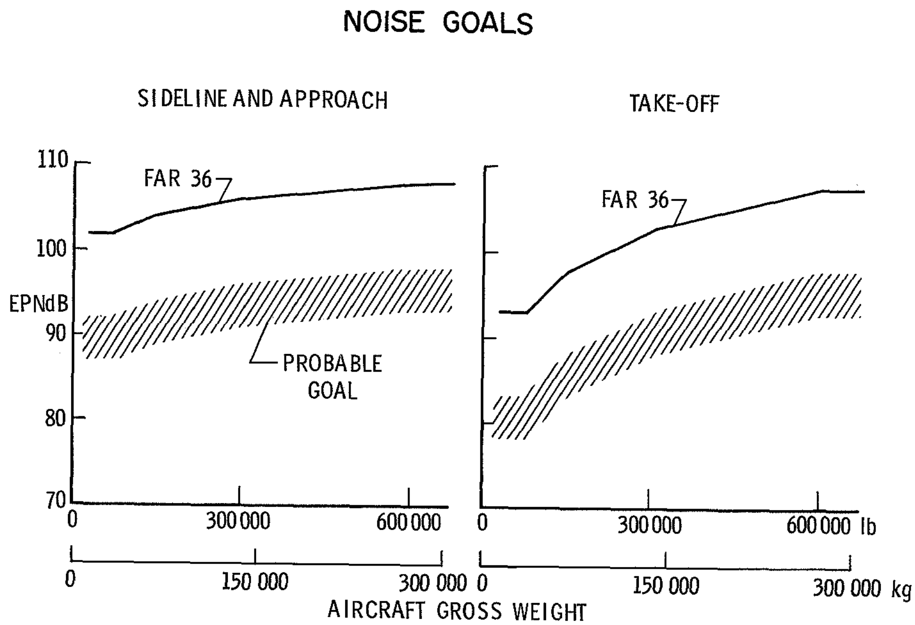


Figure 3

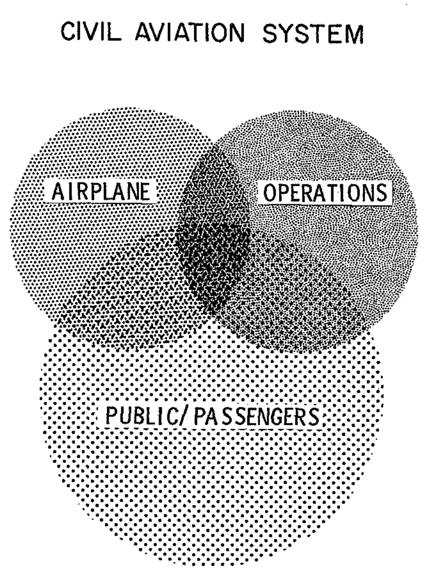
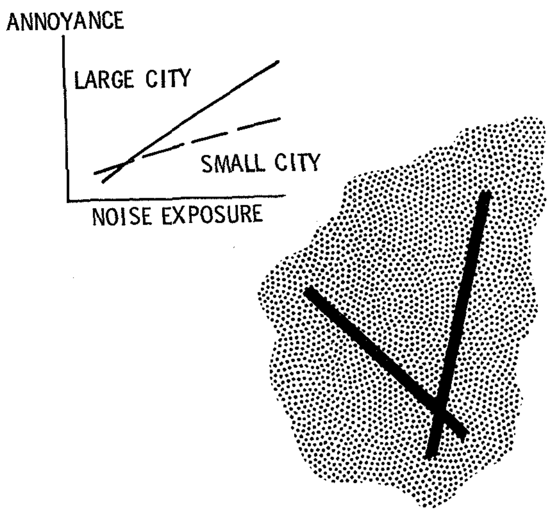


Figure 4

DYNAMIC PREFERENTIAL RUNWAY



OPERATING FACTORS

- TYPE OF OPERATION (TAKE-OFF OR LANDING)
- COMMUNITY NOISE EXPOSURE
- PREDICTED WEATHER
- TIME OF DAY AND WEEK
- DWELL TIME

Figure 5

PASSENGER ENVIRONMENT
CRUISE FLIGHT

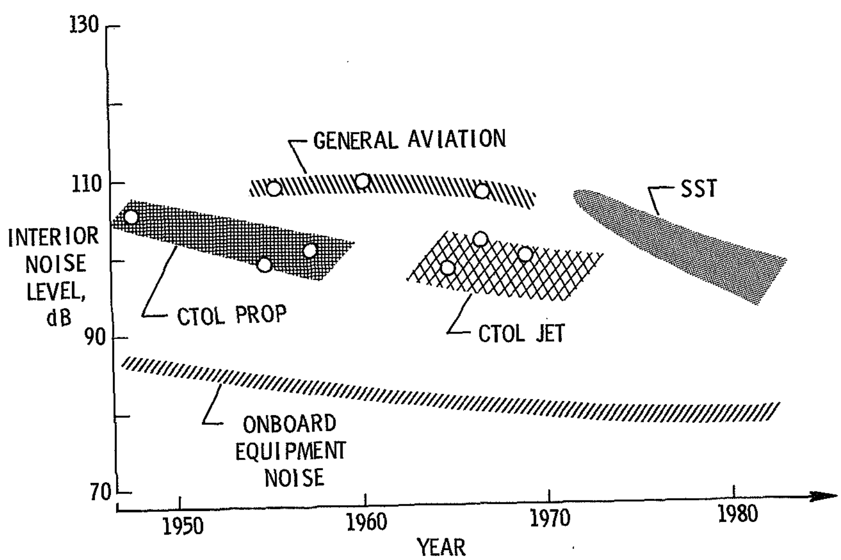


Figure 6

ACOUSTIC INSULATION REQUIREMENTS

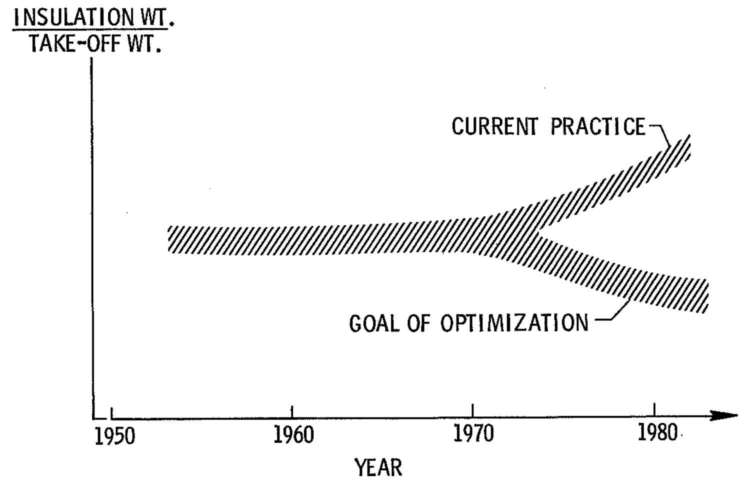


Figure 7

EFFECTIVENESS OF INTERIOR NOISE CONTROL MEASURES

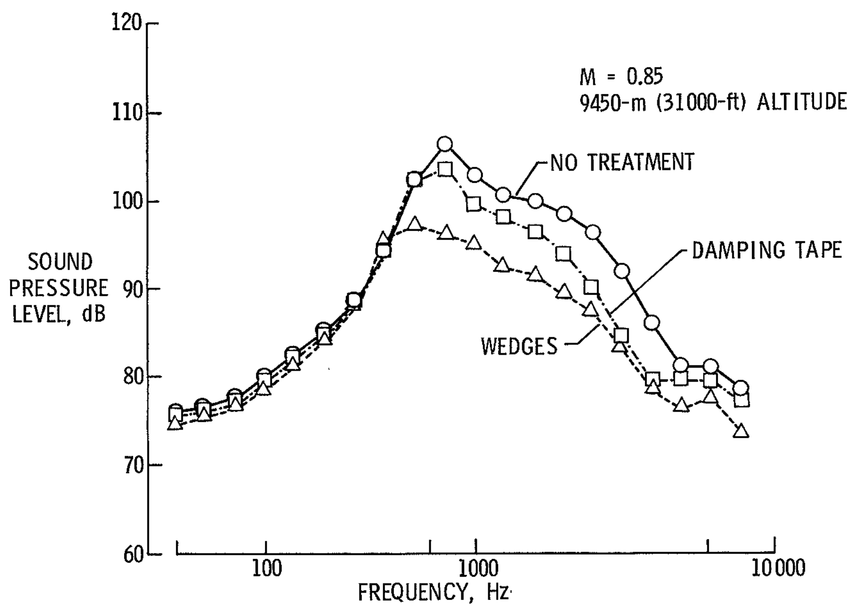


Figure 8

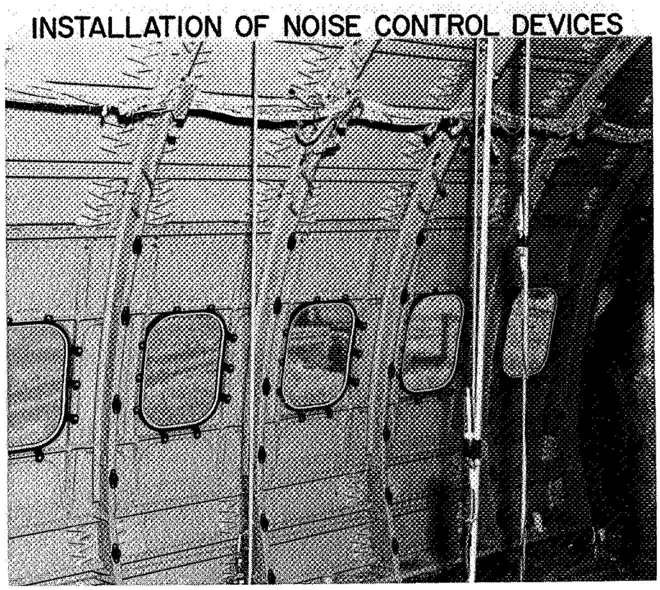


Figure 9

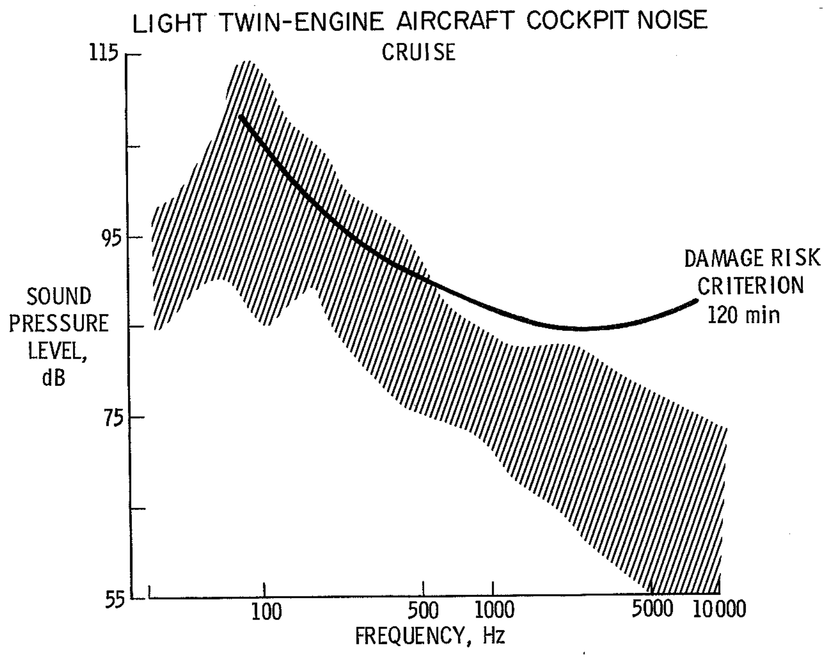


Figure 10

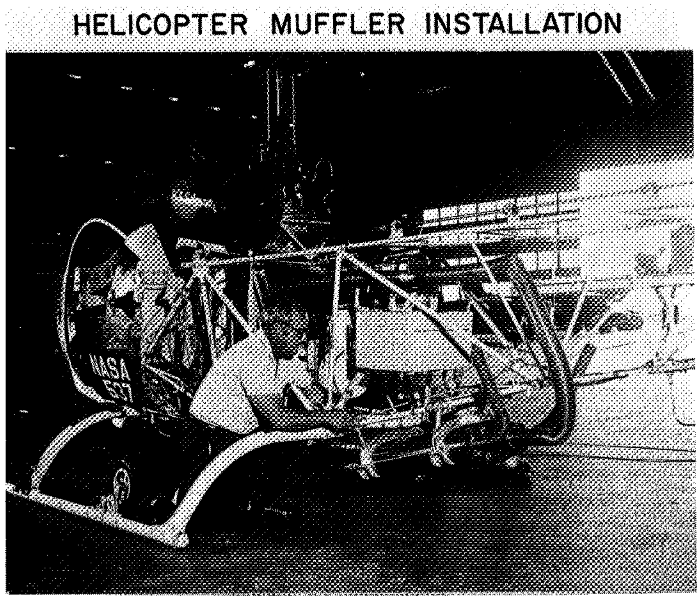


Figure 11

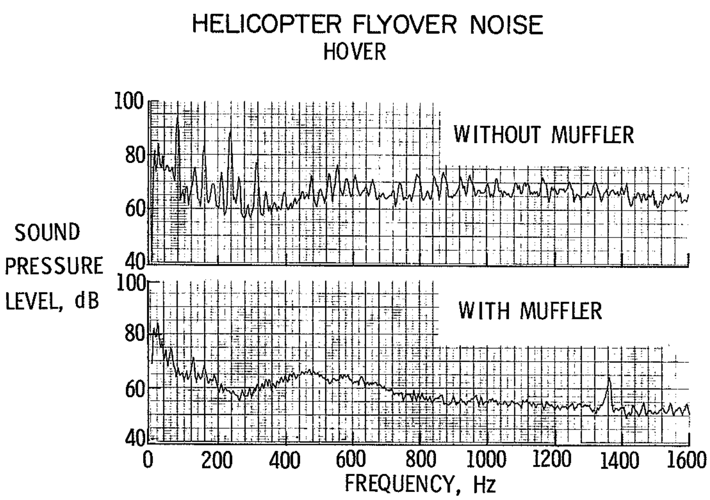


Figure 12

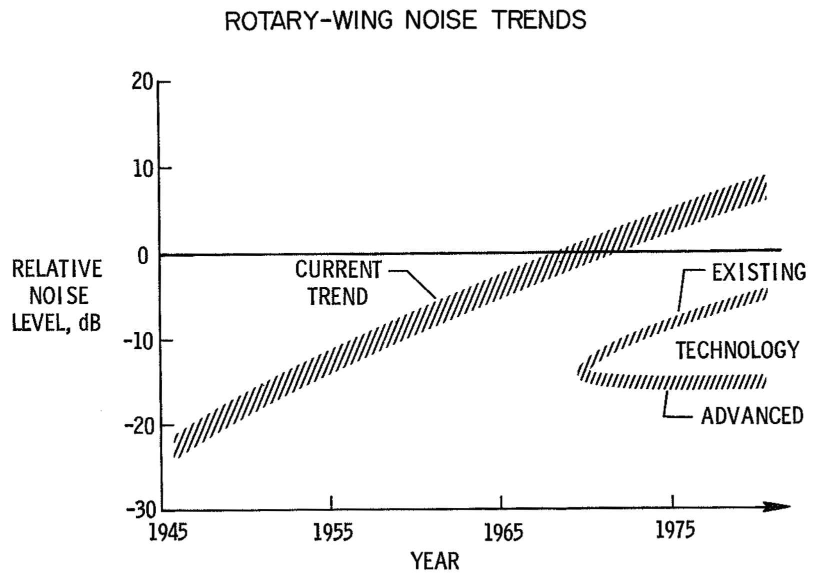


Figure 13

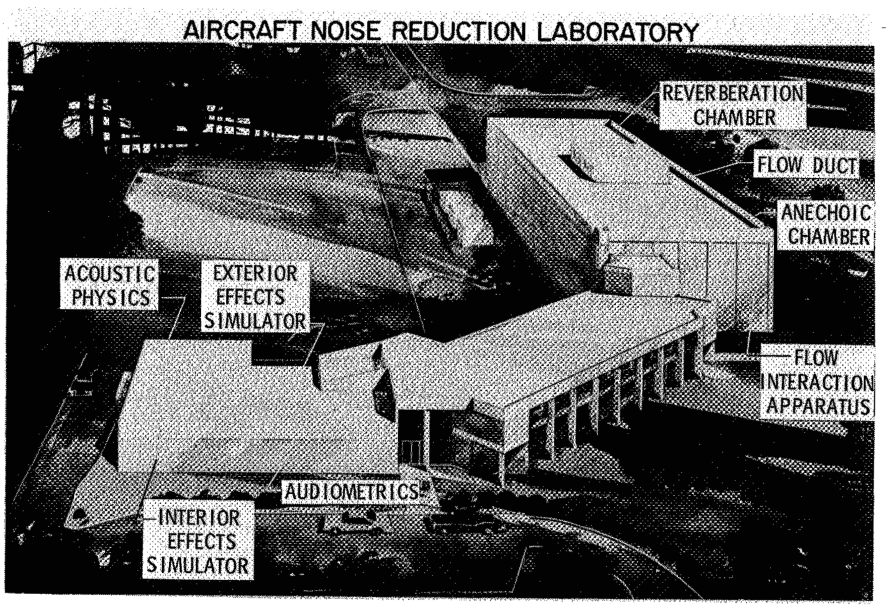


Figure 14

GENERAL AVIATION

THE SEVENTIES AND BEYOND

By M. R. Barber and Jack Fischel
Flight Research Center

INTRODUCTION

The term "general aviation" is primarily a definition of aircraft utilization. Since it includes everything from Cessna 150's to F-28 and DC-9 jets, it can hardly be construed as a classification of aircraft physical characteristics. Technically, general aviation is comprised of those elements of U.S. civil aviation which are neither certificated nor supplemental air carriers. It includes commuter airlines, private "pleasure" aircraft, corporate air transports, manufacturers of business and utility aircraft, unscheduled air taxi operations, and fixed-base operations.

This paper discusses the possible advancements in general aviation through the applications of technology during the next decade in terms of aircraft performance, utility, safety, and public acceptance. Reference 1 contains a much more complete treatment of many subjects discussed in this paper.

AIRCRAFT PERFORMANCE

As a measure of the current status of aircraft performance and cost, figure 1 (based on data from ref. 2) presents the consumer price, in 1971 dollars, of general aviation aircraft as a function of the product of cruise velocity and useful load. The figure includes all types of general aviation aircraft and shows that increases in aircraft performance are paid for at a relatively linear rate. The figure also shows distinct groupings for the three primary propulsion systems used in general aviation (that is, reciprocating engines labeled "PROP," turboprops, and turbine jets).

The progress of general aviation aircraft, in terms of performance, over the last decade is shown in figure 2 by using the same parameters as in figure 1 and superimposing data from reference 3 for the year 1962. The 1962 monies were corrected to 1971 values at a ratio of 1.35. The figure shows that essentially the only progress during the last decade was the advent of turboprop aircraft; the other types of aircraft remained in about the same economic-performance relationship. Comparison of these data with the same types of data for 1952 shows that jet-engine aircraft have been developed since that time, but that no significant changes have been made in the reciprocating-engine aircraft.

It should be noted that the performance index used does not take into account advancements that have been made in areas such as avionics, engine reliability, and aircraft accouterments. However, these types of advancements are more alined with refinements rather than with technological advancements or breakthroughs. The width of the band in the figure is probably a function of these refinements.

What then can technology be expected to provide through the next decade? Obviously, the main thrust should be toward improved economy, inasmuch as most general aviation aircraft are in the lower part of the performance curve of figures 1 and 2 and the sonic barrier with all its inherent problems and costs bounds the upper part.

Supercritical Airfoils

A technological development expected to have a significant effect on general aviation performance is the supercritical airfoil. Figure 3 presents the hypothetical cruise Mach number M_{CRUISE} (actually the drag rise Mach number) as a function of thickness ratio t/c for conventional and supercritical airfoils on straight wings. For the same thickness ratio, a supercritical airfoil can delay the drag rise Mach number by 15 percent. This effect could be used to produce a 15-percent increase in the cruise-velocity capability of those general aviation jet airplanes that now cruise at the drag rise. The figure also shows that for the same drag rise Mach number, supercritical airfoil sections can be used with a greater thickness ratio. Their use would make it possible to reduce wing weight or increase aspect ratio, or to provide volume for more fuel, and thereby to extend the range of the aircraft.

Supercritical airfoils could also significantly improve general aviation performance if applied to aircraft propellers. Figure 4, based on data from reference 4, indicates the sea-level static thrust of a typical general aviation aircraft which has a 196-kilowatt (250-horsepower) engine and uses a conventional 1.8-meter (6-foot), two-bladed, constant-speed propeller. The circular symbol indicates the maximum rpm capability of this particular aircraft. The propeller-tip Mach number at this point is 0.81. By assuming that supercritical airfoils would allow the propeller to reach Mach 0.93 before encountering the drag rise and that a supercritical propeller could be designed with the same subsonic aerodynamic efficiency as the existing propeller, a 15-percent increase in thrust with the same input horsepower could be achieved (square symbol, fig. 4).

The thrust increase is possible because with the same input horsepower the increased rpm facilitated by the supercritical propeller requires a reduced blade angle relative to the thrust axis. This reduced blade angle decreases the amount of lift translated into torque and increases the amount translated into thrust. Therefore, the magnitude of the performance increase should remain relatively constant for all variable-pitch propellers. The resulting increase in aircraft performance would be an increased gross weight or an increased rate-of-climb capability, since calculations for the cruise velocity indicate negligible thrust increases. It is possible that the performance increase promised by the supercritical propeller could be traded for noise reductions, since the delayed shock wave might result in a quieter operation.

Several factors make the application of supercritical airfoils to propellers attractive. They are

(1) The propeller tip operates at a relatively constant velocity.

(2) There would be no change in manufacturing tolerances.

(3) The method of manufacuring would remain unchanged.

Factors that are working against the development of supercritical propellers are

(1) Lack of aerodynamic data for thickness ratios conducive to propeller tips

(2) The tendency of the government to direct most of its research efforts toward the cruise transport application of supercritical airfoils

Structures

As indicated in paper no. 10 by Richard A. Pride, there is much interest in the aerospace industry in applying composite materials (primarily, boron and graphite) to military and commercial transport aircraft. The primary advantages of using these materials are reduced labor costs and weight saving, since the materials are amenable to compression or injection molding techniques and they have a higher strength-weight ratio than existing materials. Full utilization of composites could reduce the weight of an aircraft by approximately 20 percent. However, recently published projections (ref. 5) indicate that by 1980 the cost of graphite composites will average $40 a pound. When this cost is compared with $1 to $2 a pound for aluminum, it can only be concluded that composite structures will not be used in general aviation by 1980 unless a breakthrough in materials technology significantly reduces the cost of these materials.

Perhaps a more promising application of structural technology lies in using hand-laid sandwich-construction fiber glass. Several disadvantages of this type of construction have been hypothesized, such as greater weight, fatigue, stiffness, and higher labor costs than for conventional construction. However, the recent introduction of the fiber-glass aircraft (fig. 5) to the general aviation fleet has shown that most of these disadvantages have been overcome. The remaining question, that of high labor cost, cannot be answered conclusively until a production run can be established on the aircraft.

Performance increases of the order of 10 percent can be expected from the utilization of fiber-glass structures. With fiber glass, it is possible to produce more compound curvatures for wing-fuselage fairings and similar components without increasing manufacturing costs; also, in-flight aerodynamic contours can be maintained more precisely than with aluminum. The performance of fiber-glass gliders over the last few years would certainly indicate that a performance increase of 10 percent can be achieved. Virtually no metal gliders compete with fiber-glass gliders in the more serious competitive events.

Fiber-glass structure also would enable more high-lift devices to be used because the additional attach points, hinges, and tracks could be integrated in the molding process and would therefore not raise labor costs appreciably. A quieter cabin results with fiber-glass construction because of its improved noise-absorption characteristics.

Fiber-glass construction can also be applied to high-speed aircraft. Tests conducted by an aircraft manufacturer and a chemical company have shown that resin systems exist that would enable the construction of a fiber-glass aircraft with a flight capability of Mach 1.6.

Another structural improvement which will probably be used in general aviation aircraft in the near future is metallic bonding. This method is well proved in military and commercial applications, and light-aircraft manufacturers are beginning to use it. The aircraft shown in figure 6 incorporates a large amount of bonding. In general, a saving of approximately 3 to 6 percent in structural weight can be achieved through the use of bonding. In addition, labor costs are lower because it is faster than riveting.

Propulsion

The two most significant advancements anticipated in general aviation propulsion systems in the next decade are the rotary combustion engine and the low-cost gas turbine engine. The characteristics of these engines relative to present-day engines are as follows:

(1) Rotary combustion engines:
- (a) 30% lower cost
- (b) 50% less weight
- (c) Less pollutants
- (d) Less noise

(2) Low-cost gas turbine engines:
- (a) 75% lower cost
- (b) 25% higher fuel consumption
- (c) Less fuel cost than reciprocating engine
- (d) Less pollutants

The rotary combustion engine has already entered the automotive market in two makes of automobiles, and the largest automobile maker recently purchased manufacturing rights. In addition, one light-aircraft manufacturer is flight testing a rotary combustion engine. The simplicity of the rotary combustion engine is illustrated in figure 7. The rotor and rotor shaft are the only moving parts in the engine. The engine's simplicity should make possible a 30-percent initial cost saving over reciprocating engines in addition to a 50-percent weight reduction.

Because of higher engine rpm, the exhaust noises from rotary combustion engines are easier to attenuate and the smoother engine operation creates less vibration. In automotive applications, the rotary combustion engine already meets the emission pollution standards that were proposed by the Department of Health, Education, and Welfare for 1975. The low emissions are made possible by the use of an exhaust reactor. Therefore, the rotary combustion engine promises improvements in both economy and performance for general aviation aircraft.

The low-cost gas turbine engine is being developed by the Lewis Research Center by making design trade-offs in favor of cost at the expense of performance. Preliminary research indicates that a gas turbine engine could be manufactured at a cost of approximately $5 per pound of thrust. If a markup of 2 is considered, this cost would result in a consumer price of $10 per pound of thrust compared with approximately $40 per pound of thrust for existing turbine engines. This price amounts to a 75-percent reduction in initial cost. The engine would have higher fuel consumption than existing turbine engines, but would still result in a lower fuel cost per trip than a reciprocating engine of the same cruise-thrust capability. Because of lower exhaust temperatures, the engine would emit less nitrous oxide than current turbines. The emission of hydrocarbon and carbon monoxide would be about the same.

Generally, the cost of general aviation engines is about 20 percent of the total cost of the aircraft. Therefore, the cost reduction provided for conventional turbine-powered aircraft by the low-cost engine would be 15 percent. However, if the 20-percent engine-airframe cost ratio could be approximated in a less sophisticated aircraft, a reduction in cost of as much as 40 percent over existing turbine-powered models might be achieved.

The effect of technological advances on general aviation costs is summarized in figure 8. This figure shows the projected 1981 cost and performance of the three major classes of aircraft (reciprocating engine, turboprops, and jets) in terms of the percent of their 1971 cost and performance. Performance is again in terms of the product of cruise velocity and useful load. As shown, the largest improvement probably will be in the jet class of aircraft, primarily because of the cost reduction provided by the low-cost gas-turbine engine. A performance increase of approximately 25 percent should be provided by the supercritical airfoils in conjunction with fiber-glass or bonded-metallic structures. In all, this performance increase would be a significant improvement over the turbine-powered aircraft of today.

The cost of turboprop aircraft should also be reduced greatly as a result of the low-cost turbine engine; however, the reduction will not be as large as that provided for the pure turbine aircraft because of the need for propellers and a gear box. In addition, a 12-percent increase in performance should be realized through the use of supercritical propellers and fiber-glass or bonded-metallic structures.

A 15-percent reduction in the cost of reciprocating-engine aircraft should be realized with the use of the rotary combustion engine, and a 20-percent performance increase should result from the use of supercritical propellers, the use of fiber-glass or bonded-metallic structures, and the weight decrease provided by the rotary combustion engine.

The cost-performance gains depicted in figure 8 represent the expected maximum available through the next decade. Probably some of the gains will be traded for improvements in aircraft utility and safety and greater public acceptance.

AIRCRAFT UTILITY AND SAFETY

Avionics

The cost and time required for a nonprofessional pilot to obtain and maintain a functional instrument flight rules (IFR) capability is obviously a deterrent to many who would otherwise use general aviation as a mode of transportation. To increase the utilization of this class of aircraft, the complexity surrounding IFR capability must be reduced.

Figure 9 indicates, in part, why current IFR requirements are so stringent. The figure shows professional pilots' Cooper ratings (ref. 6) of their ability to perform ILS approaches in a typical light aircraft as a function of turbulence intensity, ranging from light to moderate. The instrument landing system (ILS) approach task is a good measure of IFR handling qualities because it enables the pilot to evaluate aircraft stability, control harmony, and display efficiency in a unified manner, rather than discretely. The Cooper pilot ratings indicate that even in smooth air, handling-qualities improvement is desired, and in turbulent air improvement is required for satisfactory ILS approaches. Obviously, if professional pilots believe such improvements are necessary, general aviation pilot training levels must be high enough to compensate for the needed improvements. A detailed discussion of the aircraft characteristics on which these ratings were based is presented in reference 7. In that study advanced displays and control systems were evaluated to determine their effects on alleviating IFR handling-qualities problems. Included was a flight-director display of the type currently used in airline operations. It consisted of a command steering display that utilized the existing VOR, ILS, and DME radio signals. Among the advanced control systems evaluated, an attitude command control system provided the most significant improvement in the IFR handling qualities.

In simplified terms, an attitude command control system is a system in which aircraft pitch and roll attitude, rather than aircraft angle of attack and roll rate, are proportional to changes in the stick or wheel position.

The results of these evaluations are shown in figure 10, in which professional pilots' ratings for ILS approaches are again presented as a function of turbulence intensity. The flight-director display alone provides a significant improvement in smooth air, but its effectiveness decreases as turbulence increases. The attitude command control system provides the same improvement in smooth air that is provided by the flight director, but it is not affected by turbulence. Turbulence effects are compensated for by the gust-alleviation capabilities of the highly responsive control system that is needed to provide the attitude command characteristics. When the attitude command control system and the flight-director display are used together, the aircraft approaches perfection from an IFR handling-qualities standpoint.

Obviously, a system that elicits a Cooper pilot rating of 1.5 for a task as difficult as the ILS approach could reduce required piloting capabilities. The present cost of these types of systems is high. Hopefully, however, the decreasing costs and increasing miniaturization of electronic components will put the systems within economic reach of the small airplane owner. Because these types of control systems require additional equipment, such as actuators, for which the costs are not decreasing, they probably will follow the utilization of advanced displays.

With the advent of continuous navigation systems, such as the very low frequency system depicted in figure 11, the sequencing of a flight-director type of display could be greatly simplified. The pilot would not have to navigate in a tick-tack-toe fashion as required by current VORTAC navigation systems (fig. 11) but, rather, could navigate in a continuous fashion. By combining this type of system with the displays and control systems previously discussed, general aviation pilots would be able to perform complete IFR cross-country missions with greater ease. Piloting skills would not have to be greater than those necessary to complete a task with a Cooper pilot rating of 1.5.

Low-Speed Aerodynamic Devices

Landing accidents have historically accounted for greater than 50 percent of all general aviation accidents. The advanced control systems and displays previously discussed should significantly reduce these accidents. However, utilization of low-speed aerodynamic devices should help to alleviate these problems even sooner.

Figure 12, which shows the final approach speed of commercial jets, general aviation jets, general aviation conventional light aircraft, and light STOL aircraft as a function of wing loading, illustrates current utilization of low-speed high-lift devices in the final approach. The figure shows that jet transports, which have sophisticated high-lift devices, are utilizing lift coefficients C_L of the order of 1.5 but that they do not penetrate the unaugmented roll control boundary of $C_L = 2$. A small segment of light STOL aircraft is making approaches at the same C_L as the jet transports, but a larger segment of general aviation aircraft is not making maximum use of high-lift devices.

The primary effects of the high-lift devices on general aviation aircraft are not only a slightly reduced speed on approach but also improved handling qualities as a result of the lower angle of attack in the approach. The aircraft in the light STOL class, which are primarily modifications of aircraft in the light general aviation class, are becoming popular not only because of the improved STOL capability but also because of the improved low-speed handling qualities.

In addition to the improved characteristics provided by high-lift devices, tests recently conducted for NASA have shown that spoilers applied to a typical light aircraft as a means of improved energy management can significantly improve the ease with which these aircraft can be landed.

PUBLIC ACCEPTANCE

The general aviation industry may justifiably be optimistic about the future public acceptance of its product. A recent study by the Advanced Concepts and Missions Division, NASA Headquarters, showed that depending on annual utilization, the general aviation aircraft is today the most economical mode of transportation for 80.5- to 241.4-kilometer (50- to 150-mile) trips if a single traveler has a time value of $5 an hour or more. The study also shows that by 1982 this characteristic will be extended to 64.4- to 724.1-kilometer (40- to 450-mile) trips for a single traveler whose time value is greater than $1 per hour. When several travelers are considered, general aviation travel becomes even more attractive. However, if the volume of aircraft that should materialize because of these economic considerations is to become a reality, there must be marked improvements in the general aviation safety record. Presently, general aviation aircraft have a per-passenger mile fatality rate an order of magnitude greater than that for automobiles. These accidents are primarily attributed to pilot error, but aircraft must be made easier to fly. If they are not, the increased volume of trafic will amplify the problem. In addition to the previously discussed improvements in safety provided by advanced avionics, future use of improved structures, occupant shielding techniques, and fire retardant materials should help to reduce fatalities.

The noise caused by reciprocating-engine general aviation aircraft has not yet become a significant problem. Recent studies (ref. 8) have shown that the noise exposure 12.9 kilometers (8 miles) from the take-off end of an international airport runway is greater than the noise exposure 0.8 kilometer $\left(\frac{1}{2}\text{ mile}\right)$ from the take-off end of a general aviation airport with 200 000 operations per year. However, as current ecological efforts reduce surrounding noise sources, general aviation aircraft will be forced to reduce their noise output levels. Recent tests at the Langley Research Center showed that the noise levels of reciprocating-engine aircraft can be reduced significantly without paying large performance penalties. Large-diameter propellers driven from geared engines will alleviate the noise problem until supercritical propellers, rotary combustion engines, and "prop" fan systems give the aircraft designer more noise-alleviation capability. General aviation jet aircraft create more severe noise problems, but they will be able to use the research applicable to commercial transport aircraft to reduce the problem.

Currently, Environmental Pollution Agency (EPA) thinking indicates that general aviation will be forced to reduce their emission levels. Exhaust gas recycling devices in addition to improved fuel mixture control will probably be utilized to comply with the EPA requirements. The general aviation jet aircraft will inherit the systems currently being developed for commercial jet aircraft.

CONCLUDING REMARKS

The gains made in general aviation in the last decade were primarily performance gains provided by the turbo shaft and turbine engines. On the basis of current and projected near-future technology, it is believed that the main technological effort in the next decade will be devoted to improving the economy, performance, utility, and safety of general aviation aircraft.

The technology for significant improvements either is or will be available. The real challenge is in the economic application of this technology.

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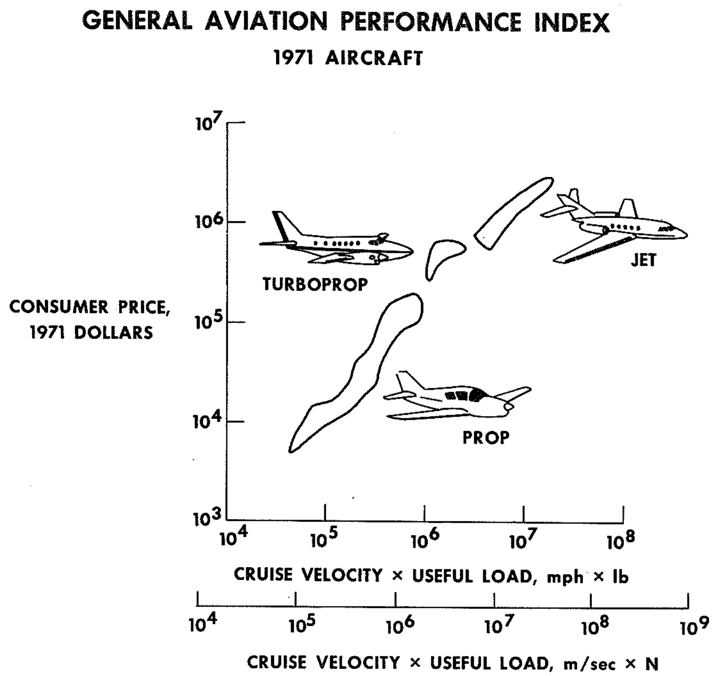

Figure 1

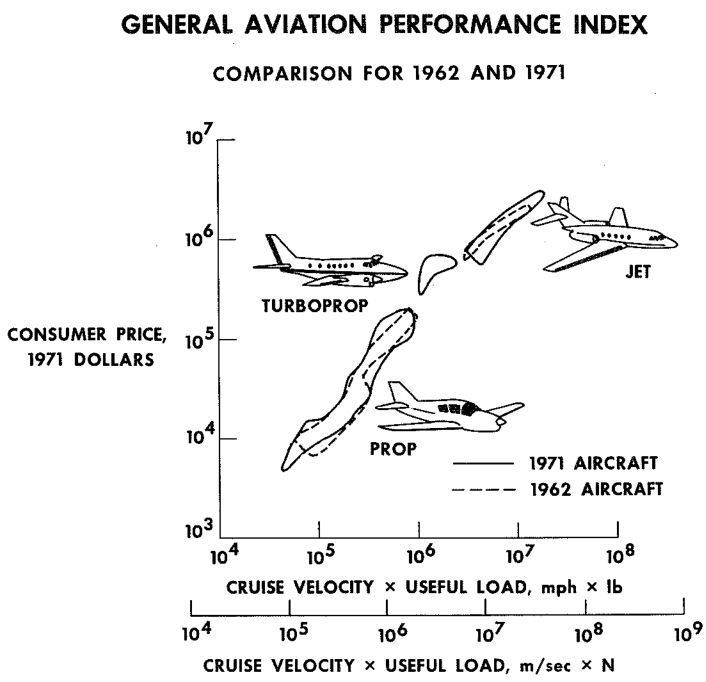

Figure 2

PERFORMANCE IMPROVEMENT WITH SUPERCRITICAL AIRFOILS

STRAIGHT WINGS

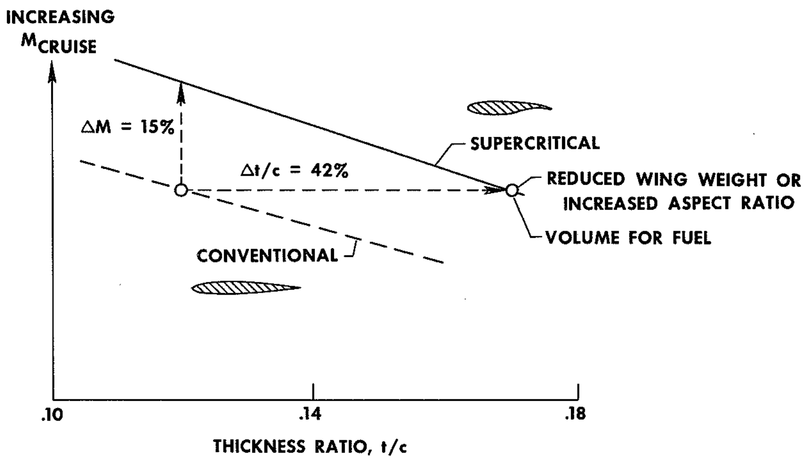

Figure 3

EFFECT OF SUPERCRITICAL PROPELLER ON STATIC THRUST

TYPICAL LIGHT AIRCRAFT CONSTANT SPEED PROPELLER

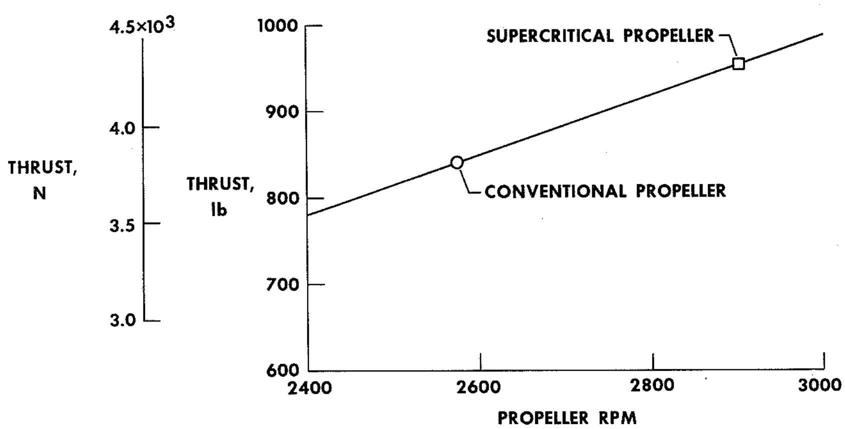

Figure 4

FIBER-GLASS GENERAL AVIATION AIRCRAFT

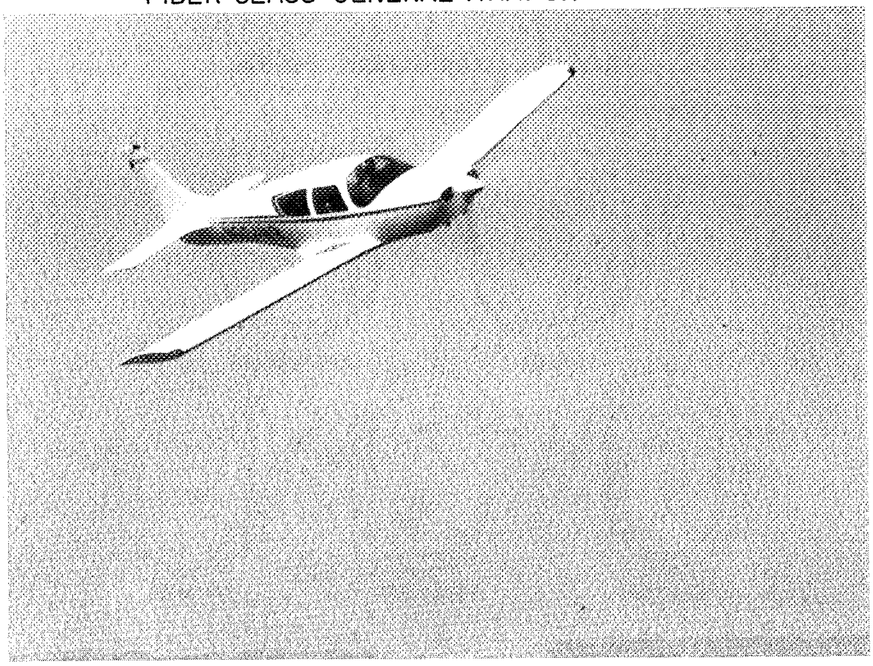

Figure 5

GENERAL AVIATION AIRCRAFT WHICH USES METALLIC BONDING

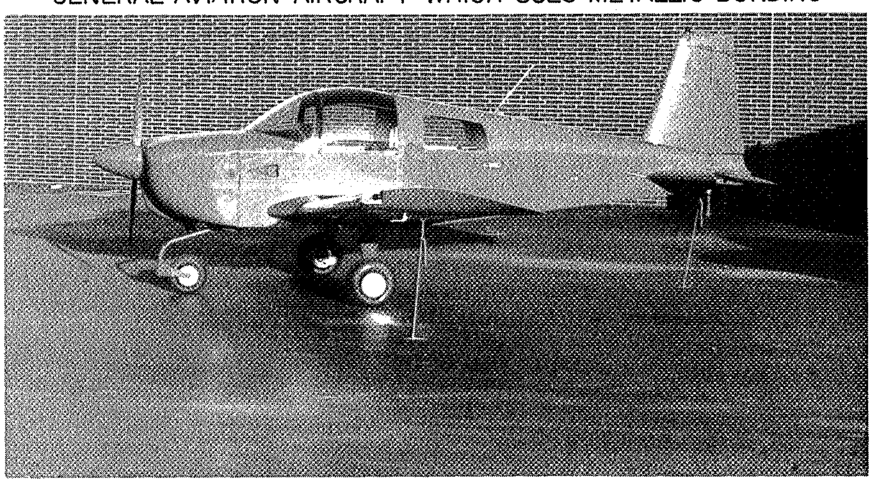

Figure 6

ROTARY COMBUSTION ENGINE

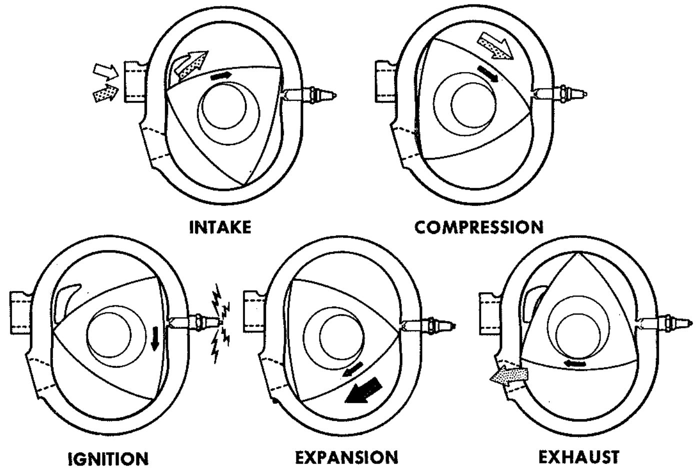

Figure 7

CUMULATIVE EFFECTS OF TECHNOLOGY ON GENERAL AVIATION AIRCRAFT

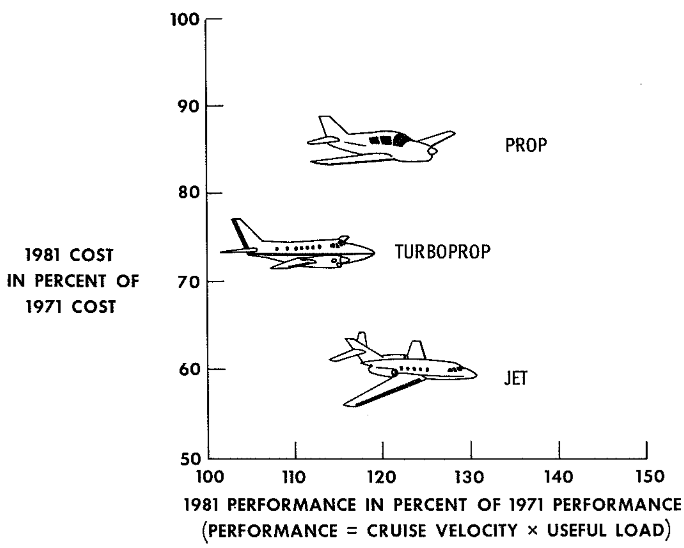

Figure 8

PILOT RATINGS OF ILS APPROACH TASK

TYPICAL LIGHT AIRCRAFT

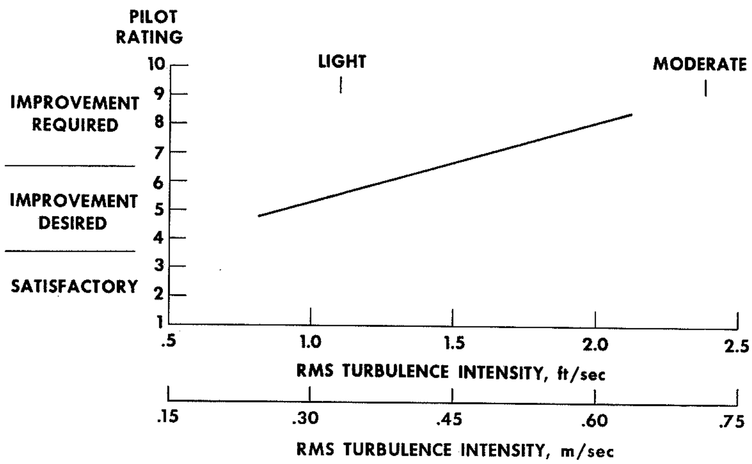

Figure 9

EFFECT OF ADVANCED AVIONICS ON PILOT RATINGS OF ILS APPROACHES

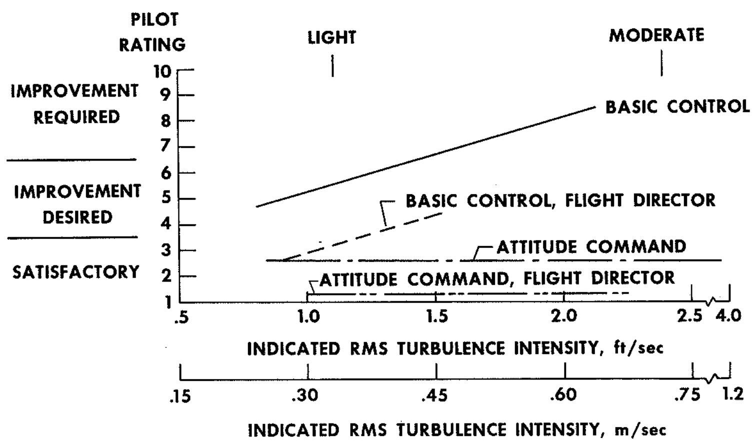

Figure 10

COMPARISON OF VORTAC AND VERY LOW FREQUENCY NAVIGATION SYSTEMS

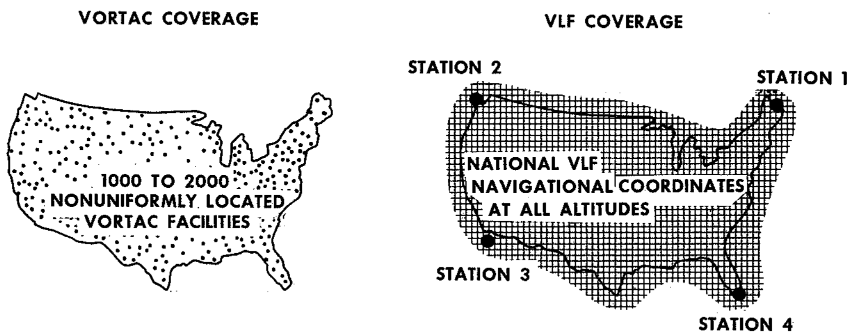

Figure 11

CURRENT UTILIZATION OF HIGH-LIFT DEVICES

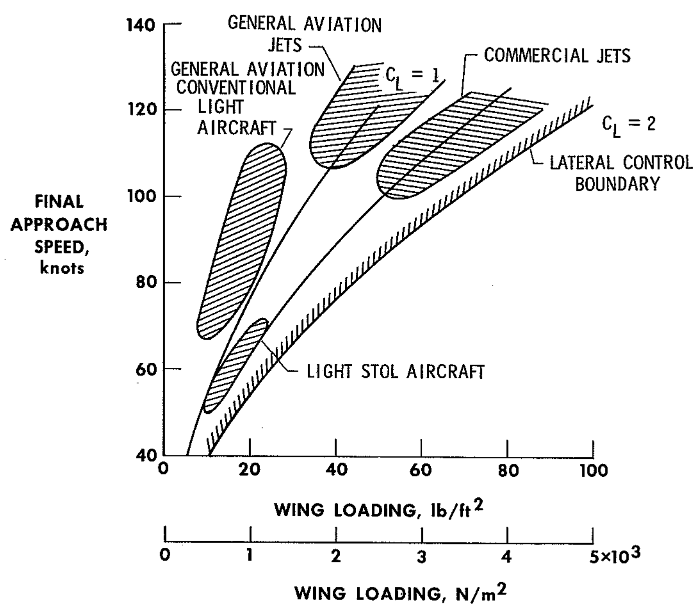

Figure 12

DIRECT-LIFT JET V/STOL CONCEPTS

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INTRODUCTION

For this paper, it has been assumed that a commercial V/STOL transportation system is desired and will not be debated; furthermore, analysis of costs and profits will not be made. NASA has concluded that direct-lift jet-powered V/STOL concepts have the greatest potential for future short-haul transportation. Such aircraft can be produced with current technology; however, they would be deficient in weight, noise, and operational characteristics. This paper will show how advanced technology can improve direct-lift jet V/STOL transports. Anticipated improvements in structures, propulsion, and aerodynamics were presented in previous papers. Their effects on advanced V/STOL transports designs will be noted. In addition, low-speed control and operation must also be considered because of the unique performance capability of V/STOL aircraft. Recent flight experience with direct-lift jet V/STOL aircraft will be used as a basis of the control and operational considerations.

REPRESENTATIVE DESIGNS

Several V/STOL transport studies that are most pertinent to this paper are outlined as follows:

Design Studies

1965-1966	Feasibility of different concepts
1967	Application of advanced technology
1970-1972	Direct-lift jet transports

Flight Test

1965-1971	Remotely driven lift fan (XV-5)
1969-1971	Integral jet (DO-31)

Preliminary design studies of various concepts were made in 1965-1966 (refs. 1, 2, and 3). Several concepts were then studied with advanced technology applied (ref. 4). These studies showed that many concepts were feasible, but jet-powered types were more attractive over 200- to 500-nautical-mile routes. Direct-lift concepts with V/STOL propulsion separate from cruise merited greater attention, and these are presently being studied for NASA by McDonnell Douglas Corp., The Boeing Co., and North American Rockwell. The studies include lift fans remotely driven

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and integrally connected to the gas generators. The designs are for 100-passenger V/STOL transports with a noise level goal of 95 PNdB at a 150-meter (500-ft) sideline distance.

Flight tests have been made by NASA with a number of V/STOL research aircraft to evaluate and define the low-speed performance, handling, and operating characteristics that will be needed for commercial operation. Two of these aircraft have characteristics most representative of the concepts discussed in this paper, the Ryan XV-5 with remotely driven lift fans (refs. 5 and 6) and the Dornier DO-31 with integral jet engines. Their planforms are compared with the advanced counterparts in figure 1. The upper aircraft have fans driven by gas generators remotely located. The fans and gas generators are interconnected by ducting to maintain symmetry with a gas generator failed and to utilize efficiently the available energy for V/STOL control. The advanced design would have higher pressure ratio fans than the XV-5 so that they can be better arranged to obtain a higher wing loading. For the XV-5 aircraft, the conversion from V/STOL to cruise directs all the gases to conventional jet tailpipes. One set of fans is positioned vertically for cruise in the advanced design to reduce fuel consumption and noise, and it has exhaust vectored for V/STOL. The other concept being studied is a jet concept where each lifting unit is integral with the gas generator. The DO-31, representative of this class, has wing pods each containing four small turbojet lift units that are modulated for V/STOL control, and they are shut off above 160 knots; under each wing panel is a turbofan unit with nozzles that deflect the thrust from a vertical force in hover to the cruise thrust. The advanced design would be similar, but would use lower pressure ratio fan engines for V/STOL to reduce the noise. It must be recognized that the advanced designs are only illustrative; there are many ways to arrange the propulsion systems, and the study has not progressed far enough to sort out the most desirable designs.

In the following discussion the concepts related to current state of the art are based on XV-5 and DO-31 aircraft types, assuming that the aircraft and propulsion system had entered prototype construction this year (which of course is not the case). The advanced technology version would represent advances expected in 10 to 15 years.

PROPULSION AND CONTROL

V/STOL aircraft are unique in requiring their propulsion system to supply the moments for aircraft attitude control. This means that the lift fan engines must be modulated more rapidly and precisely than conventional jet engines. Figure 2 presents the variation of time constant with fan pressure ratio. The time constant (time to 63 percent of the steady-state commanded thrust change) indicates the ability of the unit to be used for control; pressure ratio is related to fan diameter which affects the inertia and hence the time constant. V/STOL transport studies have shown that the thrust changes of about

±20 percent of rated thrust must be considered, and the time constant must be no greater than 0.2 second for satisfactory control of the aircraft attitude. The lift fans of current interest are in the pressure-ratio range of 1.25 to 1.35 so that low noise can be obtained with a reasonable size. Estimated time constants for the remote fans are shown by the shaded band; estimates are not available for the integral fans, but the values will probably not be smaller than for the remote-fan units. These lift-fan units have not been built, and there is a large range of uncertainty in the values of time constant. Consequently, some measurements from the XV-5 remote fan and DO-31 integral lift units are included for reference. The time constants of only the fan on the XV-5 was nearly 1/2 second, which would have been unacceptable for attitude control. Consequently, exit louvers were used to throttle the fan thrust. (It should be noted that the time constant of the XV-5 fan plus engine was over 1 second.) Using this technology and increasing pressure ratio in the range of 1.25 to 1.35 gives the larger time constants for remote fans. Improved fan design and lighter weights should reduce the fan time constant to less than 0.2 second. By transferring gases without changing gas generator speed, low time constants for the total system should be achieved. Information is not presently available on integral lift-fan engines in the pressure-ratio range of 1.25 to 1.35. The nearest comparable experience is with the DO-31 main engines, which are turbofan units with a pressure ratio of about 1.8. These VTOL engines are designed to be modulated for height control, and for this purpose the 1/2-second time constant was satisfactory. For the more critical control of attitude on the DO-31, lift-engine thrust was modulated. These small lightweight turbojet units with time constants of about 1/4 second were satisfactory for lateral control. These flight tests and other simulation studies have shown that time constants around 0.2 second are needed for satisfactory attitude control. Since the time constants for the future lift-fan units have not been well defined and may be unsatisfactory, more detailed analysis is needed to define time constant better, to study the engine in the whole control and stabilization loop, and to define methods to provide satisfactory response, if necessary.

The influence of the control system on the design of the aircraft is shown in figure 3 by the total available thrust divided by aircraft weight (T/W) for V/STOL aircraft with remote and integral fan engines. For commercial aircraft sufficient thrust-weight ratios must be provided for satisfactory control and also to compensate for a gas generator failure. For the integral lift-fan design, additional thrust must be installed for modulation in a positive and negative direction; many lift units are used to ease the problem of a gas generator failure. The major reduction in thrust-weight ratio from current to future designs results from lower control requirements (produced by advancements in control and stabilization systems) and by optimizing the lift-engine locations. With the advanced designs there is a large spread in thrust-weight ratio due to configuration differences, and the lower values correspond to the use of 12 or more lift units. Figure 3 shows that the remote fan designs generally have a lower installed thrust-weight ratio; however, there is considerable overlap with integral designs. Current remote fan designs use some thrust spoiling to achieve good response for attitude control, and this imposes a thrust penalty. With advanced fan and control designs, the thrust penalty is minimized by modulating fan speed with transfer of gases through the interconnecting ducts. The large range of thrust-weight ratios for advanced remote designs is due to different considerations for fan failure. (Gas generator failure will probably always have to be considered.) The larger thrust-weight ratios permit a fan failure at hover and throughout the flight envelope. The lower values do not permit hover with a fan failed. This can be taken in one of two ways: (1) The reliability of the simple fixed-pitch fan will be high and the exposure time to a catastrophic accident, very low; or (2) hovering is unnecessary and the aircraft will be operated at speeds and altitudes that permit a fan failure. Although lower thrust-to-weight values are shown for remote designs, the added weight and complexity of interconnecting ducting must be recognized.

Another design consideration is the effect of advanced aerodynamics on V/STOL designs. First, what about the use of a supercritical wing? For a short-haul mission, cruise speeds at a Mach number of about 0.8 have little advantage; however, the ability to use a thicker wing and retain the same Mach number offers significant structural advantages. Referring to the earlier aircraft sketches (fig. 1), it is seen that the design is dominated by the propulsion system, the wing is small, and there are a number of wing-body-propulsion junctures which will make it difficult to take advantage of supercritical technology to improve structural efficiency without seriously incurring propulsion system penalties. A typical V/STOL design should be tested to determine the possible improvements. Second, high-lift devices to delay leading-edge separation are needed to fly the aircraft in steep approaches with a nearly level fuselage. This is more an application of existing knowledge than advanced technology.

AIRCRAFT WEIGHT

At this point the effect of advanced technology in propulsion, controls, and structures on the gross weight of a 100-passenger V/STOL transport designed for a 400-nautical-mile range will be shown. Figure 4 is a bar chart where the component weights are indicated by the length of the bar, and the ratio of components to total weight $\left(\Delta W/W_{TO}\right)$ are given within the bars. (The absolute magnitude of the weights and weight fractions varies significantly for different V/STOL concepts and designs; however, the effect of advanced technology is fairly representative for all concepts.) For example, an aircraft based on current technology would weigh about 670 kN (150 000 lb); by taking advantage of improved technology the weight could be reduced to 400 kN (90 000 lb). Of particular interest is that the payload fraction has increased from 13 to 22 percent, which implies a considerable increase in profit potential. The large reduction in fuel is primarily due to the use of higher bypass-ratio fan engines in the cruise mode. The propulsion-system weight reduction is primarily due to the recycled design. The propulsion weight and thrust penalties for noise treatment offset improvements in aircraft and propulsion thrust-weight ratio. Since little change in fixed equipment weight is anticipated, the ratio is greater for the lighter vehicle. The reduction in structural weight fraction results from the use of composite structures. Thus, by combining the different items of advanced technology, the aircraft takeoff weight can be reduced by about 40 percent.

OPERATIONAL CONSIDERATIONS

The operational characteristics of V/STOL transports are important because their unique low-speed performance offers a large flexibility in flight paths that cannot be provided by CTOL aircraft. This flexibility can reduce the time at high thrust levels with beneficial effects on airspace, fuel expended, and noise pollution. Figure 5 compares the takeoff flight paths for V/STOL and CTOL aircraft. (The flight paths have been scaled up by a factor of 2 for clarity.) The V/STOL can be at an altitude of 300 meters (1000 ft) and conversion speed of 140 knots at the time when a CTOL aircraft is just lifting off the runway. The resulting noise footprints are compared in figure 6 for the same noise levels (95 PNdB). Even though the current V/STOL has a greater sideline noise level than the CTOL, the area exposed to the noise is the same because of the increased altitude attained by the V/STOL. Recognizing the need to make future V/STOL aircraft much quieter, research has been performed on remote lift fans to determine the noise sources and to reduce the noise levels by proper selection of fan pressure ratio, design, and noise treatment. With a design fan pressure ratio of about 1.3 advanced lift-fan transports should approach 95 PNdB at a 150-meter (500-ft) sideline distance. More important than the reduction in sideline noise is the large reduction in area exposed by properly using the low-speed performance. (For the example shown the footprint area of the advanced V/STOL is less than 2 percent of the CTOL.) Criteria for acceptable noise levels must be developed for V/STOL aircraft; like conventional aircraft, the exposure time must be included to evaluate the annoyance factor.

Similarly, there are large benefits to be gained in V/STOL approaches illustrated in figure 7. However, it has been difficult to fly the desired profiles on instruments in a routine fashion because of V/STOL handling and operating characteristics. Recent NASA flight tests with the DO-31 showed that simulated IFR approaches could be made with a path three times as steep as used by CTOL aircraft. While on this path the aircraft could be converted to the V/STOL mode and decelerated to 50 knots. The lack of suitable guidance and display information precluded an IFR hover and landing. The research pilots noted that the advanced stabilization system made it easier to fly along this profile;

however, too many propulsion system controls had to be modulated to manage the powered lift during the precision approach and landing (ref. 7). More simulation and flight work must be done to integrate these controls and provide some automation to reduce the pilot's workload, particularly for the more complex curved decelerating approaches shown for the advanced V/STOL. Advanced guidance and displays must also be provided so that precision approaches and landings can be routinely made under instrument conditions with the pilot in the control loop. Ultimately, it should be possible to fly curved, decelerating approaches. Our experience has shown that such approaches will require aircraft with extensive leading-edge devices to avoid stall when the fuselage is nearly level for passenger comfort. When this is done, the aircraft's performance can be fully utilized and the approach time can be cut in one-half and the noise pollution reduced.

CONCLUDING REMARKS

Our studies have shown that future V/STOL transports should be jet powered with cruise Mach numbers of at least 0.7, and with direct-lift V/STOL systems designed for a pressure ratio of about 1.3. The lift fans may be remotely driven or integral with the gas generators. Each type has advantages and disadvantages. The control system must be integrated with the propulsion system and aircraft design so that the V/STOL performance capability can be utilized by the pilot to increase the usable flight path to reduce the time and noise during the approach and landing. To be acceptable, commercial transports must rely on stabilization systems with some degree of automation and improved guidance in the terminal area.

Significant improvements are offered by advanced technology in reducing the aircraft weight and in improving the low-speed operation in the terminal area to reduce further the time at high thrust and related noise.

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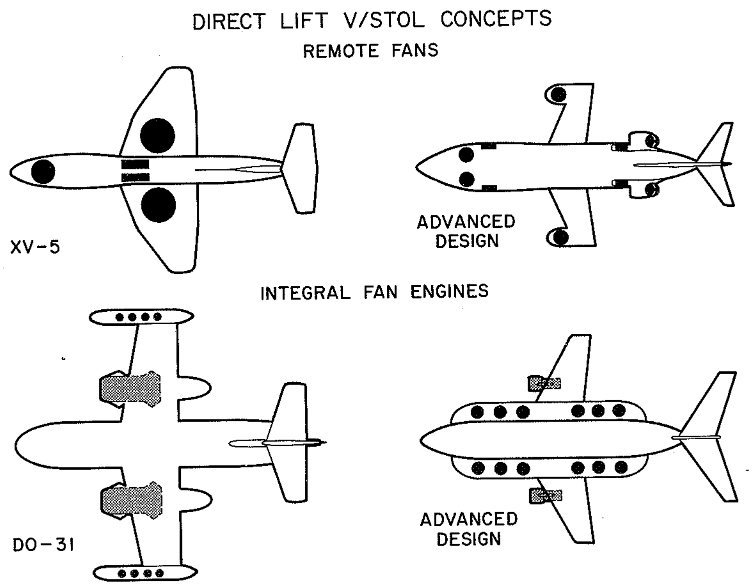
DIRECT LIFT V/STOL CONCEPTS
REMOTE FANS
XV-5
ADVANCED DESIGN
INTEGRAL FAN ENGINES
DO-31
ADVANCED DESIGN

Figure 1

FAN THRUST MODULATION FOR V/STOL CONTROL

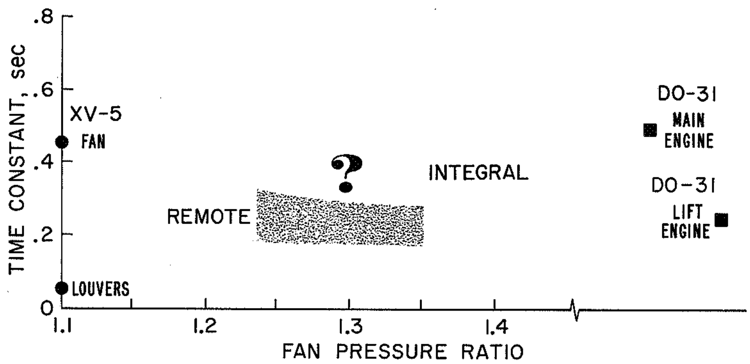

Figure 2

THRUST - WEIGHT RATIO FOR V/STOL

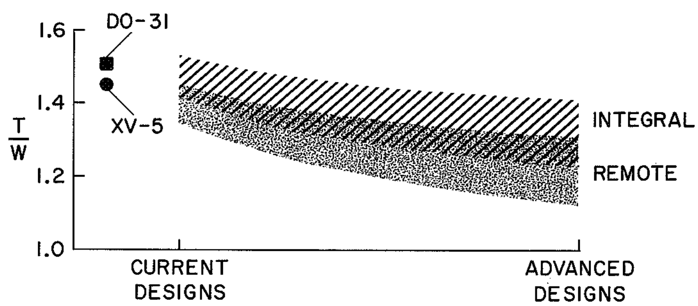

Figure 3

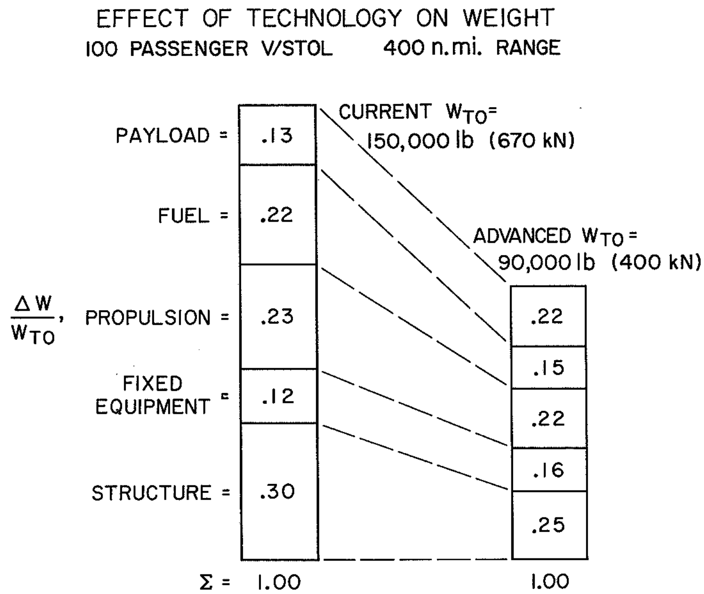

Figure 4

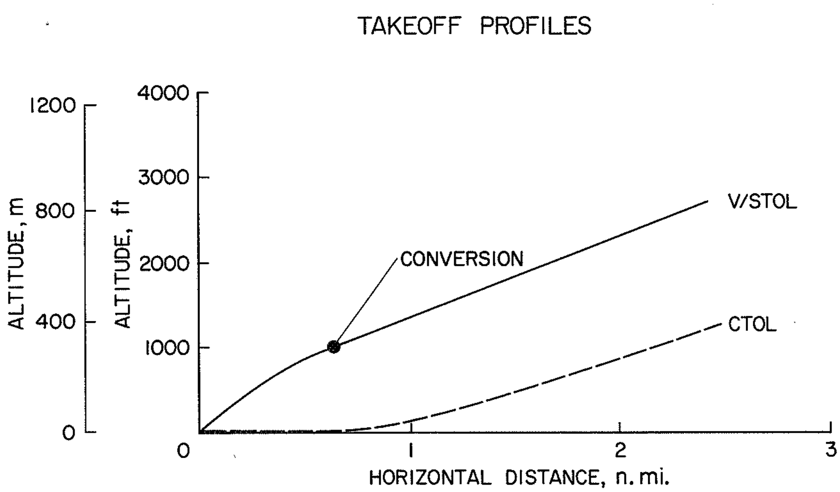

Figure 5

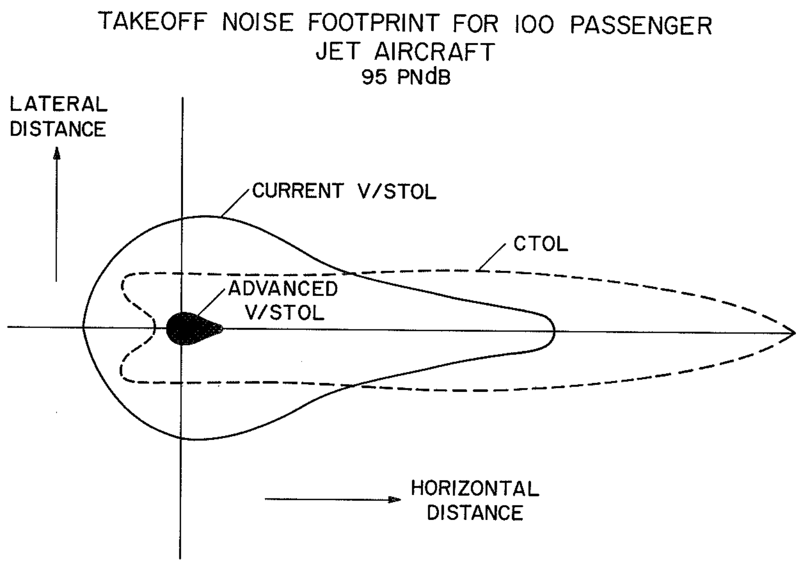

Figure 6

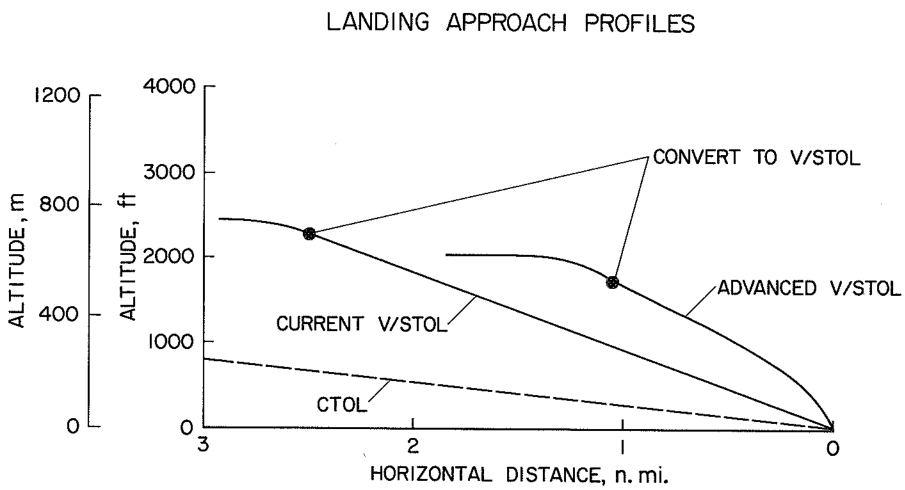

Figure 7

ROTORCRAFT APPLICATIONS AND TECHNOLOGY

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INTRODUCTION

Commercial use of the helicopter has been slow in expanding to provide the services of which it is potentially capable. Military uses have expanded significantly over the past two decades and have resulted in technology that can be used to improve the vehicles. There have been technology advancements in the past which, had they been carried to operational status, might well have reduced the costs, vibration, noise levels, and maintenance and improved the flying qualities and all-weather capability to the point where expansion of civil applications would have been self-generating.

Recently there has been an increase in the application of helicopters to civil missions. This is probably attributable more to the large numbers of trained military helicopter users returning to civilian jobs and bringing with them experience and knowledge of the helicopter's capability than to technological factors relating to the vehicle.

The series of photographs in figure 1 shows a spectrum of the helicopters in civil activities today; the uses include ambulance service, police work, mapping and geologic explorations, crane operations, work-personnel transportation, scheduled passenger service, crop dusting, and executive transportation. The helicopter, because of its inherent advantage of lifting capability at hover or low speeds for long periods of time, is finding increased usage even without special attention to the development of economical and operationally suitable vehicles for these kinds of applications. With some directed application of technology, however, to produce vehicles specifically for the civil requirements, the world could be on the threshold of a possible revolutionary expansion of the use of the helicopter.

TARGETS FOR TECHNOLOGY APPLICATIONS

The principal targets for the application of technology to improve the helicopter are propulsion systems, noise, vibration, structural integrity, and instrument flight capability. Propulsion systems are covered in some depth in other papers in this compilation; there are several items, however, which merit specific consideration for helicopter engines. Because of the short flight leg of most helicopter operations, the number of engine on-off cycles is greater and, consequently, thermal fatigue is of much more importance. Helicopter operations are more frequently conducted at partial power, which makes fuel consumption at off-design power levels of considerably more interest. Also, one of the early promises of the turbine engine in helicopter use was that the rotor would no longer have to be a constant-speed system; that is, the turbine engine could deliver essentially constant power over a relatively wide range of shaft output speeds. This feature, if it can be used, would eliminate the need for major compromises now necessary in rotor design for low-speed performance and high-speed limitations.

The discussion to follow covers the remaining principal targets. The noise problem is discussed; however, rather than discuss vibration and structural integrity, per se, the technology which bears most directly on these problems – aeroelastic analysis capability, structural concepts, and rotor geometry – is discussed. Also, some of the technology relating to instrument flight for helicopters is indicated.

NOISE

The trends of noise with weight for turbine-powered helicopters are shown in figure 2. At gross weights of about 156 kN (35 000 lb) – the largest helicopters for which there are much data – the noise levels are well above currently acceptable standards. The crosshatched area shows what it is believed the noise level for the next generation of helicopters of these sizes could be, if current technology is brought to bear on the problem. For helicopters larger than the 156- to 178-kN (35 000- to 40 000-lb) vehicle, it appears that new technology will be required to meet the projected noise criteria.

Figure 3 shows a number of significant noise reduction factors and their cumulative effect in reducing noise. A primary parameter to which much of the noise can be related is the rotor-blade tip speed. The amplitude of both broadband and discrete-frequency noise is reduced as the tip speed is reduced. The design tip speed, then, establishes peak levels of noise for a given size of helicopter and hence becomes a primary design parameter for noise control. If the designer has flexibility enough to make the necessary cumulative design changes for minimum noise, then the figure shows the approximate order of importance and how much noise reduction might be expected for a given change before additional effects must also be considered to make further reductions. For example, the figure shows that, generally, tip-speed noise is about 12 dB higher than the next most important source. Hence, in the typical case, after tip speed has been lowered to decrease noise by that amount, blade loading has to be reduced to achieve additional noise reduction; similarly, for successive noise reductions, the additional factors in turn must be considered in combination to achieve further results. Thus, a noisy helicopter can be as much as 30 dB quieter if emphasis is placed on the cumulative effect of these design features.

Although the penalties associated with noise control are not discussed in this paper, it can be said that measuring the penalties in terms of performance loss, or even cost, may not be valid if quiet helicopters are acceptable and noisy ones are not.

AEROELASTIC ANALYSIS CAPABILITY

One of the major tools for attacking the problems of helicopters is a design analysis capability, which includes coupled aerodynamic and elastic effects. The rotor has not yet been explored sufficiently experimentally to provide a reservoir of aerodynamic and dynamic design data as have been airfoils, wing-bodies, or turbine cascades, for example. Primary reasons are that such experimentation has been too costly and there have been no strong incentives. An adequate design capability for rotor vehicles, therefore, must depend on the development and use of a sophisticated and complex analytical capability. The status of the two disciplinary factors – aerodynamics and dynamics – provides some measure of the capability to make the necessary analyses.

Figures 4 and 5 provide some definition of the dynamics and aerodynamics disciplines as they now relate to the rotor requirements. Figure 4 illustrates the dynamics design problem which includes all the elements of the vehicle, with the pilot and the controls as a coupled system. The rotor is an elastic device, with many modes within itself, flexibly attached to the rest of the vehicle with all its masses in a springy relationship to one another. The control system represents yet another elastic path through which many of the dynamic degrees of freedom can couple and interact. The electronic, central-computer, active-control concepts could find particularly fruitful application in decoupling much of this activity and for permitting the construction of desirable or vibration-alleviating feedback paths. All the degrees of freedom depicted are being excited continually by periodic inertial forces and by complex aerodynamic loadings represented by the hammer.

Figure 5 indicates the nature of the aerodynamic activity in each revolution of the rotor. On the advancing side, where the vehicle speed and the blade rotational speed add, there are shock effects; flow characteristics – Mach number, angle of attack, skew angle, Reynolds number, and dynamic pressure – change continually as the blade moves to the retreating position, where high angle of attack and stall occur. Continuing in its revolution, the blade encounters vortices shed by its predecessor and this induces stall and impact loadings. In essence, the rotor blade encounters with each revolution just about all the aerodynamic variations and problems that a fixed wing might experience only a few times in its entire life.

Figure 6 summarizes the present capability and what the current efforts are expected to accomplish with respect to the capability to make combined aerodynamic-elastic analyses. The first boundary in the figure represents a judgment of where current capabilities lie in terms of handling the combined dynamic and aerodynamic effects.

The second boundary shown in the figure indicates the analytical and experimental studies underway which will improve these capabilities; these include wake effects in the aerodynamics and vehicle and control-system degrees of freedom in the dynamics. Future developments within the next decade will provide necessary understanding and computation capability to permit handling all the important dynamic and aerodynamic phenomena in an integrated manner for design purposes.

ADVANCED STRUCTURAL CONCEPTS

Three structural concepts with potentially significant impact on the helicopter design are prestressed structural components, elastic rotors, and composite structures. The first, the use of prestressed structural components, is presently being explored for blade construction in the form of a cryogenically stretched spar. The second and third, elastic rotor systems and use of composite structures, are being explored in the construction of an elastic rotor with carbon composite materials.

The prestressed rotor blade spar, shown in figure 7, consists of a core of stainless-steel sheet about which a fiber-glass wrap is placed. The spar is fabricated by internally pressurizing and cryogenically stretch-forming the core against the fiber-glass overwrap, inside a die. The result is a permanent compressive prestress in the metal, with the promise of lighter weight, improved fatigue life, a fail-safe system if a crack should develop, and blade design characteristics controlled by variations in the wrap pattern.

The elastic or bearingless rotor concept is shown in figure 8. The torsionally flexible spar is constructed of uniaxial carbon epoxy and enclosed by a torsionally stiff aerodynamic shell, also of carbon-epoxy material. The construction can be very simple, as indicated by the figure. Advantages are a more efficient structure with reduced complexity that is easily designed to obtain the desired frequency characteristics.

Significant reductions should result in maintenance requirements for a helicopter with these advanced structural concepts in the rotor and, eventually, as material costs come down, significant cost reduction for the rotor components should be possible.

ADVANCED ROTOR CONCEPTS

The advanced rotor concepts indicated in figure 9, all of which have considerable promise of improving rotorcraft versatility and operational suitability, are currently being explored. The rigid coaxial rotors illustrated on the upper left in the figure provide symmetry of aerodynamic forces, that is, the advancing- and retreating-blade problems are averaged; the result is a smoother lifting system providing the potential for high-speed vehicles based on rotor lift alone.

The variable-geometry concept shown on the lower left in figure 9 is aimed at blade geometry and arrangements which are designed to control blade-wake interactions. These designs include unequal blade lengths, vertical spacing between adjacent blades, and uneven azimuth spacing. This concept is being considered, not only to minimize unfavorable effects but, possibly, to generate favorable interactions and thus to improve performance and reduce vibration and noise.

The variable-diameter rotor permits use of low disk loading for compound helicopters in hover and at low speeds with advantages of reduced noise and downwash effects and improved performance. At the higher speeds, where the rotor is being carried along with a penalty, the blades can be retracted.

Similarly the jet-flap rotor concept can provide the compound helicopter with potential benefits by reducing the size of the rotor required. The jet-flap rotor can be used to obtain increased lift in hover and at low speeds. This results in a smaller rotor size for a given vehicle with attendant improved cruise performance.

Each of these concepts has yet to be incorporated into a flight vehicle for exploration of its total potential. Plans are being made for the joint NASA/Army development of a rotor test vehicle capable of flight-test operations to explore these concepts.

Another rotorcraft concept which continues to show promise is the tilt rotor, where the rotors are in a vertical position for landing and take-off and are tilted through 90° to become propellers for cruise. Work is currently underway by NASA and the Army jointly to bring the technology for this concept up to the level required for vehicle development.

INSTRUMENT FLIGHT CAPABILITY

The helicopter operational environment, consisting of take-offs and landings in confined areas without runways, points to the need for a steep, circuitous, instrument approach path with the aircraft decelerating to zero speed as it descends to the landing spot. Figure 10 depicts the important factors in helicopter instrument flight systems: the aircraft and avionics systems, the control system, the displays, and the pilot. Because of the number of variables in the steep, decelerating approach, the level of pilot workload is sufficiently high currently to directly limit the achievable instrument performance capability. Helicopters designed for commercial operations, then, will require that these systems alleviate many of the control tasks for the pilot. The most efficient solution clearly relies on advancements in the control and display systems. As shown by figure 11, relationships exist between the degrees of sophistication required in the two systems. The problem is to examine these systems and define the capability provided by the various possible combinations of these concepts. Flight experience has shown, for example, that the handling qualities and guidance provided by improved control systems and displays (such as attitude command, instead of simple attitude stabilization, and moving-map pictorial displays with steep-path flight-director guidance) provided capability to perform smoother, more accurate approaches with significantly reduced pilot workload – an important factor for anticipated short stage-length operations. This combination of systems permits the use of a wide range of instrument flight paths.

It has been found that instrument operational capability improves consistently as improvements in each of these systems are made. The requirement for flight-director information has been firmly established for the approach tasks, but displaying the information to the pilot in a form that is adequately integrated with the situation information is a major problem yet to be solved. A cathode-ray-tube display will provide flexibility for command and situation information to be integrated in more suitable display formats. The command control concept benefits handling qualities by suppressing basic instabilities and providing a nonvariant response to controls or disturbances from hover to high speed. The application of fly-by-wire technology should be explored further for control of the vehicle and for possible automation of many piloting tasks, as well as for improving dynamic characteristics of the rotors and for alleviating problems arising from interaction and coupling of the various components of the vehicle. For the most operationally suitable helicopter it appears both feasible and desirable, at the present time, to consider stability and control augmentation up to the level required for automation, with pilot backup provided through much improved integration of displays to achieve essentially a display simulation of the real world.

To date, under test conditions, instrument approaches have been flown to a hover at the landing pad for glide-path angles up to 15°. In addition, constant-speed approaches have been successfully completed at angles up to 25°, in order to build up to the near vertical, as required in the final stages of the decelerating approach. Figure 12 is a summary of the steeper, constant-speed approaches to date in terms of the glide-slope envelopes obtained for angles up to 25°. The conventional 3° glide-path angle which is current practice is shown for reference. Although there is some deterioration in glide-slope performance at the steeper angles, as indicated by the increased scatter band, the pilots indicate that the steeper approach angles are not any more difficult to fly than the shallower angles. The deterioration in performance is believed to be due to the inadequacy, for the higher angles, of the flight-director logic which in this case was the same as that used for the lower angles.

ROTORCRAFT OPERATIONS IN THE 1980's

The ultimate goal for helicopter operations is illustrated in figure 13. These types of operation should continue routinely under foul-weather conditions; the illustration also provides an appropriate indication of the results of applying the technology discussed herein. Rotorcraft such as that indicated in the figure will have had their more adverse characteristics corrected and with the application of new control and avionics technology will be capable of providing unique, safe, comfortable, and reliable service for a very great number of transportation needs. Table I summarizes some of the projections of the technology for this helicopter and indicates what may be expected of two other promising rotorcraft, the compound helicopter and the tilt-rotor configuration. Improved materials and structural designs will reduce empty-weight fractions. Improved propulsion systems and vehicle design will decrease fuel requirements. Both of these advancements bear directly on the payload fractions. Comfortable cruise speeds will be up as indicated in the table. The net effect of all these factors will result in very significant improvements in productivity.

TABLE I

POTENTIAL IMPROVEMENTS IN ROTORCRAFT DURING THE 1970's AND BEYOND

	1970	1980		1985
	HEL.	HEL.	COMP.	TILT ROTOR
$\frac{\text{EMPTY WEIGHT}}{\text{GROSS WEIGHT}}$, percent	65	55	60	55
$\frac{\text{FUEL WEIGHT}}{\text{GROSS WEIGHT}}$, percent	11	9	8	7
$\frac{\text{PAYLOAD WEIGHT}}{\text{GROSS WEIGHT}}$, percent	24	36	32	38
CRUISE SPEED, knots	130	190	250	350
$\frac{\text{PAYLOAD X VELOCITY}}{\text{EMPTY WEIGHT}}$	55	124	133	230
$\frac{\text{TON MILES}}{\text{DOLLAR}}$	0.9	1.6	1.5	1.8

HELICOPTERS IN CIVIL APPLICATIONS

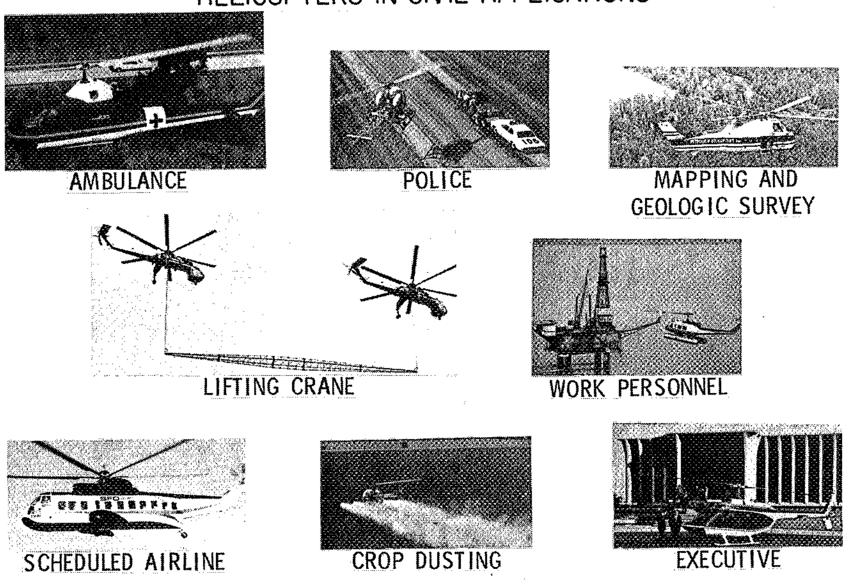

Figure 1

TREND OF PERCEIVED NOISE LEVELS WITH HELICOPTER SIZE

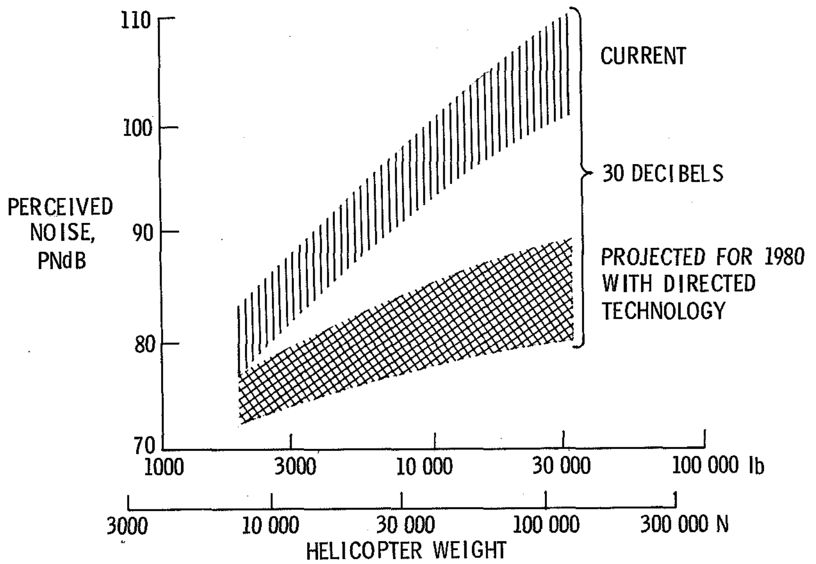

Figure 2

SIGNIFICANCE OF VARIOUS FACTORS IN NOISE REDUCTION

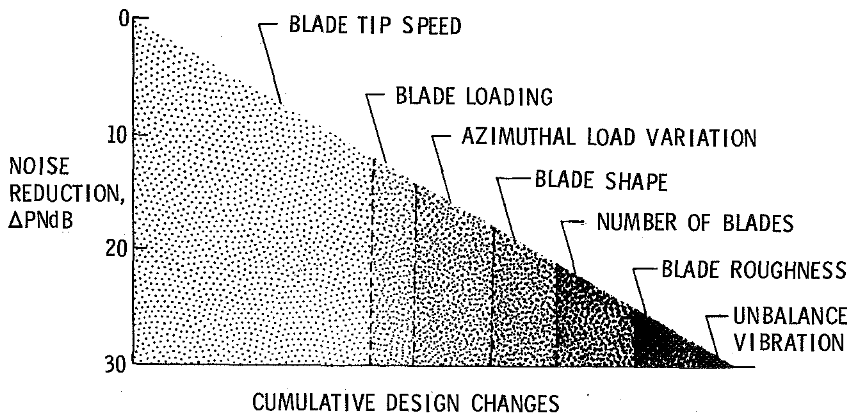

Figure 3

DYNAMICS COMPLEXITY

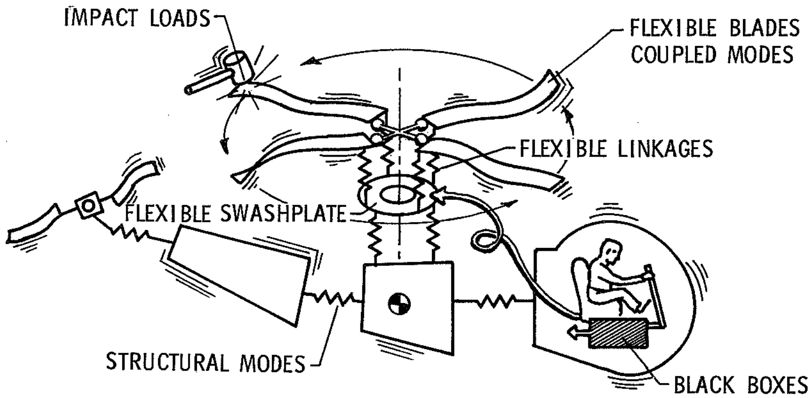

Figure 4

ROTOR TRANSIENT AERODYNAMICS
FULL CYCLE IN 1/5 second

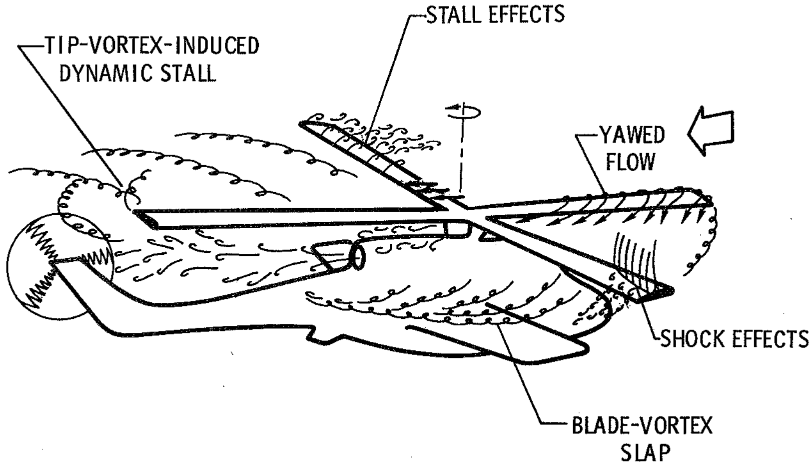

Figure 5

AERODYNAMIC AND DYNAMIC CONSIDERATIONS REQUIRED IN DESIGN ANALYSES

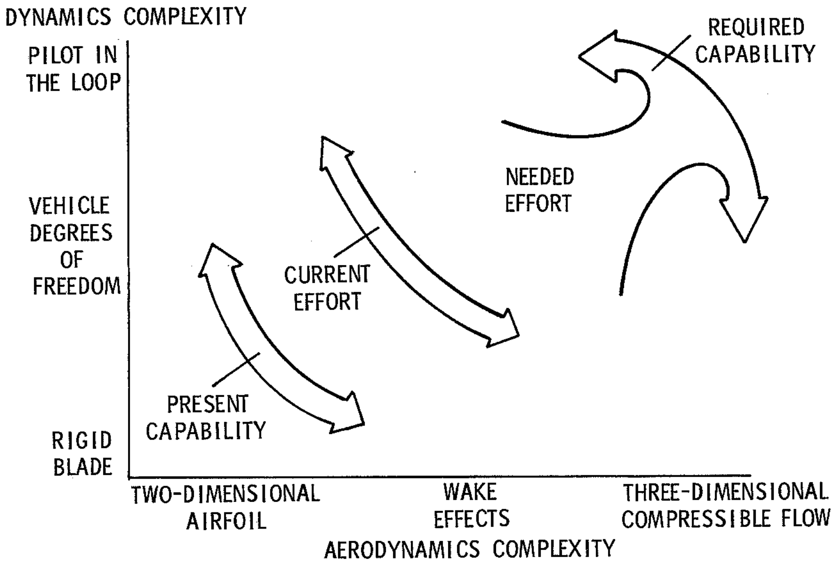

Figure 6

PRESTRESSED ROTOR BLADE SPAR CONCEPT

SPAR IS FABRICATED BY INTERNALLY PRESSURIZING AND CRYOGENICALLY STRETCH-FORMING AGAINST FIBER-GLASS OVERWRAP INSIDE A DIE.

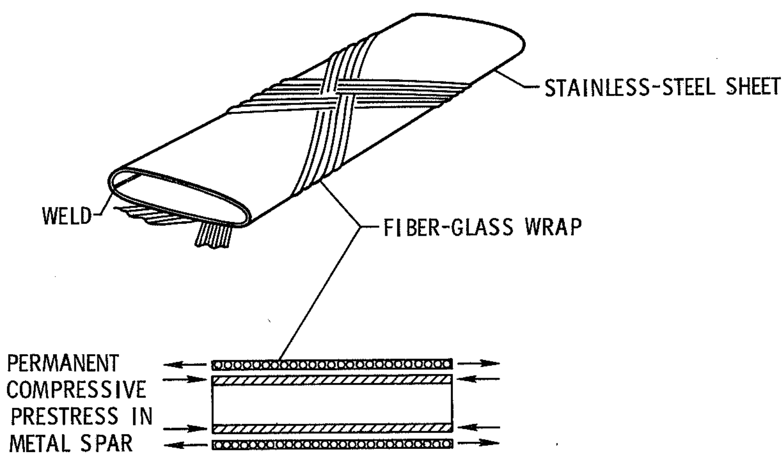

Figure 7

CARBON COMPOSITE, ELASTIC ROTOR CONCEPT

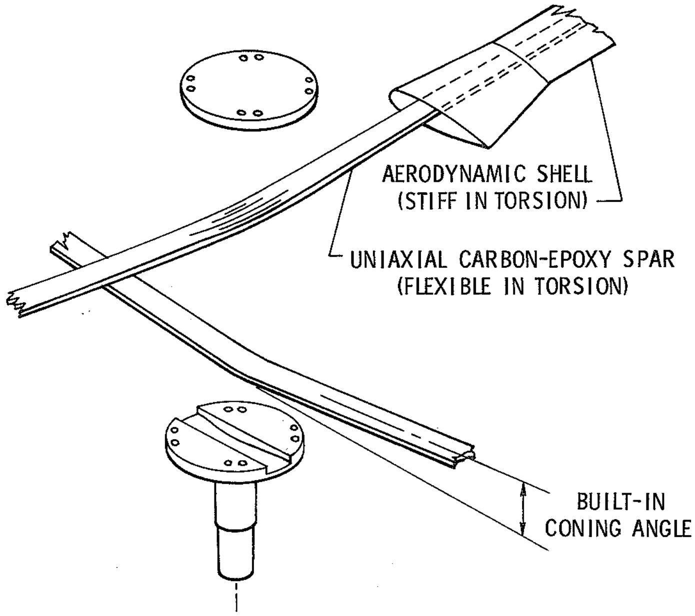

Figure 8

ADVANCED ROTOR CONCEPTS

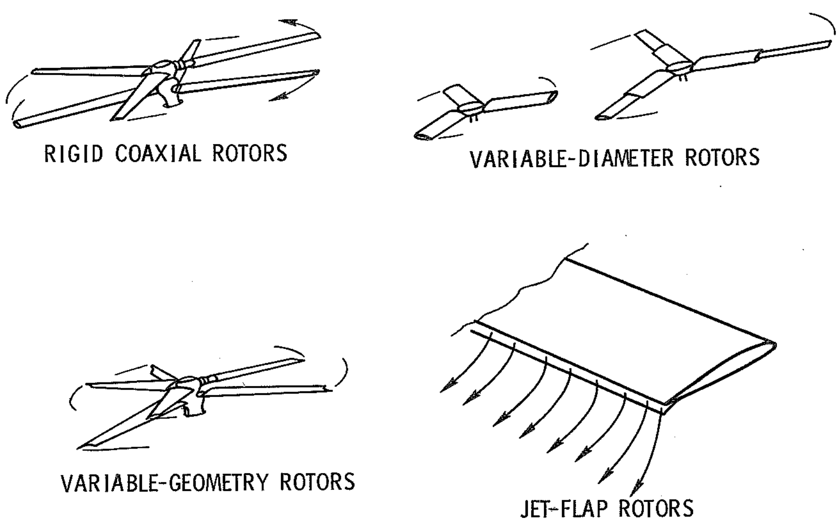

Figure 9

HELICOPTER INSTRUMENT FLIGHT SYSTEMS

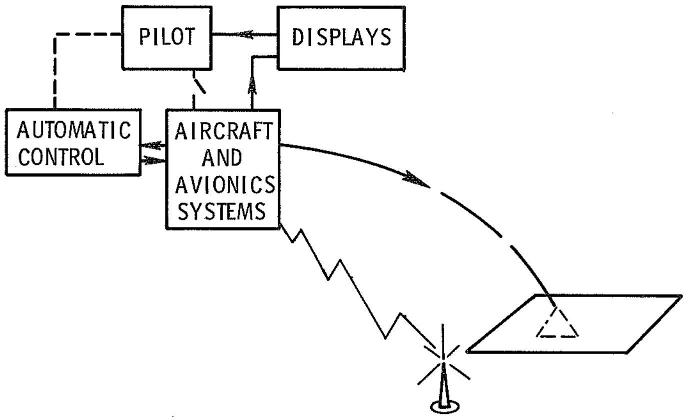

Figure 10

ADVANCED CONTROL AND DISPLAY CONCEPTS

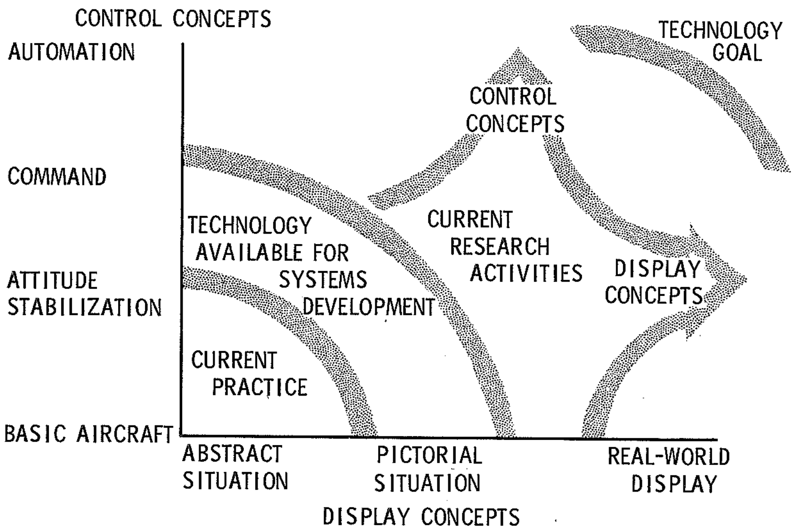

Figure 11

SUMMARY OF CONSTANT SPEED APPROACHES

Glide-path angle, γ, deg	Ground speed, knots, for rate of descent of –		
	152 m/min (500 ft/min)	228 m/min (750 ft/min)	305 m/min (1000 ft/min)
9	31	47	62
12	23	35	46
15	18	27	37
20	14	21	28
25	11	17	22

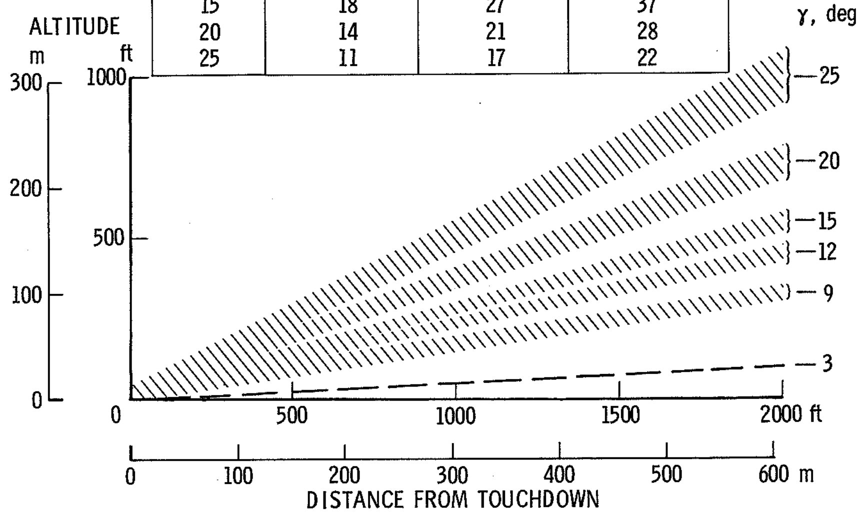

Figure 12

TYPICAL ALL-WEATHER OPERATION OF FUTURE ROTORCRAFT

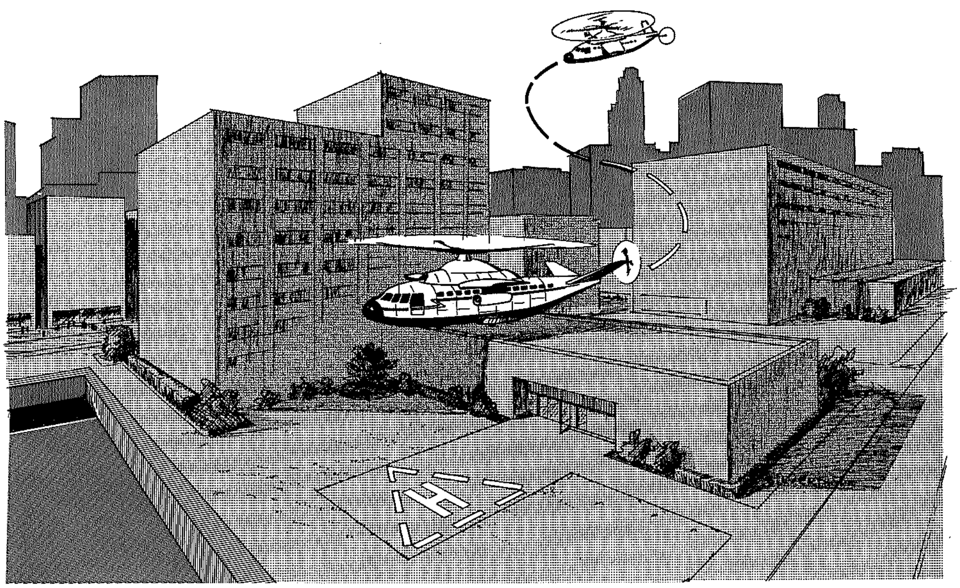

Figure 13

EFFECTS OF ADVANCED TECHNOLOGY ON STOL TRANSPORT AIRCRAFT

By Woodrow L. Cook
Ames Research Center

INTRODUCTION

Commercial transports have realized great success in recent years by exploiting the economic benefits of increased range and payload. These have been achieved most directly through steadily increasing size and only secondarily through exploitation of very advanced technology. Improvements in aerodynamics, propulsion, structures, and flight dynamics have been cautious and evolutionary. The one area where very advanced technology may be required is that in improving STOL transport aircraft to regain public acceptance through noise reduction.

The objectives of this paper are as follows:

(1) Application of specific technology advances to commercial STOL transportation

(2) Total effect of technology advances on STOL transport aircraft

(a) Gross weight
(b) Direct operating cost
(c) Acceptance

(3) Assessment of advanced technology progress for STOL transportation in the 1980's

It can be anticipated that the success of the STOL transport will depend very heavily on the incorporation of advanced technology. The reason for pursuing STOL capability is to meet a short-haul market demand where short-segment high-frequency service is the objective. Thus, the beneficial effects of long range and high payload (very large aircraft) must be offset by the economic benefits realizable from our technology which improves basic system efficiency. Beyond this, the public acceptance factor becomes exceedingly important because of frequent exposures. Obviously the aircraft must be quieter than any yet developed. Probably some upper size limit will appear which the public will accept close overhead. The purpose of this discussion is to highlight some of those principal areas of technology which will require maximum exploitation if STOL aircraft are to become sound economic ventures.

Many STOL system analyses have served to identify the characteristics which a transport aircraft must have to allow successful development of a STOL short-haul system. These are as follows:

(1) 50 to 150 passengers – a smaller transport size to supply high service frequency

(2) 750 to 1100 km (400 to 600 n. mi.) range – it seems clear that if a STOL system is to be economic, it must capture at least the shorter segments of DC-9, 727, and 737 service, particularly to allow utilization at off-peak travel hours

(3) 0.7 and 0.8 Mach number cruise speed – a requirement to attract passengers and operate efficiently for the longer segments; this requirement also almost dictates the jet-powered concept

(4) 4 to $5\frac{1}{2}$ kN/m^2 (80 to 110 psf) wing loading – both for efficient cruise and for passenger comfort

(5) 610 m (2000 ft) or less balanced field length – a balance between aircraft sophistication to achieve low flight speeds and real-estate costs for runways or STOL runway availability

(6) Low noise – no more than 95 EPNdB initially with the potential of significant further reductions

Design studies of aircraft to meet the general specifications for fan-jet STOL transport have been undertaken by many groups. NASA, with industry, has and is supporting aircraft design and system studies to identify critical technology areas as a guide to its research program. Major study efforts from which NASA's present program has evolved are outlined as follows:

Past Studies

1965-1966	Transport aircraft feasibility
1967	Advanced technology short-haul transport systems

Ongoing and Planned

1971-1972	Augmentor wing transport
1971-1972	Direct-lift V/STOL transport
1972	STOL transport system studies

Past studies are presented in references 1 to 4, and several studies are being conducted for NASA under contract with The Boeing Co., McDonnell Douglas Corp., and North American Rockwell.

The principal concepts emerging from these studies are shown in figure 1. In many ways the turboprop STOL aircraft is very attractive but as the economic importance to the system of the longer high-speed stages is emphasized, the economic potential weakens. Therefore, attention is directed to the turbofan concepts. On the left is shown one version of the internally blown wing where engine air is ducted through the wing and blown over the flaps in a distribution pattern designed to give maximum usable lift augmentation. At the center is shown the simpler externally blown flap concept where engine air is directed externally over the wing and flap but less control is available to optimize flow distribution. On the right is sketched the direct-lift engine principle which has the unique potential of development into a true V/STOL machine when such systems are required.

From detailed design studies of the turbofan concepts, it has been possible not only to identify technology advances critical to success of STOL aircraft but, most importantly, to make quantitative assessments of the sensitivity. Some of the most important results obtained from the studies are discussed in the remainder of this paper.

SYMBOLS AND ABBREVIATIONS

Æ	aspect ratio
M	Mach number
M_{CR}	cruise Mach number
T/W	thrust-weight ratio
t/c	thickness-chord ratio
Λ	angle of sweep, deg
PNL	perceived noise level, PNdB
EPNdB	unit of effective perceived noise level
DOC	direct operating costs
S.L.	sideline

DISCUSSION

Aerodynamics

A technical aerodynamic advance of particular importance to STOL aircraft is the supercritical wing principle developed by Richard T. Whitcomb of the Langley Research Center. The economics of all STOL or V/STOL aircraft are particularly sensitive to structural weight fractions. For high-speed aircraft the supercritical wing principle offers the potential of reduced sweep, increased thickness, or increased aspect ratio or combinations of these to achieve lower-than-conventional wing structural weights. The internally blown flap concepts benefit in another way since supercritical airfoils have greater thickness far aft, thus easing the internal ducting problems by providing more internal volume and, in turn, easing the engine fan design and noise problems by enabling lower pressure ratio and, as a result, lower velocity gas flow.

Figure 2 shown previously in paper no. 3 by Edward C. Polhamus illustrates quantitatively some of these effects. Increasing cruise Mach number is on the vertical scale and wing thickness ratio, on the horizontal scale. For example, a variation of allowable Mach number, from drag-divergence standpoint, is shown for a conventional straight wing as the lower curve. The effect of employing a supercritical wing is shown by the upper curve. Clearly, speed can be increased 15 percent for the same wing thickness, or the wing thickness can be increased 42 percent for the same cruise Mach number. The latter case reduces the structural weight and provides volume for fuel or internal ducting.

In figure 3 the unit wing weight as affected by wing sweep, wing thickness, and the supercritical concept for a fixed speed is examined. As the example shows, a supercritical wing could be 6 percent thicker, with a 12-percent reduction in unit weight, than a conventional wing for the same design cruise Mach number of 0.82. It is evident that the supercritical concept enables many trade-offs, which need study in detail to find the optimum confirmation, and that the gains can be large. Increased understanding of and confidence in the supercritical wing concept will be of major importance to STOL aircraft and should be established as quickly as possible.

Structures and Materials

STOL aircraft design studies have also provided some quantitative assessment of the increase in structural weight for a given payload resulting from incorporation of higher power, sophisticated powered-lift systems, noise-reduction requirements, and special STOL operating equipment. For example, the empty weight of a 150-passenger STOL built along conventional lines would be about 35 percent greater than current aircraft. This change certainly justifies a close examination of new structural materials and techniques since smaller structural savings have a bigger impact on payload and economics than heretofore.

Figure 4 shown previously in paper no. 10 by Richard A. Pride indicates structural weight trends as composite material utilization is increased. The specific illustrations arise from detailed design studies of several airframe elements. The band encompasses the range obtained from less refined studies. The point marked "Target for 1981" was chosen as a reasonable base from which to estimate airframe structural weights and, in turn, assess the importance of pursuing the technology relating to composite material utilization. Table I gives the important characteristics of two aircraft configurations designed for the same payload and mission, one using current materials and structural techniques and the other using advanced composite materials with efficient fabrication techniques. The structurally advanced machine is about 120 kN (27 000 lb) lighter, which could result in an 11-percent reduction in direct operating costs (DOC), a gain very hard to achieve in any other way. As noted in the table, this reduction does not take into account the costs associated with developing mass production of these materials nor the costs for proper manufacturing techniques. Until these costs are better known, the practicality of these technology advances remains obscure and no, or small, reduction in DOC may result. The potential benefits, however, do indicate that this technology should be pursued actively, at least to the point where gains due to cost of fabrication can be assessed more clearly.

Propulsion and Noise

Up to this point the impact of advanced technology on STOL development has been reviewed in terms of its economic impact. There is no doubt, however, that obtaining community acceptance, from users and nonusers, will be an absolute necessity if any STOL system is to succeed. Foremost among the community acceptance problems is noise.

Obviously the STOL propulsion system development is the key to the noise problem since full power is used for steep climbout and steep descent and reduced-power operations are incompatible. Also, because power is used to produce lift, the principle used affects the noise exposure. Figure 5 shows the 90-PNdB contours for a 150-passenger STOL transport for the three leading STOL concepts using current technology. If the values of acceptable PNdB previously shown in paper no. 7 are meaningful, it is clear that STOL operations would be excluded from close-in community operation, a potentially important market, which would greatly reduce the utility and economic return of the STOL transport. Because of this fact, examination has been made of possible noise reduction resulting from properly matching the engine to the powered-lift concept, that is, new cycle designs, from use of advanced structures and advanced concepts for reducing generated noise.

Advanced propulsion techniques affect noise in two primary ways: (1) by reducing the noise generated and (2) by reducing aircraft size for a given payload through increased powered-lift efficiency. Figure 6 shows the variation of relative gross weight with noise levels for current technology and advanced technology propulsion systems. For this illustration the internally blown flap was used, but the results are typical of the other concepts. It is clear that advanced propulsion systems can provide either substantial noise reduction at a given weight (as much as 10 to 12 PNdB if small weight advantage is taken) or substantial weight reductions (as much as 15 percent if no noise reduction is required). Again, more detailed designs are required to define the optimum trade-off between noise reduction (operational flexibility) and weight reduction (initial cost). Nevertheless, the value of pursuing advanced propulsion technology is apparent.

Operational Aspects

Some individual impacts of advanced technology in STOL development have been discussed. It is useful to examine at least some of the combined impacts. Figure 7 shows how the several technology areas discussed affect the variation of relative gross weight (initial costs) with approach speed (field length). At speeds of 80 knots, advanced technology can gain about a 25-percent reduction in gross weight for the same payload. Obviously this would have a significant impact on system costs, not considering the research and development involved. A gain of this magnitude is certainly worth a major effort to achieve.

The effects of these technology advances are shown in figure 8 in terms of projected relative DOC for current and advanced aircraft. The cumulative effects have the potential of reducing the DOC approximately 15 to 20 percent. If this reduction can be achieved, the economic viability of the potentially important short-haul STOL system becomes much closer to reality. If the cost of fabrication and manufacturing of composite structures can be reduced, then the DOC can be improved to greater than the maximum value shown by the range of values in figure 8.

Up to this point, the effect of advanced technology has been considered in the light of its impact on aircraft STOL performance capabilities. Operational studies have highlighted the increased difficulties to be overcome in achieving full utilization of the performance. For example, the Federal Aviation Administration (FAA) is devoting increasing attention to the modifications necessary to the air traffic control (ATC) system in order that STOL capabilities can be used effectively. Complementary studies are being made of STOL aircraft guidance and control capabilities needed to take advantage of these advanced ATC developments.

The nature of the problem can be illustrated by examining allowable touchdown dispersions as a function of approach speed. See table II. If a 610-m (2000-ft) runway and reasonable decelerations are to be used, an 80-knot approach speed requires touchdown dispersions of less than ±120 m (±400 ft) from the chosen touchdown point; at a 65-knot approach speed the dispersions can increase to ±180 m (±600 ft). Even in the 65-knot case, the allowable dispersions are much less than found in today's jet transport operation and much upgraded guidance and control techniques are required. The one-third reduction in allowable dispersions when increasing approach speed from 65 to 80 knots intensifies the problem. Not only must guidance and control techniques be advanced significantly but information must be generated to enable trade-off studies between costs of advanced high-lift technology and advanced guidance and control technology so the best economic balance can be established.

Community Acceptance

It is quite likely that by the time a STOL system could be introduced noise-control procedures will be far more sophisticated than the simple two or three point controls now used. With industry, NASA has taken a first look at how these more complete controls might affect allowable noise levels. Table III shows levels of PNdB and frequency of flights for several classes of land use which might be considered unobtrusive. Although these absolute values cannot be considered conclusive, they do indicate that very low noise levels must be achieved if community acceptance is to be had.

An illustration of the operational flexibility for two aircraft configurations based on community acceptance to noise was obtained by examining noise exposure for various classes of land use when operating out of a typical airport where, at the present time, jet aircraft are not allowed to operate due to noise. See figures 9 and 10 where the vertical scale shows the increment in noise level above the background noise. The horizontal scale shows the area exposed to the noise in m^2 (acres). For each class of land use, a dashed horizontal line indicates the acceptable value for 100 flights a day determined from table III. The solid curved lines show landing approach area levels for current and advanced technology aircraft. If the goals are correct, current technology aircraft operation would not be accepted; whereas advanced technology aircraft would have complete community acceptance, at least from the standpoint of noise. This operational flexibility and community acceptance control the utility that STOL transport aircraft could provide and point to the urgent need for continued effort in noise reduction and development of flight profiles and techniques to reduce noise and time of operation in close proximity to the community adjacent to the air terminal.

CONCLUDING REMARKS

The results of the NASA/Industry studies indicate that the STOL transports are on the verge of becoming reality. Technical advances will do much to improve the aircraft for meeting the community acceptance requirements and to provide the utility and economic viability necessary for airline transportation during the 1980's.

First, the progress in the field of aerodynamics appears to be adequate to meet the high-speed cruise and low-speed take-off and landing performance as well as stability and control. Second, the progress in propulsion and noise reduction for STOL transport aircraft has accelerated in the past year or so, but much work is required in providing improved performance in terms of thrust-weight ratio with large reductions in propulsion system noise to meet the community acceptance levels. Third, much work is in progress related to advanced structures and materials that applies to STOL transport; however, the effort to reduce costs of composite structure fabrication for aircraft and engines should be increased to allow taking full advantage of these materials for reduction in DOC as well as gross weight. Fourth, in order that the STOL transport can take full advantage of its versatility and performance capability for reduction of terminal area noise and flight time required, the efforts in ground based and aircraft guidance and control system developments and flight tests in these areas should be increased. Fifth, all the foregoing factors lead to increased utility of the aircraft provided community acceptance is obtained, which should be the subject of indepth studies and experiments to attain realistic values of acceptable noise levels in various sections of the community.

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2. Fry, Bernard L.; and Zabinsky, Joseph M.: Feasibility of V/STOL Concepts for Short-Haul Transport Aircraft. NASA CR-743, 1967.
3. Lockheed-California Co.: Study on the Feasibility of V/STOL Concepts for Short Haul Transport Aircraft. NASA CR-902, 1967.
4. Boeing Co.: Study of Aircraft in Short Haul Transportation Systems. NASA CR-986, 1968.

TABLE I

EFFECT OF ADVANCED STRUCTURES AND MATERIALS

VARIABLE \ CONFIGURATION	CURRENT TECHNOLOGY AIRCRAFT	1985 TECHNOLOGY AIRCRAFT
TAKE-OFF GROSS WEIGHT, lb	203,000	176,000
(N)	(900,000)	(780,000)
OPERATING WEIGHT EMPTY, lb	147,600	123,000
(N)	(660,000)	(550,000)
WING AREA, sq ft	1790	1550
(m^2)	(166)	(145)
T/W	0.495	0.516
RELATIVE DOC	1.0	*0.89

* INCLUDES NO INCREASE IN FABRICATION COSTS FOR ADVANCED STRUCTURES

TABLE II

GUIDANCE AND CONTROL SYSTEMS FOR 2000 ft LANDING FIELD LENGTH (610 m)

APPROACH SPEED	80 knots	65 knots
ALLOWABLE DISPERSION DISTANCE, TOUCHDOWN	≤ ± 350 TO 400 ft (110 TO 120 m)	≤ ± 550 TO 600 ft (170 TO 180 m)
GUIDANCE AND CONTROL SYSTEM REQUIREMENTS	VERY ADVANCED	ADVANCED

TABLE III

URBAN NOISE GOALS
BASED ON LAND USE AND FREQUENCY OF OVERFLIGHTS

ITEM AND NORMAL BACKGROUND NOISE	LAND USE DESIGNATION (URBAN)	POTENTIAL GOALS		
		50 flt/day, PNdB	100 flt/day, PNdB	250 flt/day, PNdB
CLASS I 67 PNdB	RESIDENTIAL, SCHOOLS, LIBRARIES, THEATERS, CHURCHES, HOSPITALS	90	87	84
CLASS II 72 PNdB	OUTDOOR SPECTATOR SPORTS, SMALL OFFICES, RETAIL STORES, MOTELS, HOTELS	100	97	94
CLASS III 77 PNdB	OUTDOOR RECREATION (GOLF, PLAYFIELDS), LARGE BUILDINGS, COMMERCIAL, INDUSTRIAL	105	102	99
CLASS IV 77 PNdB	AGRICULTURE, PARKING, FREEWAYS, AIRPORTS	110	107	104

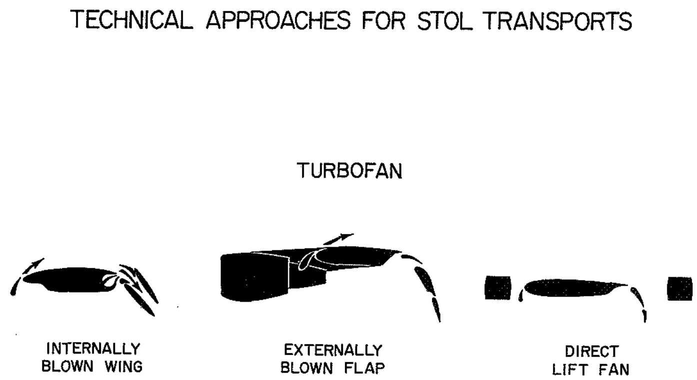

Figure 1

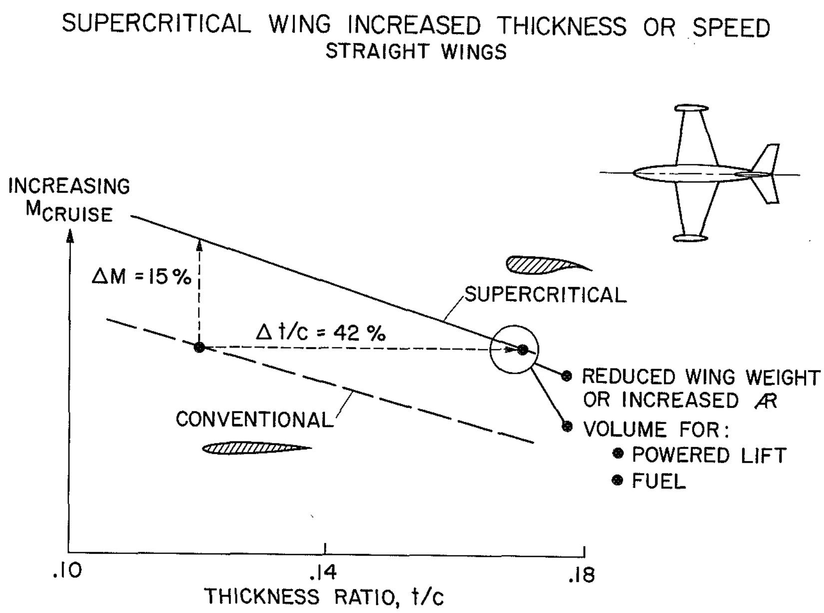

Figure 2

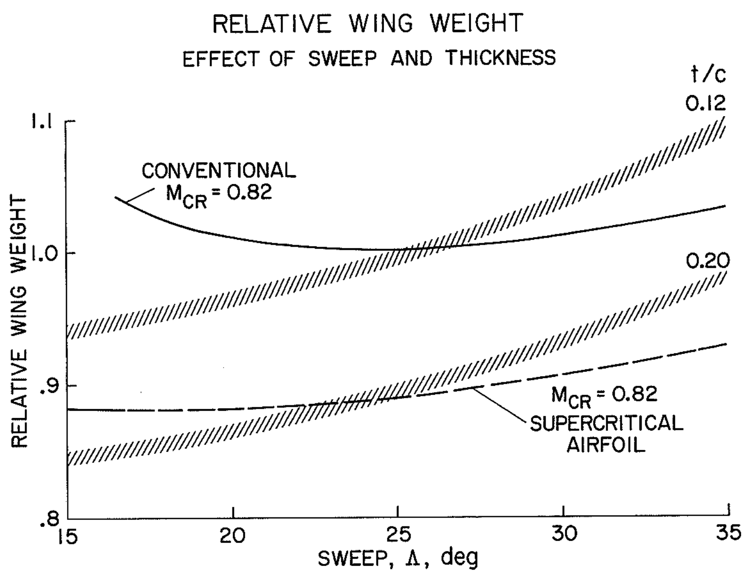

Figure 3

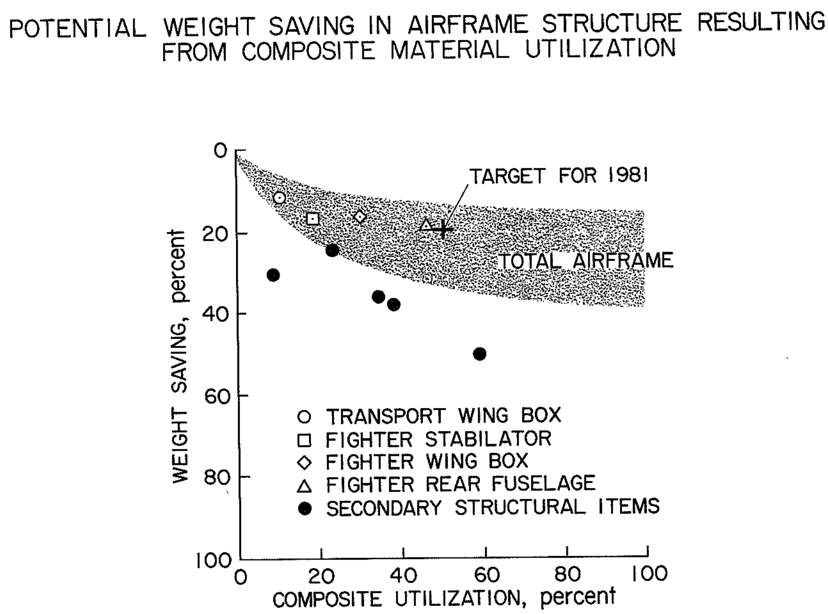

Figure 4

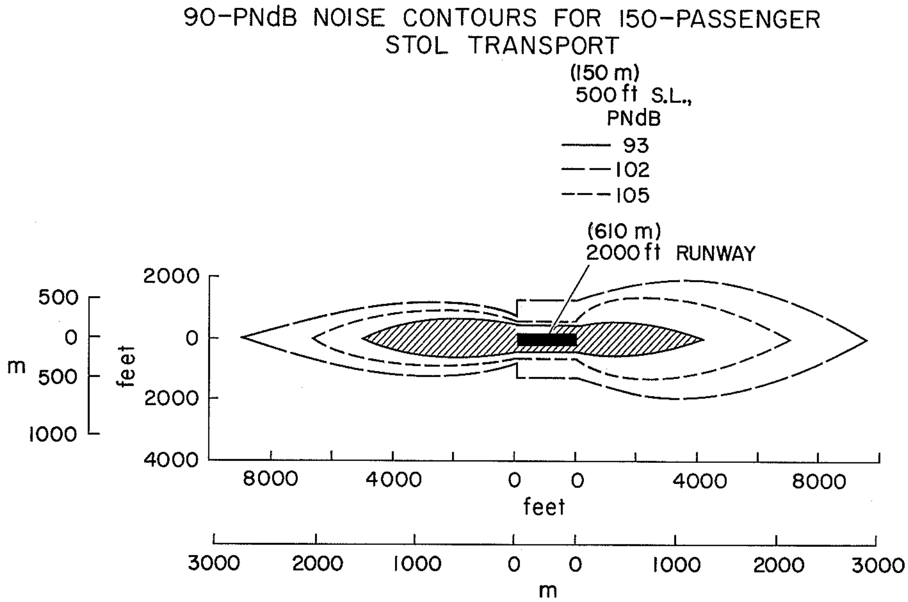

Figure 5

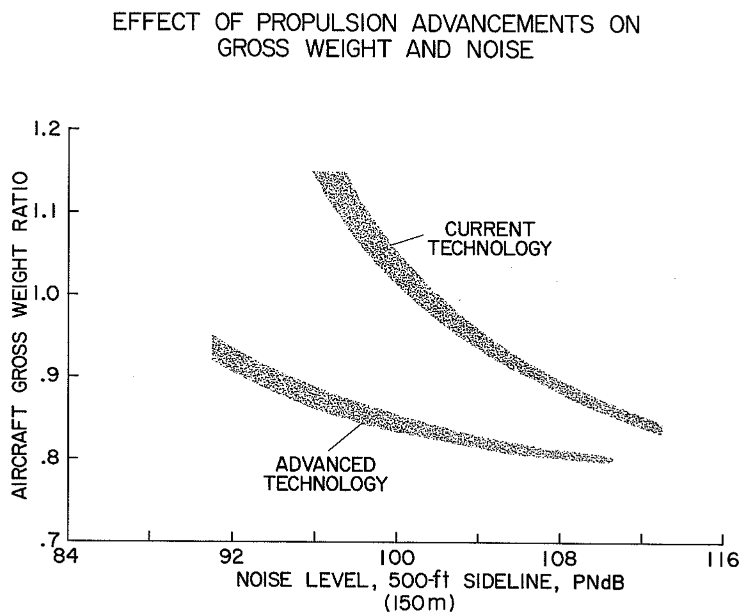

Figure 6

EFFECT OF ADVANCED TECHNOLOGY AND APPROACH SPEED ON GROSS WEIGHT

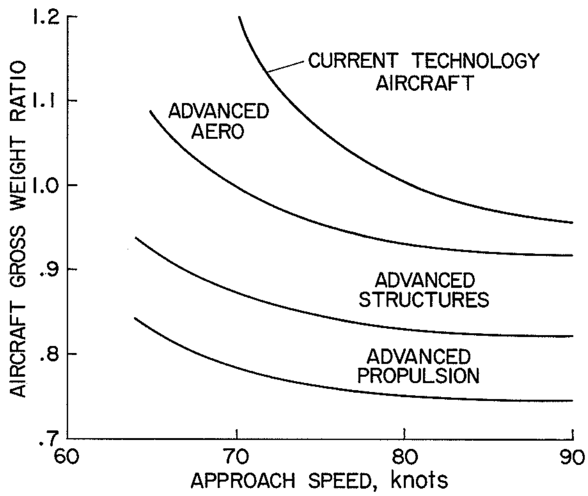

Figure 7

EFFECT OF TECHNOLOGY IMPROVEMENTS ON STOL TRANSPORT DOC

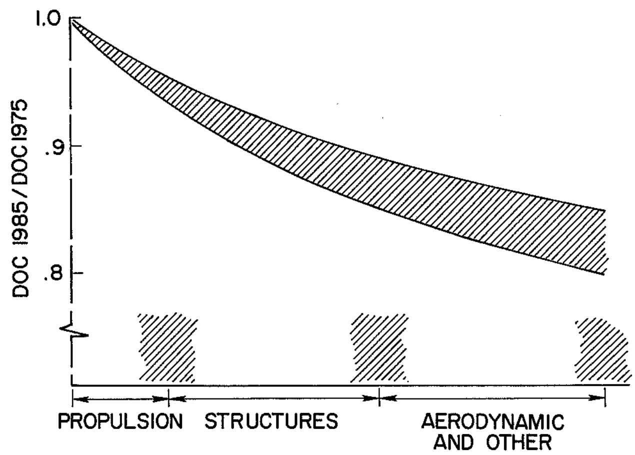

Figure 8

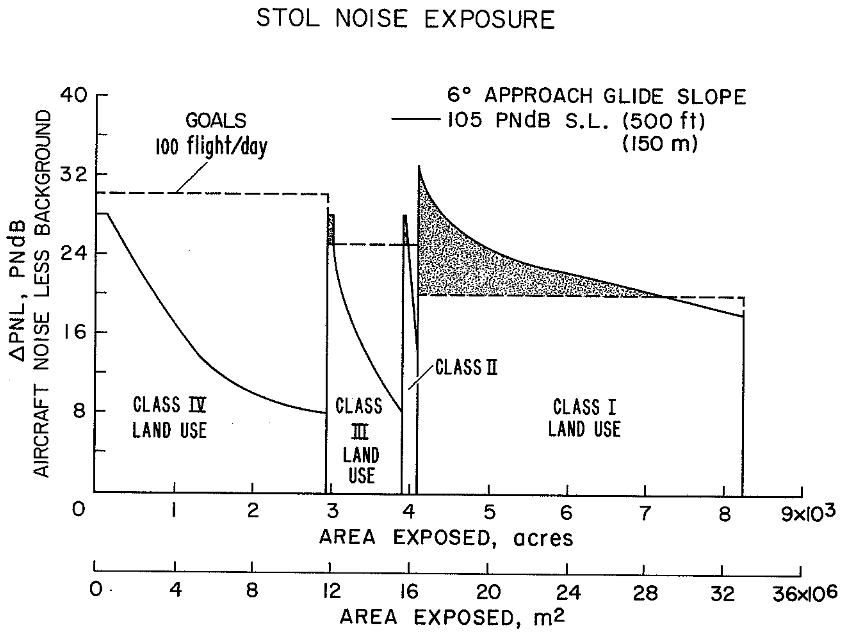

Figure 9

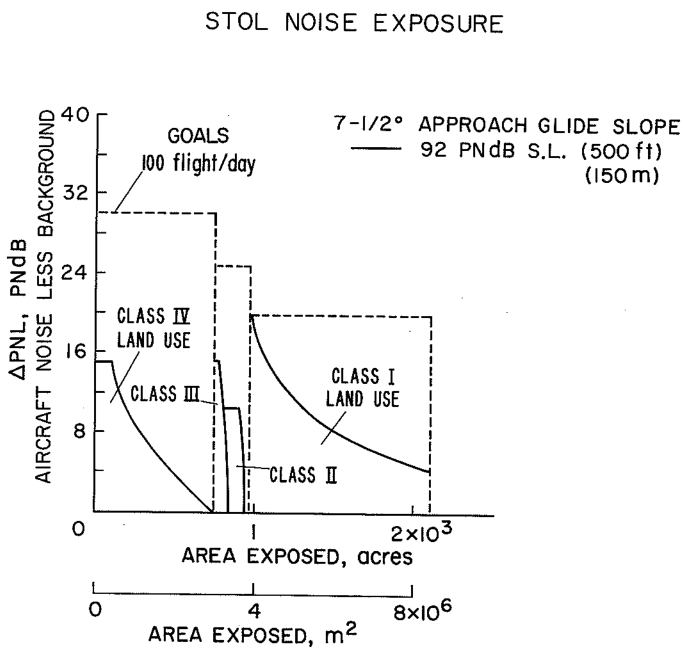

Figure 10

SUBSONIC-TRANSONIC TRANSPORT AIRCRAFT PROJECTIONS

By William J. Alford, Jr.
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22

INTRODUCTION

The purpose of this paper is to discuss the application of advanced technologies to subsonic-transonic conventional take-off and landing (CTOL) transport aircraft and to project what their characteristics might be in the 1980's. The goals to be demanded of aircraft of this time period are (1) reductions of the effects of noise and pollutant emissions on the environment and (2) the maintenance of competitive superiority in the world's aircraft market. Achievement of both of these goals is considered to be of critical importance to the nation. The discussion deals primarily with long-haul aircraft; however, brief consideration will be given to the outlook for the short/medium-haul class. The material presented is based on results from NASA in-house programs and from contracted advanced transport technology studies being conducted by The Boeing Company, General Dynamics Corporation, Lockheed-Georgia Company, General Electric Company, and Pratt & Whitney Aircraft. Since these studies will not be completed until early 1972, the present paper constitutes a status report.

ABBREVIATIONS AND SYMBOLS

CO carbon monoxide

THC total unburned hydrocarbons

NO nitric oxide

EPNdB effective perceived noise level

DOC direct operating costs as calculated by the 1967 Air Transport Association method using a labor rate of $5.00/hr, a domestic-crew cost constant of 160, an international-crew cost constant of 180, and a 15-year depreciation period with zero residual value

L/D lift-drag ratio

M Mach number

LONG-HAUL AIRCRAFT

Two configurations that are considered representative of what the next generation of long-haul aircraft may look like are shown in figure 1. The sketch at the bottom illustrates a 200-passenger version of a transcontinental-range aircraft designed to cruise at Mach numbers from 0.95 to 0.98; the sketch at the top illustrates an intercontinental-range aircraft designed to cruise at Mach numbers from 0.90 to 0.95. These aircraft most likely would be members of families that would have variations in range and payload, depending on the needs of the operating airlines.

Environmental Acceptability

As noted in the discussion of paper no. 7, the prime factors impacting the environment are engine noise and pollutant emissions. Promising approaches for reducing the objectionable characteristics are engine cycle selection and design, acoustic treatment of the inlet and exhaust systems, ground and flight operational procedures, and innovative airplane design techniques.

Emissions.- In table I are presented the prime pollutants (carbon monoxide, total unburned hydrocarbons, and nitric oxide), along with some NASA advanced transport technology (ATT) study goals that are being used on an interim basis, pending the publication of recommendations by the Environmental Protection Agency. In the last column are presented the levels that NASA and engine-company studies indicate to be achievable. The gas emissions are listed in terms of grams of pollutant per kilogram of fuel (pounds of pollutant per thousand pounds of fuel). The smoke particulates are indicated in terms of smoke number, where an index of 25 is associated with the threshold of smoke visibility. (See ref. 1.)

Emissions resulting from incomplete combustion during low-power engine operation are carbon monoxide and total unburned hydrocarbons. The levels indicated to be achievable are obtained with combustors that provide improved fuel atomization. These levels more than satisfy the NASA goals. For nitric oxide, whose formation is associated with high-power engine operation, the goal of 3 g/kg fuel (3 lb/1000 lb fuel) probably cannot be realized, even with advanced combustors, unless water injection is used. However, water injection of about 200 to 300 kg (500 to 700 lb) per engine for each take-off will be required to achieve this goal. Attainment of the smoke goal does not appear to be a problem.

Noise.- Predictions of the economic impact – in terms of direct operating cost (DOC) – of achieving varying amounts of noise reduction are shown in figure 2. The data presented apply to transports having transcontinental range and sized to accommodate approximately 200 passengers. Comparable data for transports of greater size and/or longer range probably would be generally similar although somewhat different in magnitudes.

The design noise levels are shown as increments of EPNdB below the current Federal Air Regulations presented in reference 2 (designated FAR 36 herein). DOC is expressed incrementally as percent of the DOC associated with a 10-EPNdB reduction below FAR 36, since it is anticipated that a reduction of that magnitude will be demanded in the early 1980's.

The data indicate that a reduction of 10 EPNdB from FAR 36 will be accompanied by approximately a 3.5-percent increase in DOC. To reduce the noise by 20 EPNdB, with only improved engine design and acoustic treatment being used, will incur an additional increase of 15 percent in DOC. By utilizing flight operational procedures, such as steep climbouts and power reductions and steep and curved descents, significant DOC reductions are possible (about 5 percent at -20 EPNdB).

Some recent exploratory configuration research indicates the possibility that innovative airplane design can further reduce engine noise. Two such concepts are shown in figure 3. These airplanes would incorporate the foregoing techniques and technology in combination with engines buried in the wings (a potential reduction of 2 to 4 EPNdB) or with the engines mounted over the wing (a potential reduction of 1 to 2 EPNdB).

Competitive Superiority

Superiority relative to competitive systems implies a potential for greater profit. Profit, of course, is sensitive to costs and to passenger load factor which, in turn, is sensitive to speed, fares, and comfort. Technology advances in aerodynamics, structures, and controls offer potential gains in both costs and passenger appeal. Supercritical technology, for example, can increase cruise Mach number, increase cruise lift-drag ratio, or reduce wing weight. Advanced materials can reduce structural weight, and active controls can also reduce structural weight as well as improve ride comfort. The propulsion system characteristics and environmental constraints considered in the following section are consistent with those already identified in the discussion of paper no. 7.

Envelopes of the potential full-scale cruise lift-drag ratios, based on the supercritical aerodynamic technology discussed earlier in paper no. 3 by Edward C. Polhamus, are presented as a function of Mach number in figure 4. For these data the wing sweep angle, aspect ratio, and thickness have values appropriate to the corresponding cruise Mach numbers. For reference, L/D values representative of today's wide-body transports are also presented.

For speeds comparable to today's levels, or slightly higher (that is, $M = 0.85$ to 0.90), supercritical technology can be used to allow increases in wing thickness, with corresponding reductions in wing weight while maintaining essentially the same L/D levels, or increases in aspect ratio and thereby increases in L/D. Alternatively, by using thinner sections and increases in wing sweep, current wind-tunnel tests indicate L/D values in the range of 14 to 15 to be feasible at a cruise Mach number of 0.98. Based on past experience, it is expected that additional design refinements should raise the attainable cruise L/D to the level of about 16.

To illustrate the sensitivity of direct operating costs to lift-drag ratios at $M = 0.98$, where supercritical technology is required, figure 5 presents the characteristics for a design range of 3000 n. mi. and a passenger capability of 200. For reference, the DOC level estimated to be representative of a current wide-body jet transport with the same payload and range, but cruising at about $M = 0.85$, is presented. The advanced high-speed transport, with an L/D slightly below 15, is seen to provide the same DOC as the current transport; however, at the projected L/D of 16, the advanced transport has an advantage of about 7 percent in DOC. It must be borne in mind, however, that, in addition to its speed advantage, the advanced transport has been designed for engine noise levels 10 EPNdB below FAR 36.

Comparisons of the estimated effects on direct operating cost of applications of supercritical technology (SCT), active control systems (ACS), and advanced composite materials are presented in figure 6. These results are based on estimates that a 6-percent structural weight reduction will result from use of the ACS as described by A. Gerald Rainey in paper no. 13 and a 20-percent structural weight reduction will result from a 50-percent utilization of composites as described by Richard A. Pride in paper no. 10. An L/D of 16 is assumed attainable with supercritical technology. As shown in figure 5, supercritical technology alone provides a reduction of 7 percent. Adding ACS allows a further 3-percent reduction; whereas composites, without ACS, gives an improvement of 6 percent. In combination, the total reduction in DOC is about 12 percent relative to the current wide-body transport aircraft. The individual technology benefits are not directly additive because the DOC also depends upon the initial cost of each technology and the aircraft gross weight. As previously mentioned, the advanced transport also has an advantage of about 10 EPNdB in noise.

Figure 7 presents comparisons of technology benefits for an advanced transport designed for intercontinental range, a capacity of 400 passengers, and a design Mach number of 0.95. The reference shown in the left-hand side of the figure is representative of a current four-engine aircraft, also estimated to carry 400 passengers. Supercritical technology, of course, is required to attain the speed of Mach 0.95 but its advantage in wing structural efficiency does not completely counteract relatively larger weight increases necessary for increasing the speed of a large quiet intercontinental aircraft as compared with the smaller transcontinental aircraft. For example, the weight penalty for a 10-EPNdB noise attenuation is greater because of the larger thrust required; the fuselage area ruling required increases the fuselage length (and, therefore, weight) necessary for seating a given number of passengers; and the overall aircraft fineness ratio and resulting weight need to be larger for the higher speed. The utilization of composite material on the heavier aircraft, however, offers a greater advantage than for the lighter aircraft. When 40-percent utilization of composite material is combined with supercritical technology, a 20-percent reduction in structural weight and an associated 30-percent reduction in take-off gross weight are attainable. An 18-percent decrease in DOC relative to the Mach 0.85 aircraft results, as indicated in the figure.

Advanced Transport Technology Program

The results presented so far have been generated primarily from the NASA advanced transport technology program, which is outlined as follows:

<u>APPROACH</u>

- Systems Studies (Airframe and Engine)
- Exploratory Flight Tests
- Fundamental Technology

<u>TECHNICAL AREAS</u>

- Aerodynamics
- Propulsion
- Structures/Materials
- Avionics
- Control Systems
- Design Integration
- Economic Cost/Benefit

This program constitutes an effort to expedite the development of advanced technologies to insure that the next generation of subsonic-transonic CTOL aircraft can meet the goals of achieving environmental acceptability and competitive superiority in the world's aircraft market. The approach being taken includes (1) systems studies (both airframe and engine); (2) exploratory flight tests, such as have been carried out in a joint Navy/NASA program using a T-2C trainer modified to incorporate a thick supercritical wing and in current NASA tests of a modified F-8 aircraft incorporating a high-performance supercritical transport wing; and (3) fundamental technology efforts in all of the technical areas outlined above.

The results desired and expected from these efforts, which are being accomplished through in-house NASA programs and through contracted studies being carried out by the airframe and engine companies, are (1) identification of the promising advanced aeronautical technologies, (2) their associated problems and weaknesses, (3) suggestions for needed laboratory and flight research, and (4) estimates of the required resources and schedules to meet the needs of next-generation transports. Photographs of airplane concepts being studied by the airframe-company contractors are shown in figures 8 to 10.

Figure 8 presents a photograph of a 200-passenger, $M = 0.98$, transcontinental-range aircraft being studied by The Boeing Company. In their work they are stressing advanced aerodynamic configuration applications as affected by design Mach number and terminal-area considerations. Figure 9 presents a photograph of a similar size aircraft and for the same mission, as conceived by General Dynamics Corporation. In their studies they are stressing structures and materials for both area-rule and non-area-rule designs and advanced control systems. Figure 10 presents a photograph of a large long-range aircraft, being studied by Lockheed-Georgia Company, that would cruise at $M = 0.95$. Their studies are concentrating on evaluations of integrated airplane systems design, passenger-cargo relationships, and terminal-area interface considerations for large area-rule aircraft.

SHORT/MEDIUM-HAUL AIRCRAFT

Recent trends in air travel indicate a need for new short/medium-haul transports to serve the high-density traffic demands between large city pairs. Although the impact of advanced technologies has not yet been adequately defined for these aircraft, the results of the ongoing ATT program will be largely applicable to the short/medium-haul CTOL aircraft systems. Some additional work is required, however, to address problems associated with operational considerations of these aircraft.

The configuration illustrated in figure 11 is considered to be representative of the short/medium-haul class of transports. Pertinent parameters include: design ranges from 800 to 1200 n. mi., cruise Mach numbers from 0.80 to 0.85, passenger capacities of 200 to 300, and field lengths from 1.2 to 1.5 km (4000 to 5000 ft).

With regard to special features, environmental considerations will again constitute the prime constraint. Because of the large payloads and the desire for passenger comfort, wide bodies are likely to be used. Because of their moderate cruise speeds, these aircraft can realize the advantages associated with the thick supercritical wings, such as reduced wing weights or increased L/D resulting from aspect ratio increase.

Since these aircraft will spend a relatively large proportion of their block time in the terminal area, it is likely that advantages can be derived through incorporation of advanced avionics in both the airframe and ground systems to provide improved terminal-area operating efficiency. Additional studies are needed to determine the payoff from improving terminal-area related technologies and these studies should include consideration of benefits possible from terminal-area configured and control-configured vehicles. The use of design approaches to improve ride comfort and alleviate structural fatigue loads also deserves attention.

CONCLUDING REMARKS

In conclusion, it appears that the next generation of long-haul transports may be substantially improved over the present family through application of several advanced technologies. The more important technologies are being identified and developed under current programs. Present indications are that the development and application of composite materials in combination with supercritical technology is likely to provide the greatest payoff for these transports.

New short/medium-haul transports appear to be needed to serve the expanding market between large city pairs. The benefits of advanced technologies have not yet been adequately defined for these aircraft. It is probable, however, that features related to terminal-area performance will be of primary importance.

REFERENCES

1. Anon.: Aircraft Gas Turbine Engine Exhaust Smoke Measurement. ARP 1179, Soc. Automot. Eng., Inc., May 4, 1970.
2. Anon.: Noise Standards: Aircraft Type Certification. Federal Aviation Regulations, vol. III, pt. 36, FAA, Dec. 1969.

TABLE I

POLLUTANT EMISSIONS

POLLUTANT	ATT GOALS g/kg fuel (lb/1000 lb fuel)	ACHIEVABLE g/kg fuel (lb/1000 lb fuel)
GASEOUS:		
CO	40	5 TO 20
THC	8	4 TO 6
NO	3	20 TO 30 (WITHOUT H_2O)
		3 (WITH H_2O)
PARTICULATE:		
SMOKE NUMBER	15	12 TO 15

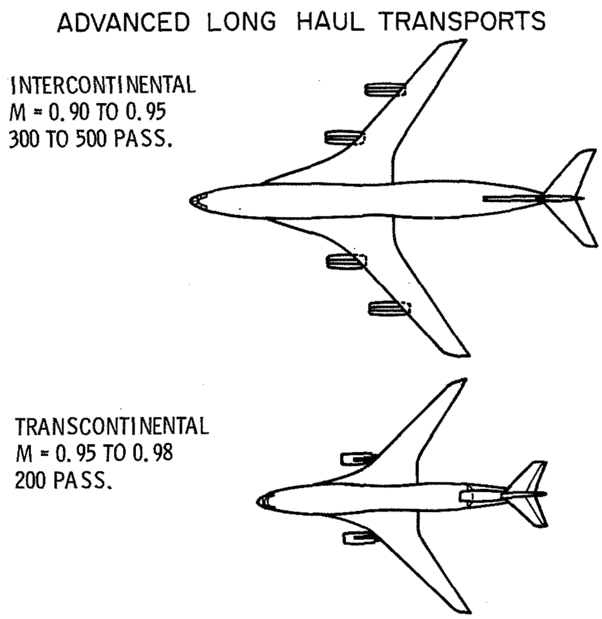


Figure 1

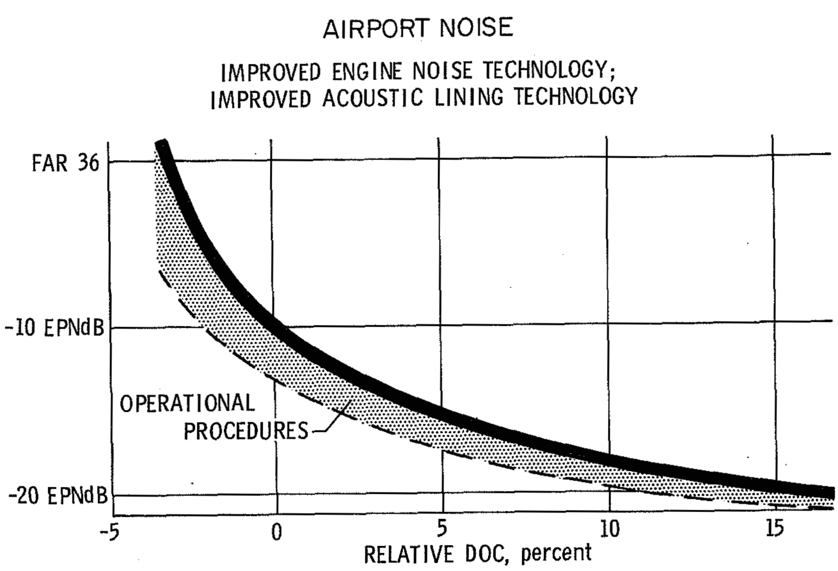


Figure 2

QUIET AIRPLANE CONCEPTS

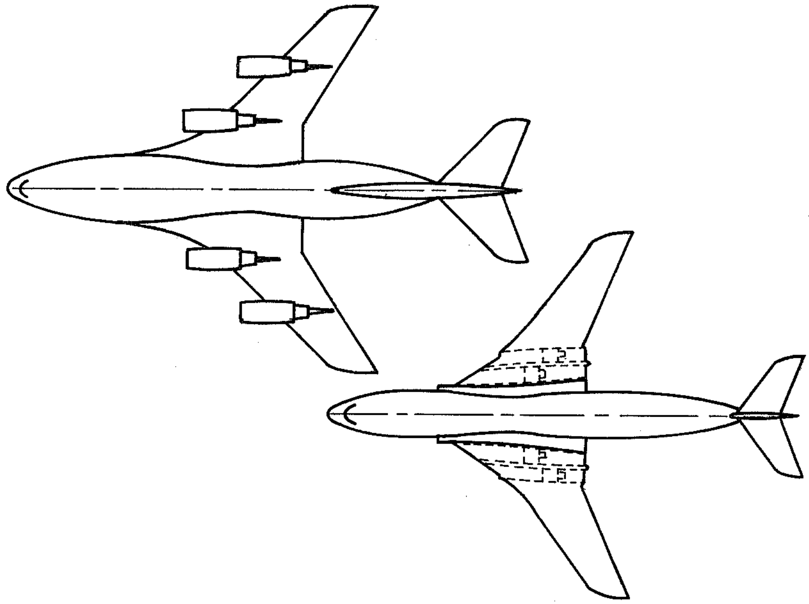


Figure 3

AERODYNAMIC EFFICIENCY

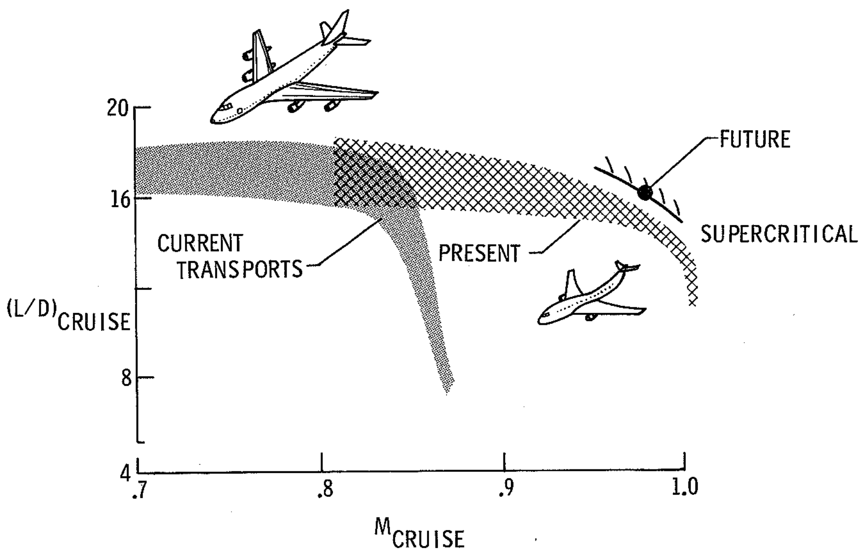


Figure 4

APPLICATION OF SUPERCRITICAL TECHNOLOGY

M = 0.98; R = 3000 n.mi.; PASS. = 200;
NOISE LEVEL = FAR 36 -10 EPNdB

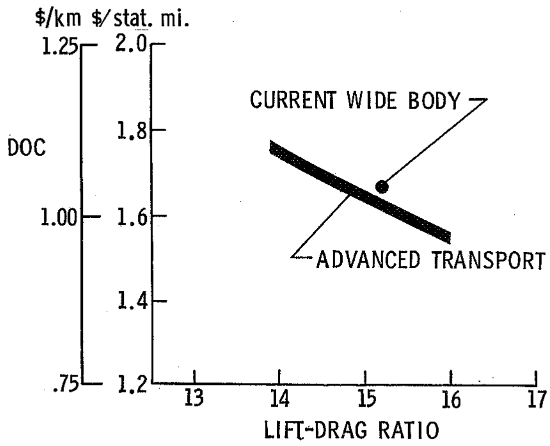


Figure 5

COMPARISON OF ADVANCED TECHNOLOGIES

R = 3000 n. mi.; PASS. = 200

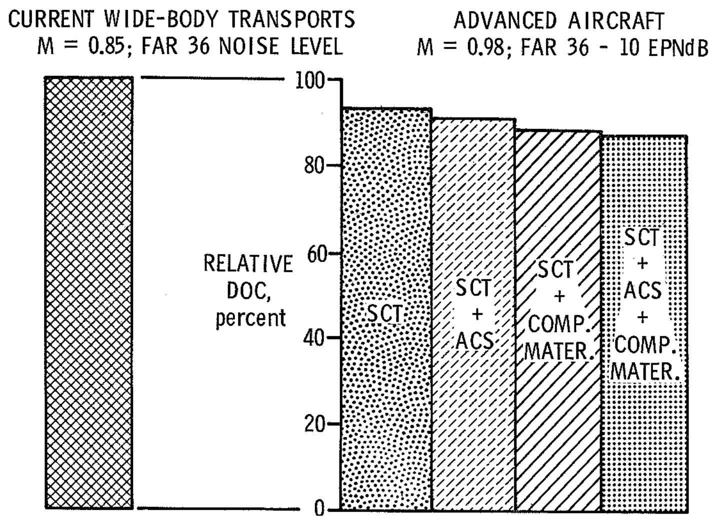


Figure 6

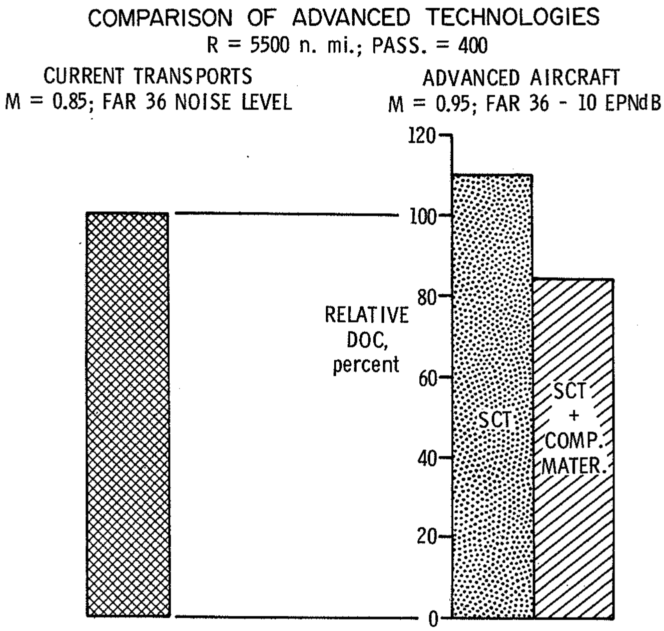


Figure 7

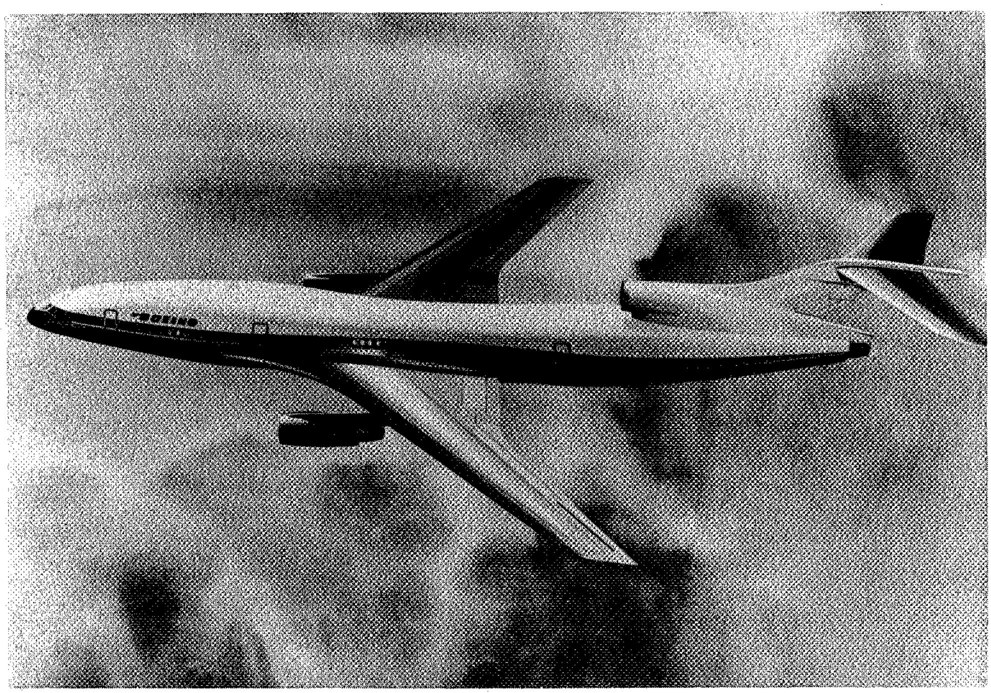


Figure 8

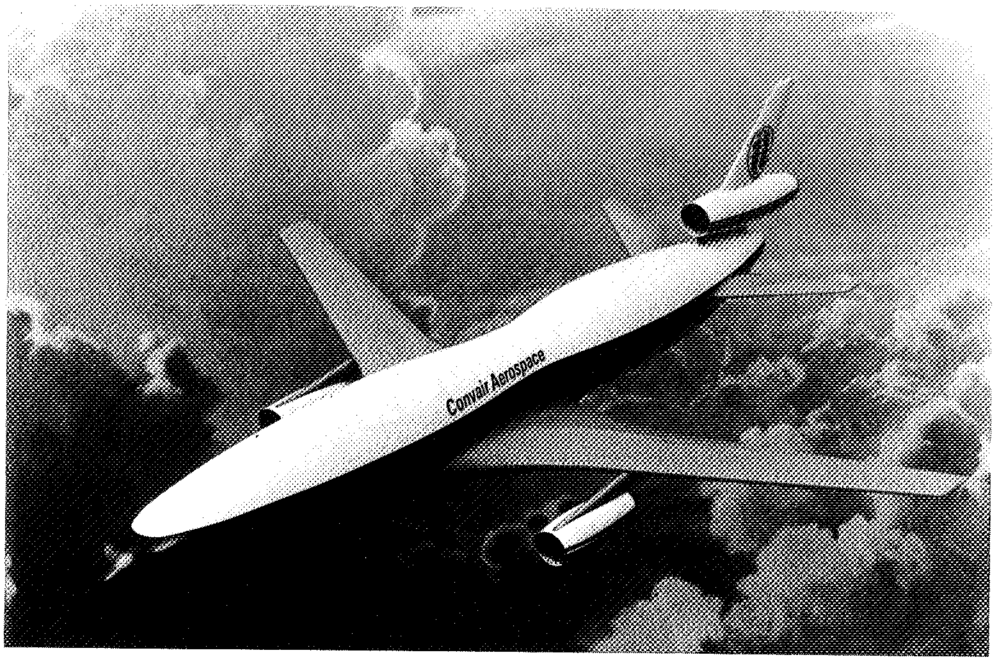


Figure 9

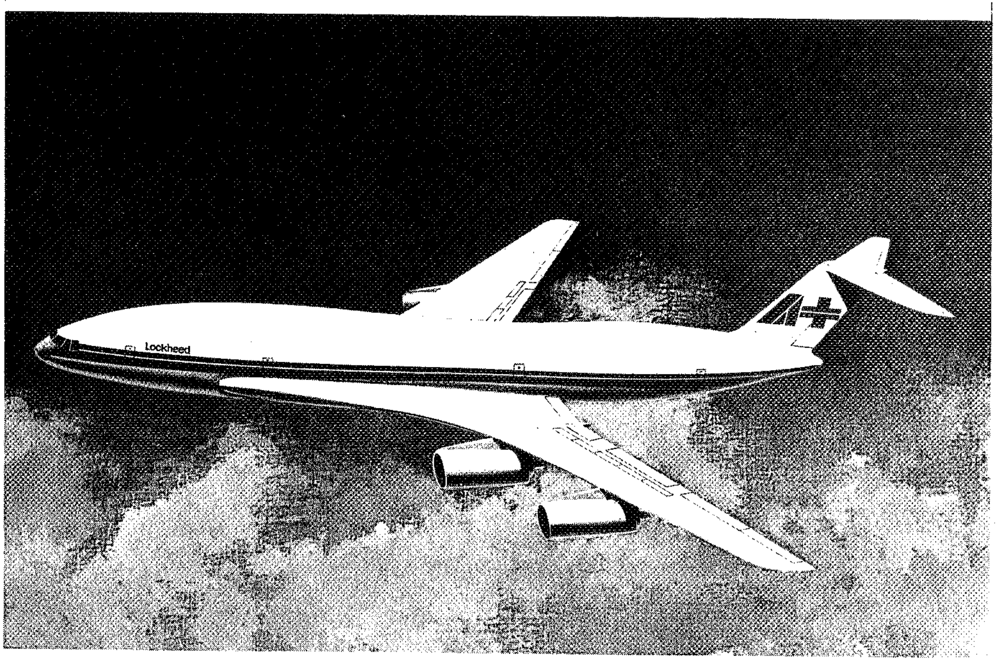
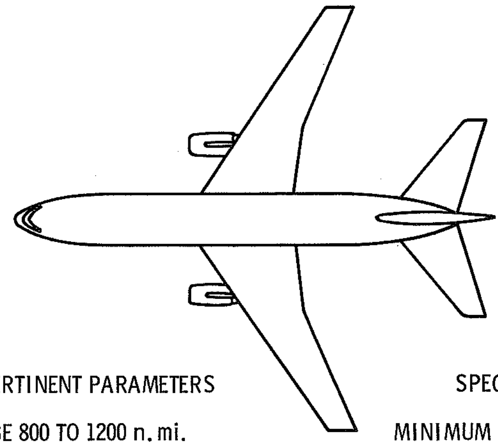


Figure 10

ADVANCED SHORT/MEDIUM HAUL TRANSPORT



PERTINENT PARAMETERS

RANGE 800 TO 1200 n. mi.

CRUISE MACH NO. 0.8 TO 0.85

PASSENGERS 200 TO 300

FIELD LENGTH 4000 TO 5000 ft
(1.2 TO 1.5 km)

SPECIAL FEATURES

MINIMUM ENVIRONMENTAL IMPACT

WIDE BODY

THICK SUPERCRITICAL WING

ADVANCED TERMINAL AREA AVIONICS

RIDE QUALITY CONTROL

Figure 11

PROPERTIES OF OBLIQUE-WING—BODY COMBINATIONS FOR LOW SUPERSONIC SPEEDS

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SUMMARY

In theory, antisymmetric arrangements of wings and bodies can have smaller wave drag than corresponding mirror-symmetric arrangements. Thus, a long narrow oblique wing which presents the same aspect for two opposite directions of flight is potentially more efficient than a corresponding (that is, structurally equivalent) swept wing. The single continuous wing panel also adapts itself more readily to varying angles of obliquity and hence to varying flight speeds.

The present paper reviews work on the aerodynamics and flight stability of oblique-wing combinations and suggests a possible mode of application to transport aircraft operating at moderate supersonic speeds.

INTRODUCTION

One of the unspoken assumptions in aircraft design is that of bilateral or mirror symmetry. At slow flight speeds, this assumption seems on rather secure ground not only because of the indications of aerodynamic theory but also because it agrees with the observed evolutionary forms of birds.

Although it is perhaps natural to extrapolate the forms of birds and animals to the supersonic flight regime, there has been no rational discussion of the merits of bilateral symmetry for supersonic flight. In fact, once the velocity of sound is exceeded, the laws of aerodynamics change in such a way as to make it seem inadvisable to arrange the components of an airplane side by side or abreast in a supersonic stream unless there are compelling reasons for such an arrangement.

Both the transonic area rule and the supersonic small-disturbance theory show large adverse interference effects for bodies or wings in a mirror-symmetric arrangement. Figure 1 shows the result of applying supersonic wave drag theory to two airplanes flying in close formation at a slightly supersonic Mach number. In the mirror-symmetric arrangement the drag of each aircraft is doubled by the interference of the other and thus results in a total wave drag of four. In the staggered or antisymmetric arrangement, however, the wave interference is favorable; thus, the drag of the two airplanes is no greater than that of a single one.

Figure 2 shows the same effect for oblique wing panels. The arrow shape, which seems intuitively correct for supersonic speed, nevertheless, has a predicted wave drag many times larger than the antisymmetric arrangement.

Elements of lift or volume show favorable wave interference if they are disposed along lines for which the normal component of velocity is subsonic. Thus, the wave drag of a long narrow wing tends toward zero if the wing is swept behind the Mach cone. (See ref. 1.) The reversibility of the wave drag (refs. 2 and 3), however, indicates that a distribution of lift or volume having a minimum drag should show the same aspect for two opposite directions of flight; that is, it would have fore-and-aft symmetry. Consideration of the vortex drag indicates further that the projected lift distribution should have lateral symmetry (for example, elliptic span loading).

It is interesting that supersonic theory favors symmetry in both longitudinal and lateral distributions of volume or lift, but evidently not mirror symmetry. Intuitively, one feels that a supersonic airplane should take account of the direction of flight in its shape; that is, it should somehow "point" in the direction that it is going. However, in view of the reversibility of the wave drag, current computer programs must give the same value of the drag with the direction of flight reversed. Figure 3 illustrates this result. The reversibility theorems are, of course, limited to the pressure drag and the lift-curve slope as determined by linear theory. Thus, the effect of viscosity demands locally different shapes for leading and trailing edges, which are not reversible in practice.

SYMBOLS

a,b	major and minor semiaxes of ellipse
C_D	drag coefficient
ΔC_D	incremental drag coefficient
C_{D_0}	drag coefficient at zero lift
C_L	lift coefficient
C_{L_n}	lift coefficient based on normal component of velocity
D	drag
$i = \sqrt{-1}$	
L	lift

L/D	lift-drag ratio
M	Mach number
p	pressure
p_a	ambient pressure
p_L	lifting pressure
q	dynamic pressure
R.P.	real part
S_W	wing area
t	thickness
V	velocity
$\overline{X}_1,\overline{X}_2$	averaged lengths of $X(\theta)$
Y	wing span
x,y	coordinate axes
α	angle of attack
$\beta = \sqrt{M^2 - 1}$	
γ	ratio of specific heats
θ	angle between Mach plane and X-axis
ψ	complement of angle of yaw

REVIEW OF AERODYNAMIC PROPERTIES OF OBLIQUE ELLIPTIC WINGS

To obtain a configuration having a minimum wave drag, it is supposed that (1) the total lift and volume are given and (2) there is a plane area within which the dimensions must be limited. As a solution of this problem it is found that for any area bounded by two streamlines and two characteristic lines, the distribution of lift and volume yielding the minimum pressure drag (that is, wave drag plus vortex drag) places all the elements of lift and volume near a diagonal lifting line. Such a diagonal line may be considered the limiting configuration of a narrow elliptic wing as illustrated in figure 4. Minimum drag occurs when the surface loading of the ellipse is constant and when the thickness is distributed so that the projected cross-sectional areas are those of a Sears-Haack body. (See refs. 4 to 6.)

The foregoing result is of interest not so much as an exact prescription of shape but because it indicates that lift and volume can be concentrated within a narrow dimension having a small wetted area and hence small friction drag, provided the "lifting line" extends in a "subsonic" direction. (Linear theory shows an infinite drag if the line becomes supersonic.)

The favorable properties of the oblique wing depend first of all on the maintenance of a subsonic type of section flow at supersonic speeds, and this condition requires that the wing be placed at an angle of yaw so that the component Mach number normal to its long axis is subsonic. If one assumes that the critical "drag divergence" Mach number of the wing sections is 0.7, the angle of yaw must be such as to reduce the component Mach number M to this value. At $M = 1.0$, the angle of yaw required is then 45°.

The advantage of the yawed wing over the swept wing depends on an increased extension of the wing in the flight direction. As is well known, spreading the lift over a greater length diminishes both the sonic-boom intensity and the drag. For a given structural slenderness, the single yawed wing panel may have nearly twice the projected length of the corresponding swept wing.

The foregoing statements may be made more quantitative by referring to various components of the drag as given by linear theory; namely,*

$$\text{Drag} = C_{D_0} q S_W + \frac{L^2}{\pi q Y^2} + \frac{M^2 - 1}{2\pi q}\frac{L^2}{\overline{X}_1{}^2} + \frac{128q}{\pi}\frac{(\text{Volume})^2}{\overline{X}_2{}^4} \tag{1}$$

In equation (1), S_W is the wing area, Y is the span, $\overline{X}_1$ and $\overline{X}_2$ are averaged lengths $X(\theta)$ of the wing as projected by characteristic planes (Mach planes) set at different angles θ around the X-axis. The lengths $\overline{X}_1$ and $\overline{X}_2$ are defined by

$$\frac{1}{\overline{X}_1{}^2} = \frac{1}{\pi}\int_0^{2\pi} \frac{\sin^2\theta}{X(\theta)^2}\, d\theta \tag{2}$$

*The distributions of lift and volume assumed in equation (1) are those giving the smallest drag consistent with the geometric constraints X and Y.

$$\frac{1}{\overline{X}_2{}^4} = \frac{1}{2\pi}\int_0^{2\pi} \frac{d\theta}{X(\theta)^4} \tag{3}$$

At low supersonic Mach numbers and large angles of sweep or yaw, the lengths $\overline{X}_1$ and $\overline{X}_2$ are close to the actual X-wise extension or length of the wing. Hence, the wave drag due to the lift diminishes approximately as the inverse square of the length, whereas the wave drag due to volume goes down with the inverse fourth power.

The second term of equation (1) is the well-known linear formula for the induced drag of a wing having an elliptic span-load distribution. The rules determining the form of large birds, sailplanes, and other subsonic aircraft are evident from the first two terms of equation (1). Here one tries to maximize the span Y and to minimize the wetted area $(2S_W)$ by reducing the width of the wing in the flight direction. According to the linear theory (induced drag theory), the drag of the wing at subsonic speeds is independent of either the extension or the distribution of lift in the flight direction. Hence, the long narrow straight wing or lifting line is ideal at subsonic speeds since it minimizes the wetted area. The success of the rule for increasing L/D by increasing the aspect ratio depends, however, on the maintenance of Kutta-Joukowsky flow. If one tries to approach the lifting line too closely, the lifting pressure becomes excessive, and nonlinear effects associated with flow separation or shock losses will intervene. In spite of these limiting phenomena, sailplanes with extreme proportions have achieved L/D ratios as high as 40 or 50 to 1.

The counterpart of the lifting line at supersonic speeds is the oblique lifting line mentioned earlier. Here the appearance of the wave drag (third and fourth terms of eq. (1)) requires that the wing have as great a length as possible in addition to a wide span and a small surface area. The rules determining the optimum wing form are then similar to those determining the form of a sailplane, except that at supersonic speeds one tries to maximize both the span and the length in the flight direction together with a minimum surface area.

Again, if one tries to approach the idealized lifting line too closely, nonlinear phenomena will intervene. The lifting pressure may exceed the maximum lift coefficient of the sections, or if the crosswise component Mach number is too high, supercritical shock losses will appear.

For the wing of elliptic planform the pressure drag associated with the lift (wave drag and vortex drag) is a minimum when the lift is distributed uniformly over the surface. The formula given by linear theory in this case is (from ref. 5)

$$\Delta C_D = \frac{C_L{}^2}{4}\,\mathrm{R.P.}\left[\beta^2 - \left(m + i\,\frac{a'}{b'}\right)^2\right]^{1/2}$$

where

$$\beta^2 = M^2 - 1$$

$$m = \frac{\left(b^2 - a^2\right)\sin\psi\cos\psi}{b'^2}$$

$$b' = \left(a^2\cos^2\psi + b^2\sin^2\psi\right)^{1/2}$$

$$a'b' = ab$$

and a and b are the major and minor semiaxes of the ellipse and ψ is the complement of the angle of yaw. (See fig. 5.)

Figure 6 shows variations of drag due to lift with angle of yaw for an elliptic wing with an axis ratio of 10 to 1. At $M = 1.0$, the value shown is simply the induced drag, or vortex drag, which is, of course, large at large angles of yaw because of the small span. Also shown on the curves are the angles of yaw at which the crosswise Mach number exceeds an assumed critical value of 0.7.

To obtain a reasonably uniform distribution of lifting pressure at large angles of yaw, the wing must be constructed with a certain camber and twist. Calculations of the required camber and twist have been made by Ralph Carmichael and A. D. Levin of Ames Research Center by utilizing the technique of reference 6. The amount of twist indicated for a yaw angle of 45°, a lift coefficient of 0.5, and a Mach number near 1.0 is illustrated in figure 7. It is seen that the forward going tip must have a positive angle of attack while the angles of the rearmost sections are negative. A practical way to provide an effective twist that automatically increases with yaw angle is to construct the wing with a certain amount of upward curvature in its unyawed aspect, as shown in the figure. Linear theory is useful for determining gross shape parameters but cannot, of course, be relied on for critical details. Thus, for the sections of the wing perpendicular to its long axis, one would select airfoil shapes capable of sustaining a high lift coefficient at a high value of the crosswise component Mach number. Increasing the sweep angle will decrease the component Mach number; however, the lift coefficient based on the reduced component velocity will then increase. These considerations involve the critical Mach number and the lift coefficient in combination and point to the choice of airfoil sections that can develop a high ratio of absolute lifting pressure p_L to ambient pressure p_a, that is,

$$\frac{p_L}{p_a} = \frac{\gamma}{2} M^2 C_L$$

without drag penalty.

The minimum wave drag for a given internal volume of the elliptic wing occurs when the thickness ratio of the sections falls off elliptically toward the tips. (See ref. 5.) The formula for the drag due to thickness or volume in the case of the yawed ellipse is given by J. H. B. Smith (ref. 7) and the results are plotted in figure 8 for an axis ratio of 10 to 1 and a root thickness-chord ratio of 0.1.

The wave drag associated with the volume of the wing shows a steep rise as the long axis of the wing turns into the wind. However, the drag increase associated with nonlinear or supercritical flow over the wing sections may dominate so that the prediction of linear theory will not be adequate. It is here that the newer developments in supercritical wing sections exemplified by the work of Piercey, Niewland, and Whitcomb may be significant for the antisymmetric wing.

The drag values given by linear theory together with a suitable estimate of the skin friction enable the prediction of lift-drag ratios of elliptic wings at various Mach numbers and yaw angles. Such predictions will be valid if proper account is taken of the limitations imposed by nonlinear phenomena. Figure 9 (taken from ref. 8) shows such estimates for an ellipse with an axis ratio of 10 to 1, 10-percent thickness, and a friction drag coefficient of 0.005. The dashed curves show the effect of limiting the section lift coefficients to values of 1.0 and 0.5. Figure 10 shows how the increase of L/D with axis ratio depends on the possibility of achieving rather high section lift coefficients.

Lift-drag ratios for slanted elliptic wings at $M = 2.0$ have been calculated by J. H. B. Smith (ref. 7) and are quoted by D. Küchemann (ref. 9). Figure 11, adapted from reference 7, shows results of this calculation. At $M = 2.0$, peak lift-drag ratios occur at $\psi = 15^{\circ}$ to 20° which correspond to sweep angles of 70° to 75°. The optimum crosswise Mach number indicated by linear theory is approximately 0.7; this value is close to a limit imposed by nonlinear effects.

WIND-TUNNEL TESTS OF YAWED-WING COMBINATIONS

Few wind-tunnel experiments have been made to test the predictions for oblique wings of high aspect ratio. One set of experiments testing yawed and swept wings in conjunction with a fuselage was made by George H. Holdaway and Elaine W. Hatfield (ref. 10) at Ames Research Center. The angle of sweep or yaw in these tests was 40° and the wing thickness-chord ratio was approximately 11 percent. Figure 12 shows the drag at zero lift for two of the combinations tested. At $M = 1.0$, the antisymmetric configuration has much smaller drag, as expected. At $M = 1.15$, however, the normal component of M is approximately 0.88 and exceeds the drag rise Mach number of the sections. Beyond this point the drag of the yawed wing is higher than that of the swept wing.

When the advantages of subsonic sweep first became evident, questions were raised about the possibility of flying an airplane with the wing set at a large angle of yaw. Perhaps the earliest experiments to test the flight stability of such an arrangement were made in 1946 by John P. Campbell and Hubert M. Drake in the Langley free-flight tunnel. (See ref. 11.)

Campbell and Drake found that the yawed wing avoided the large rolling moment due to sideslip and the consequent short-period rolling oscillations of the swept wing. They noted that the flight characteristics of the model remained essentially unchanged up to angles of yaw of 40° and were still satisfactory at 50°. Of special interest is their observation that deflection of the ailerons produced no observable pitching motion in free flight. Evidently, the change of longitudinal lift distribution produced by deflecting the ailerons is almost immediately canceled by the rolling motion of the model. The wing, in effect, simply follows the helix angle defined by an effective twist associated with the aileron deflection with no significant change in lift distribution. The longitudinal stability and the trimmed lift are then governed by the position of the aerodynamic center and the horizontal-tail setting referred to the oblique axis of the wing. Some years later, the present author demonstrated the rather surprising stability of the slanted wing by flying models at the first ICAS meeting in Madrid.

Although satisfactory stability can probably be achieved with the yawed wing in the normal flight range, some unusual effects will certainly be apparent. One effect that can be anticipated is a coupling between sideslip angle and vertical acceleration, that is, $\partial L/\partial\psi$. A simple estimate for a wing at 45° yaw shows

$$(1/L)(\partial L/\partial\psi) = 1$$

that is, 1g per radian of sideslip angle ψ. This value may be compared with the sensitivity of vertical acceleration to angle-of-attack changes. Assuming $\partial C_L/\partial\alpha = 5$ and a flight lift coefficient of 1.0 yields

$$(1/L)(\partial L/\partial\alpha) = 5$$

Hence, the sensitivity to yaw is about one-fifth the sensitivity to pitch.

Dynamic coupling between different degrees of freedom is not always undesirable since excessive damping in one mode may be distributed to a mode that would otherwise be deficient. Of course, conventional treatments of stability, which assume bilateral symmetry with the resulting division into "longitudinal" and "lateral" motions, are inapplicable in this case, and a full treatment involving six degrees of freedom as well as aeroelastic deformations will be required.

As is well known, slanted or swept wings tend to stall first at the downstream tips. With the swept wing the loss of lift at the tips leads to a nose-up tendency and aggravates the stall. In the case of the slanted wing, the situation would seem worse since the asymmetric stall would lead to bank. The special measures used to control the pitch-up tendency of swept wings may not be adequate for a yawed wing of high aspect ratio. At best, it is difficult to envision regular landings with the wing in the oblique position, and it seems desirable to incorporate variable geometry so that the wing may be straightened out for landing.

APPLICATION TO TRANSPORT AIRCRAFT

Varying the angle of sweep or yaw has, of course, marked advantages for other flight conditions, such as "holding" at subsonic speeds or adapting the airplane to cruise efficiently at different Mach numbers. Thus, overland flights of a supersonic aircraft will probably be limited to Mach numbers low enough to avoid the sonic boom. The same aircraft may fly much faster over water.

G. H. Lee (refs. 9 and 12) has suggested that the mechanical problems of variable geometry would be effectively eliminated by making an "all-wing" aircraft. Such a step is certainly attractive, but with current densities and loadings seems to be possible only for airplanes of very great size. Lee assumes an airplane designed to cruise at $M = 2.0$ and compares the yawed-wing aircraft with the more conventional delta-wing type.

The yawed wing was found to be capable of carrying twice the payload on the Atlantic flight chiefly because of its better off-design performance. It is anticipated that current supersonic transports may consume 30 to 40 percent of their fuel load in subsonic maneuvers. The ability to cruise or hold efficiently at reduced speed would thus be important for the utility of such aircraft. Current delta-wing transports require large amounts of power for takeoff primarily because of large values of weight per unit span. Since the takeoff power diminishes approximately as the 3/2 power of the span loading, extending the wing span can be very effective in reducing takeoff distance and noise.

An arrangement permitting more conventional values of wing loading and size is shown in figure 13. The use of two bodies connected across by the wing and horizontal tail in a parallelogram arrangement has certain advantages over the usual arrangement for variable sweep. Shearing the parallelogram does not displace its center of gravity and only slightly displaces the center of lift. The first-order variation of lateral trim with yaw angle can be compensated by constructing the wing with an appropriate curvilinear dihedral, as indicated earlier.

As is well known, varying the geometry of the swept wing has several drawbacks. First of all, massive bearings which can carry the wing root bending moment must be used. Secondly, movement of the lifting surface backward at supersonic speeds compounds the normal rearward travel of lifting pressure at these speeds. With the single slanted wing, however, the wing beam structure is continuous across the pivots and no primary bending loads appear on the pivots.

Figure 14 shows a simpler version in which the wing and horizontal tail are pivoted to a single fuselage. Intuitively one would expect the long forward-going tip of the oblique wing to show a tendency for aeroelastic divergence as in the case of the bilaterally symmetric swept-forward wing. Both the oblique wing and the swept wing become concave upward when deflected by an upward gust and cause an increase in the load on the forward tip. In the case of the oblique wing, however, the loading produced by the upward curvature of the wing is aerodynamically equivalent to a twist and is antisymmetric so that the wing responds by rolling. As noted in the free-flight tunnel tests (ref. 11) the wing response is such that the helix angle of the roll ($pY/2V$) cancels the loading due to the twist. Evidently, the relief provided by rolling motion reduces the tendency for aeroelastic divergence in the case of the oblique wing.

CONCLUDING REMARKS

It is admittedly surprising that aerodynamics and simple mechanics would lead to an antisymmetric form for supersonic flight. The difficulties with such forms may, however, be more conceptual than real and it is hoped that this analysis, although incomplete, shows that such configurations deserve more serious study.

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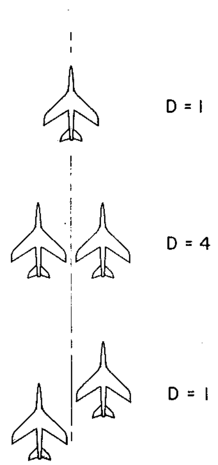


Figure 1.- Wave drag at Mach numbers near 1.0.

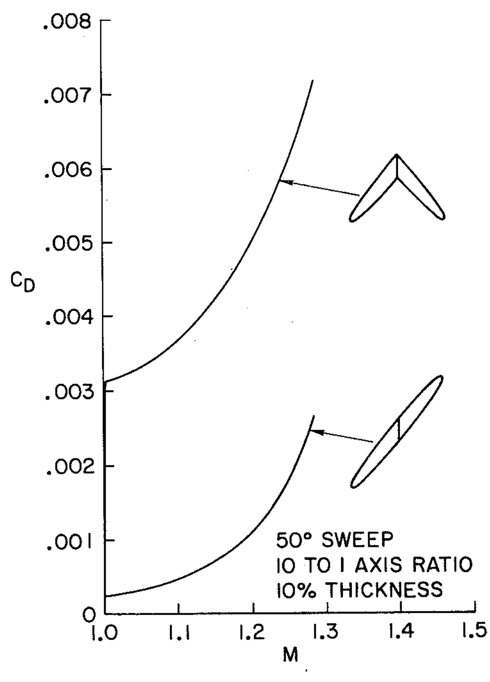


Figure 2.- Calculated wave drag of symmetric and antisymmetric wings.

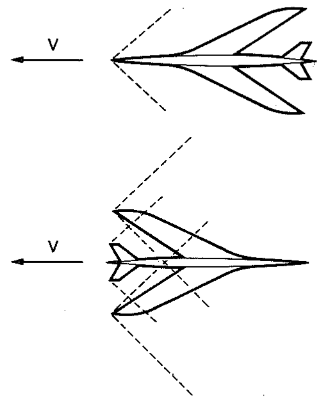


Figure 3.- Reversibility of drag at low supersonic speeds.

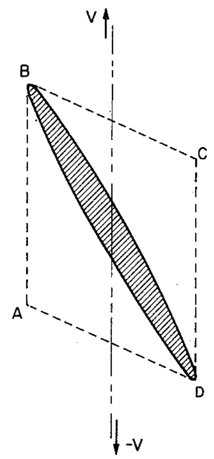


Figure 4.- Optimum distribution of lift and volume within area ABCD.

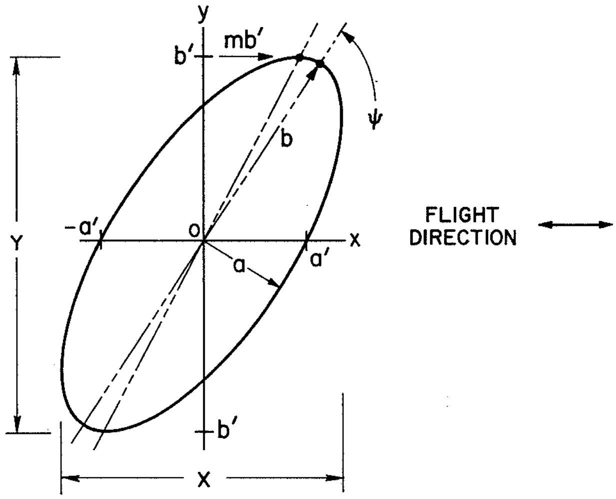


Figure 5.- Oblique ellipse notation.

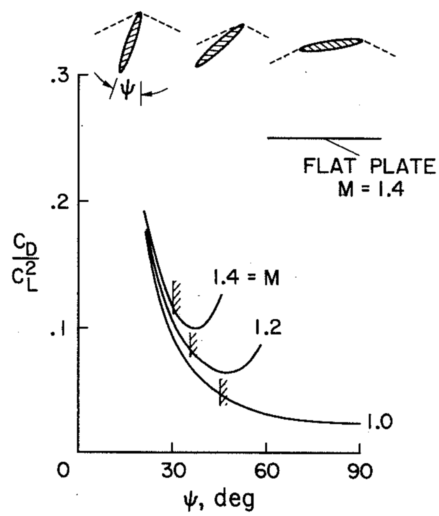


Figure 6.- Drag due to lift. b/a = 10.

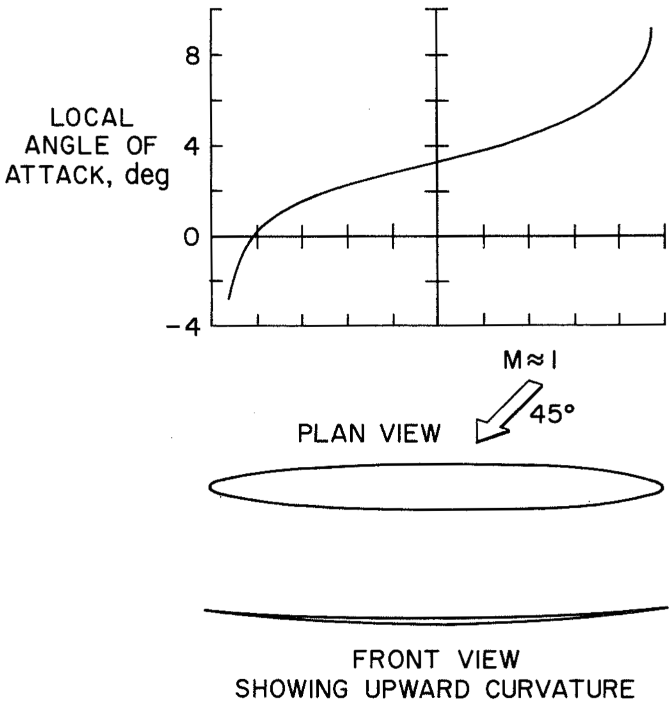


Figure 7.- Upward curvature of wing to maintain uniform lifting pressure. $C_L = 0.5$.

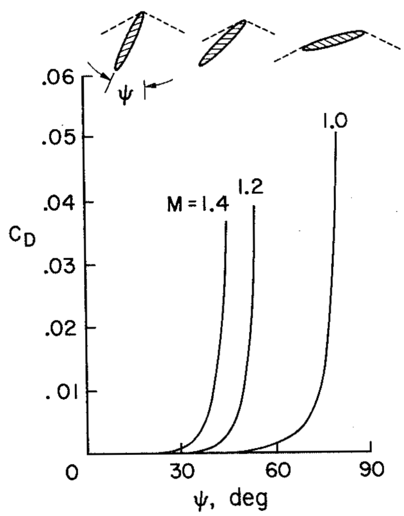


Figure 8.- Drag due to volume. Oblique elliptic wings; $t/2a = 0.1$; $b/a = 10$.

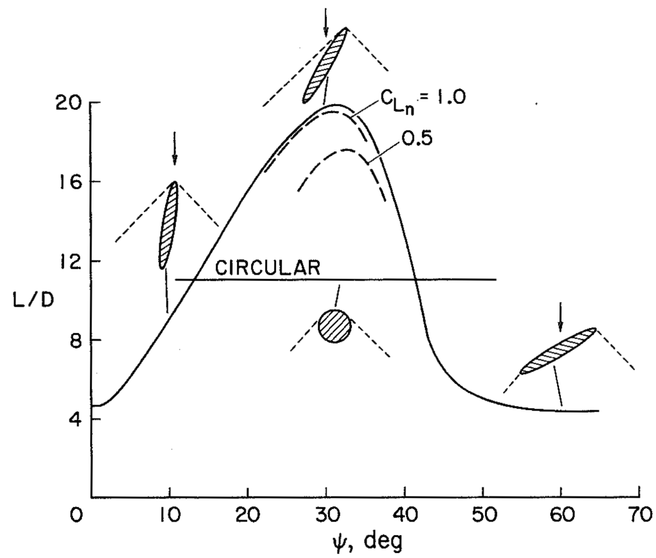


Figure 9.- Estimated L/D ratios. $M = 1.4$; $b/a = 10(t/2a) = 1/10$. (Taken from ref. 8.)

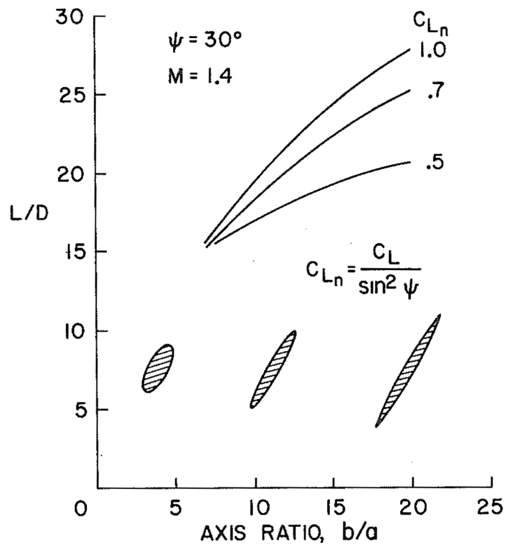


Figure 10.- Variation of L/D with axis ratio showing effect of limitation of the normal lift coefficient C_{L_n}.

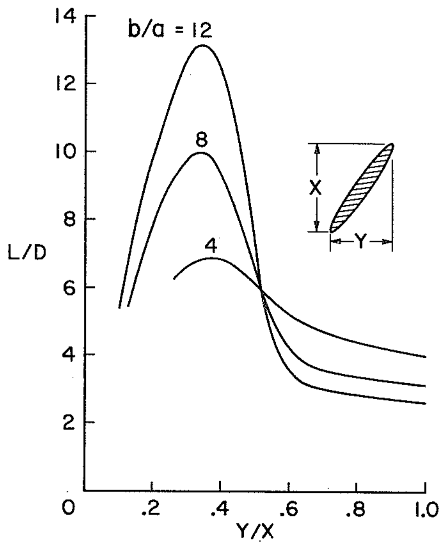


Figure 11.- Estimated L/D ratios for M = 2.0. (See ref. 7.)

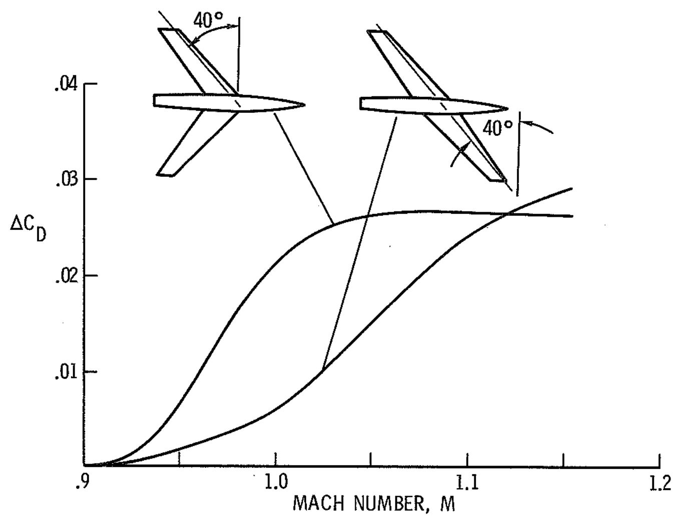
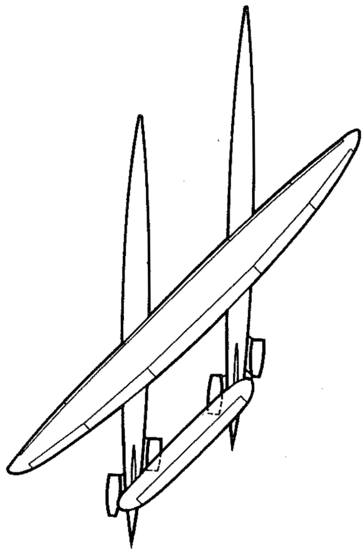
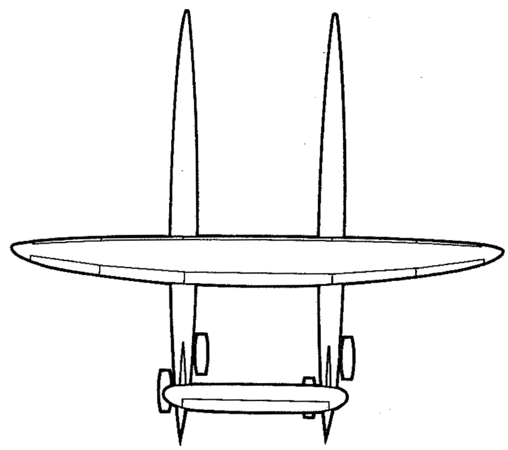


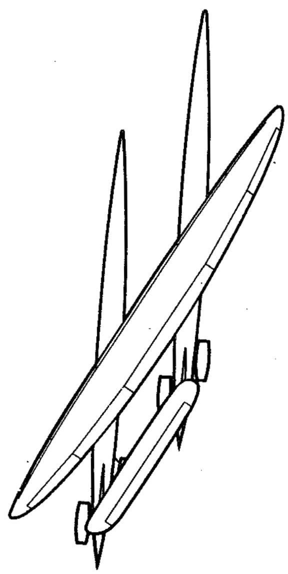
Figure 12.- Drag at zero lift for comparison of yawed and swept wings. (See ref. 7.)



(a) Trimmed for $M \approx 1.0$.



(b) Trimmed for $M < 0.7$.



(c) Trimmed for $M = 1.3$ to 1.4.

Figure 13.- Antisymmetric transport aircraft.

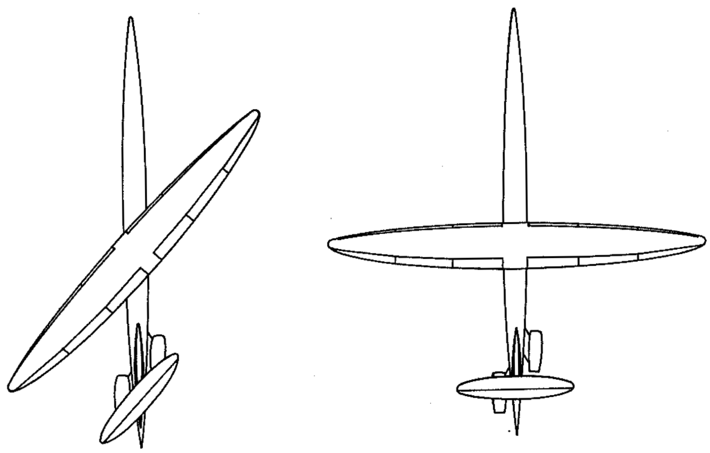


Figure 14.- Oblique-winged transport with single fuselage.

THE SECOND-GENERATION SUPERSONIC TRANSPORT

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SUMMARY

A study has been conducted to examine the impact on the second-generation supersonic transport of technology advances forecast for the 1975 to 1985 time period. A number of areas have been identified which offer the potential of major improvements in capabilities compared with the characteristics of the first-generation supersonic transports. When considered together, it is evident that the technical prognosis is good for the development of a well balanced second-generation aircraft that should be able to more than hold its own in the intercontinental air-transportation market of the 1980's.

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INTRODUCTION

There are three major classes of supersonic transports under consideration as potential contenders for a share of the long haul air-transportation market of the 1980's. As illustrated in figure 1 they are the medium-range SST, of which the Anglo-French Concorde and the Soviet Tupolev TU-144 are initial examples; the low-supersonic or "no boom" SST, one version of which is discussed in paper no. 23 by Robert T. Jones; and the so-called second-generation SST. The second-generation SST is loosely defined as an advanced follow-on to the Concorde and the TU-144 SST's which surpasses them by substantial margins in the areas of performance, economics, safety, and social acceptability. At the present time, most interest in this country is focused on this third class of SST. The purpose of this paper is to predict what can be done with this aircraft through incorporation of the technology forecast for the 1975 to 1985 time period.

SYMBOLS AND ABBREVIATIONS

C_L	lift coefficient
C_m	pitching-moment coefficient
c	local chord

L/D	lift-drag ratio
$(L/D)_{CRUISE}$	cruise lift-drag ratio
$(L/D)_{max}$	maximum lift-drag ratio
M	Mach number
R	Reynolds number
T_{STAG}	stagnation temperature
W_{EST}/W_{GE4}	ratio of estimated weight to weight of reference engine
W_{TO}	take-off weight
$W_{RESERVES}$	reserve fuel weight
α	angle of attack

A/B	afterburner
EPNdB	effective perceived noise in decibels
OEW	operating empty weight
PNdB	perceived noise in decibels
SAS	stability augmentation system
SFC	specific fuel consumption
SFC_{CRUISE}	cruise specific fuel consumption
SLS	sea level static
SST	supersonic transport

DISCUSSION

The major shortcomings of the present-generation SST's are as follows:

(1) Marginal range/payload performance

(2) Complexity and cost

(3) Environmental impact
    Atmospheric pollution
    Sonic boom
    Airport and community noise

It will be noted that lack of speed is not identified as a shortcoming. There are several reasons for its omission from the list. As shown in figure 2, the flight efficiency parameter $\frac{M}{SFC} \cdot \frac{L}{D}$ calculated from the most recent airframe-efficiency and turbojet-performance data (paper no. 4 by Harry W. Carlson and ref. 1) appears to peak at a Mach number slightly lower than 3.0 with the use of JP fuel. Beyond this peak, the stagnation temperature increases rapidly leading to rapid intensification of aircraft systems problems and to some very difficult materials problems, particularly in the nonmetallics area (i.e., transparencies, tires, sealants, lubricants, etc.) At the same time, the gain in block time associated with further increases in cruise speed becomes rather small even for mission ranges up to 4500 nautical miles. For this reason there does not appear to be much incentive for pushing beyond the $M = 2.7$ cruise speed selected for the Boeing 2707-300 SST unless very large increments in both range and cruise speed are contemplated at the same time. A Mach number of 2.7 has been selected arbitrarily as the basic cruise speed to be considered in the present study.

Environmental Considerations

Recent history has underscored the fact that the possible impact of the SST on the environment must be taken fully as seriously as its other problems. Its contribution to air pollution in the vicinity of the airport is expected to be about the same as that for other jet transports, and a satisfactory answer has not yet been obtained to the question of whether a fleet of SST's will cause any long range adverse changes to the stratosphere. These matters obviously will require intensive attention.

The situation with regard to the second environmental impact item, sonic boom, is portrayed in figure 3. Presented therein are values of sonic-boom overpressure calculated along the ground track of a well-designed intercontinental SST cruising at an altitude of 18 kilometers (60 000 feet) at $M = 2.7$. The basic machine considered is 91 meters (300 feet) long and has a cruise weight of 295 000 kilograms (650 000 pounds). With the design optimized from the viewpoint of performance capability only, the sonic-boom overpressure is predicted to be about 110 newtons/meter^2 (2.3 pounds/foot^2) at its basic weight. It is not reduced much below 96 newtons/meter^2 (2.0 pounds/foot^2) even if ways are found to reduce the cruise weight by as much as 30 percent. The lower band shows the level of sonic-boom overpressure indicated by theory to be at least potentially obtainable with detailed attention paid to boom minimizing shaping of the configuration. It appears that, even if the required shaping can be incorporated without destroying the airplane performance capability, the sonic-boom overpressure cannot be reduced below values in the neighborhood of 48 newtons/meter^2 (1 pound/foot^2). The conclusion is obvious. Unless some unforeseen breakthrough is achieved, the $M = 2$ to 3 second-generation SST must be designed as an essentially overwater aircraft. If this premise is accepted, airport and community noise becomes the dominant concern of the airframe designer in the environmental area.

Noise requirements for the SST have not yet been fully and finally resolved. Federal Air Regulation 36 (ref. 2) basically specifies a noise limit of 108 EPNdB at reference points beneath the flight path during landing and take-off and along the sideline after lift-off for future large subsonic jets. The locations of the specified points at which noise levels must be met are shown in figure 4. The designer is permitted to trade overages up to 2 EPNdB at one or more measuring stations for compensating reductions at the remaining station or stations if the total exchange does not amount to over 3 EPNdB.

The contour shown in the foreground of figure 4 is a 108 EPNdB contour calculated by the proposed SAE methods (refs. 3 and 4) for a typical intercontinental SST. It passes through the take-off reference point (primarily because the flight path is optimized and the engine is throttled back to meet this difficult requirement), bulges outside the sideline reference points, and falls inside the approach reference point. In other words, the highest noise level occurs along the sideline after lift-off. Study of a number of cases indicates that this situation is general for all SST's which must use noise-reduction operating techniques in order to meet the noise requirement at the reference point beneath the take-off flight path. Thus, the 108 EPNdB sideline noise requirement becomes a primary design point for the airplane.

The Range/Payload Problem

Airline economists appear to be in general agreement that payload fractions higher than those projected for the first-generation SST's are mandatory if a viable SST with satisfactory economic characteristics is to be attained. A glance at a world map (fig. 5) shows also that there are a number of potentially attractive SST routes that require range capability considerably in excess of the 3180 nautical miles from New York to Paris – the minimum objective of the first-generation aircraft. The choice of an optimum range capability is a very involved matter. Obviously, a machine optimized for very long routes, such as those identified by dashed lines, would be overspecialized. On the other hand, there are enough potentially heavily traveled routes longer than 4000 nautical miles to justify a range capability approaching 5000 nautical miles. This range is regarded as a reasonable objective.

The weight history of a current technology SST on a New York to Paris mission (fig. 6) can be used to discuss some of the fundamentals of the range/payload problem. At any point along the route, the total weight is made up of the operating empty weight (structures, engines, systems, furnishings, crew, etc.), the payload, and the fuel. The payload is only $6\frac{1}{2}$ percent of the initial weight; the fuel reserve, which must be carried all the way, weighs roughly $1\frac{1}{2}$ times as much as the payload. Both situations place severe limitations on the economic potential of the airplane. Obvious ways to improve the payload and range are to reduce the operating empty weight, to increase the take-off lift capability (to permit more fuel and payload), to improve the operating efficiency during the various legs of the mission, and to reduce the reserve fuel.

Sources of SST range/payload improvement considered in the present study are as follows:

(1) Upgraded arrow wing and SAS technologies (SAS used for reduction of static margin, loads alleviation, and improved ride and handling qualities)

(2) Advanced engine and noise-suppressor technologies

(3) New materials and new fabrication methods

(4) Relaxation of fuel reserve requirements (as permitted by advances in systems and operating practices)

The next several sections of the paper are devoted to brief reviews of the situation in each area. It will be noted that two frequently discussed means of SST range/payload improvement – (1) increased control of land use in the vicinity of the airport to ease noise restrictions and (2) use of new fuels such as cryogenic methane and cryogenic hydrogen – are not listed. Both innovations obviously have significant potential; however, they both also introduce new classes of problems outside the general scope of this paper. For this reason, they are not considered herein. A discussion of the use of cryogenic fuels in SST's faster than those considered herein and in hypersonic transports is presented in paper no. 25 by John V. Becker and Frank S. Kirkham.

Arrow wing and stability augmentation system technologies.- Research conducted on the arrow wing during the last several years has uncovered a major potential for the improvement of airframe efficiency. Figure 7 presents a plot of estimated maximum lift-drag ratios against Mach number for various SST concepts, established for the most part from model tests. The lower band depicts the performance of the near-delta class of designs of which the Concorde, the TU-144, and the Boeing 2707-300 are representative. At Mach numbers of 2 and above, this class of airframe generally is limited to an L/D in the 7 to 8 range. A properly designed arrow-wing SST, in contrast, appears to be capable of an L/D in the 9 to 10 range with Harry W. Carlson in paper no. 4 estimating future values as high as $10\frac{1}{2}$ for very large machines incorporating wing-body blending. These are important gains.

At subsonic speeds, the variable-sweep version of the arrow-wing concept potentially affords the highest levels of L/D. For this reason it continues to be of interest for SST and military missions involving extensive subsonic as well as supersonic flight despite obvious variable-geometry weight penalties and the fact that the range of wing loadings at which its design optimizes leads to supersonic L/D's no greater than those for the near-delta SST's.

The fixed-arrow-wing SST concept appears to afford subsonic levels of L/D approximately the same as those for the near-delta concepts. This result comes about, despite adverse differences in aspect ratio, because the fixed-arrow-wing SST typically optimizes at a lower wing loading than the near-delta SST's so that the spans of the two types end up very nearly equal. Thus, the fixed-arrow-wing SST concept compared with the near-delta SST concepts offers significantly improved flight efficiency at supersonic speeds without counterbalancing flight efficiency penalties at subsonic speeds.

Although the high potential of the arrow wing with regard to flight efficiency has been recognized for some time, it was not selected for the present-generation SST designs because of anticipated very poor longitudinal stability characteristics in low-speed flight. Early small-scale data (solid-line curve of fig. 8) showed a severe pitch-up at angles of attack lower than that required for take-off and a second trim point at high angles of attack, which is of concern with regard to deep stall.

A great deal of wind-tunnel research has now been completed on the arrow-wing SST configuration. As shown by a comparison of the long-dash and solid-line curves of figure 8, dramatic improvements in the pitching-moment characteristics have been achieved by increases in leading-edge radius coupled with the application and tailoring of leading-edge flaps. Further, it has been found that the effect of increasing the Reynolds number beyond the small-scale values of the test data shown is to further straighten the pitching-moment curve. In consequence, it is now predicted that acceptable pitching-moment characteristics will be obtainable in the full-scale case with leading-edge rounding only or leading-edge rounding combined with a much smaller degree of leading-edge flapping than that required at small scale. The reduction or elimination of the need for leading-edge flaps would of course result in significant reductions in weight, complexity, and cost of the airframe structure.

Even if completely linear stability characteristics are finally obtained for the full-scale machine through aerodynamic refinement, it is anticipated that advanced fly-by-wire stability-augmentation-system technology will be incorporated in the second-generation

SST. In addition to improvements in ride and handling qualities, which is the major use at present, it is anticipated that such SAS will be utilized to reduce the trim drag by permitting a large reduction in the static margin and to reduce the required structural weight through loads alleviation.

Another low-speed problem of the arrow-wing SST has been that of obtaining adequate lift at acceptable angles of attack during take-off and landing. Considerable wind-tunnel research effort has been focused on this problem. As shown in figure 9, the wind-tunnel model just discussed required a wing angle of attack of $9\frac{1}{2}^{\circ}$ to attain a lift coefficient of 0.55 which was desired to meet the objective take-off speed. Breakdown tests showed that the inboard segments of the trailing-edge flaps were by far the most effective. Based on this finding, as shown by the top sketch, the engine nacelles were combined and moved somewhat outboard to provide a larger and higher-aspect-ratio inboard flap. The net result was a reduction in the required angle of attack to $4\frac{1}{2}^{\circ}$. Such an improvement would reduce the length of the main gear of the full-scale airplane by over 1.2 meters (4 feet) and simplify the variable-geometry fuselage nose required for vision during take-off and landing. Both changes again provide significant saving in structural weight and reduce the complexity of the design.

Relocation of two of the engines to the positions indicated by the short-dash lines was found to provide an additional lift increment of about 10 percent with the jet flow of the forward nacelles simulated. Presumably the increase was due to the flow from the forward nacelles inducing additional circulation around the wing. Although it is not known whether such an engine position is acceptable because of obvious acoustic effects on the airframe and passengers, the lift increment is large enough to be of significant interest and will be examined further in future investigations.

Engine and noise suppressor technologies.- It was pointed out previously that the problem of limiting the sideline noise generated by the SST after lift-off to 108 EPNdB (or 110 EPNdB maximum if a total of 2 EPNdB can be saved at the take-off and landing reference stations) has become a primary design point for the airplane. If it is assumed that the nacelle inlets can be operated in a choked condition, the primary source of noise encountered is jet noise. There are two ways of reducing such noise. One is to increase the basic rate of airflow at which the engine delivers the required thrust. The other is to develop and apply a jet noise suppressor. Both approaches are compatible and it is expected that both will be needed.

The jet noise is related simply to the jet velocity and to the mass-flow rate of the engine airflow as is the engine net thrust. Hence, if the point after lift-off which is critical from the noise viewpoint can be identified, and if the SAE method of references 3 and 4 is assumed adequate for predicting the engine noise output, it is possible to calculate the approximate engine airflow necessary to satisfy the sideline noise requirement.

Figure 10 presents results of such calculations for dry turbojets which meet the 108 EPNdB sideline noise requirement for a family of 4-engine SST's on a 10° C hot day at a noise point assumed to occur just after lift-off at $M = 0.3$ at an altitude of 152 meters (500 feet). The abscissa in the plot is the net thrust per engine required at the specified operating condition, and the parametric variable is the degree of suppression afforded by the jet noise suppressor if utilized (a 1-percent loss in net thrust per PNdB suppression is assumed in the calculation). Points defining the approximate airflows and thrust requirements of the Concorde and the Boeing 2707-300 SST's are plotted in the figure for reference purposes although these two SST's do not meet the 108 EPNdB sideline noise requirement.

Several major points can be made from figure 10. First, with no jet noise suppression, much larger engines will be necessary to meet the sideline noise requirement than those for the first-generation aircraft. Second, as shown by the upward curvature of the bands, the problem becomes more difficult as airplane size increases. Third, there is a great premium attached to achieving even a small degree of noise suppression. For an SST with take-off thrust requirement of 267 kilonewtons (60 000 pounds) per engine, the achievement of only 5 PNdB jet noise suppression would reduce the engine airflow by about 18 percent. The effects of such reductions on engine weight and airplane design problems are enormous. It is obvious that a major increase in jet-noise-suppression research in continuation of that described in references 5 and 6 is urgently needed to pave the way for the second-generation SST. Such a program will be particularly pertinent if the SST along with other aircraft is expected to meet the CARD study objective (refs. 7 and 8) of a reduction in airport and community noise of 10 EPNdB per decade.

The width of the data bands in figure 10 accounts for the secondary effects of engine cycle variables – for example, compressor or fan pressure ratio and turbine inlet temperature – within the range of feasible cycles. Actually for each value of thrust along the abscissa, there are a number of engine cycles that meet the noise requirement at values of airflow which fall within the width of the band. Thus, in selecting an optimum engine cycle, it is necessary to examine the effect of the cycle parameters on engine performance at other critical flight conditions. The dry turbojet has been examined in the present study by use of the engine performance program described in reference 1 with essentially the same component performance assumptions specified therein.

Figure 11 presents a projection of the $M = 2.7$ cruise performance at an altitude of 18 kilometers (60 000 feet) of a family of dry turbojets sized to meet the take-off requirements of an advanced 4-engine 363 000-kilogram (800 000-pound) SST including the sideline noise limit of 108 EPNdB. Each engine is assumed to be equipped with a 5 PNdB noise suppressor and is sized to provide 267 kilonewtons (60 000 pounds) net thrust at the M = 0.3, 152-meter-altitude (500 feet) climbout point on a 10° C hot day. For the engines of greatest interest, the inset table identifies the sea level static compression ratios, the maximum continuous turbine inlet temperatures, and estimated bare engine weights relative to the guarantee weight of the GE 4 engine of the Boeing SST. (The engine weight ratios shown apply to the 1980 time period and were predicted by the procedure published in reference 9.) The rectangular stippled area provides a framework of reference by indicating the approximate performance characteristics of the partial afterburning GE 4 engine in the Boeing SST airplane for the same operating condition.

Inasmuch as the advanced 363 000-kilogram (800 000-pound) SST being considered would require about the same level of cruise thrust as the Boeing 2707-300, it is probable that the most nearly optimum engine in the family identified would be the one with a SLS compression ratio of 15 and a turbine inlet temperature of 1200° C (2190° F). This engine as projected would have a bare weight approximately the same, have approximately 30 percent more airflow, and have a cruise specific fuel consumption about 15 percent less than the partial afterburning GE 4 engine with which it is compared. The weight of its nacelle would, of course, be greater. The reduction in specific fuel consumption indicated represents a potential gain in range/payload performance similar to that obtainable by changing from the near-delta to an optimized arrow-wing airframe configuration. This is an important gain.

The new engine discussed in the preceding paragraph incorporates advanced technology in the engine weight and component efficiency areas. The turbine inlet temperature selected because of the take-off noise constraint, however, is well below values that will be feasible by 1980. It is clear that there is still incentive for invention in the engine cycle design area. What is needed is a high-temperature nonaugmented variable-cycle engine having the airflow characteristics at take-off of the turbofan combined with the good cycle efficiency characteristics of the turbojet in supersonic cruise.

New materials and fabrication methods.- The NASA arrow-wing SST concept known as SCAT 15-F has been studied extensively by the industry as a candidate configuration for future high-performance SST's. The results of these studies, which indicate no insurmountable problems, have been utilized to establish the basic weights of the designs considered in this paper. The use of advanced materials and the use of new fabrication methods for accomplishing weight reduction are particularly attractive because the weight ratio term of the range equation is more critical for the SST than for subsonic transports. A wide variety of innovative changes from present practice are possible. For the purpose of this paper, however, attention is restricted to only one example possibility: the use of a moderate amount of composites in the basic structure. As shown in figure 12, which is derived directly from the data presented by Richard A. Pride in paper no. 10, it appears that with about 30 percent composites in the structure, the overall structural weight can be reduced approximately 15 percent. In subsequent performance projections it has been assumed that this amount of weight reduction below the basic structural weights established from the industry studies could be achieved by the use of composites in combination with the loads alleviation SAS mentioned previously.

Relaxation of fuel reserve requirements.- As for the noise requirements, the fuel reserve requirements for the SST have not yet been fully and finally defined. As illustrated in figure 13, the fuel reserve for the so-called "current technology" SST on its assumed New York to Paris mission amounted to roughly $1\frac{1}{2}$ times the payload. These reserves were calculated in accordance with tentative standards for the SST. There is no question but that adequate reserves must be maintained; however, the level shown constitutes a severe penalty.

A major part of the reserve fuel is needed to cope with the situation of an emergency diversion of the airplane to an alternate airport following the landing approach to its original destination. However, there does not appear to be any overriding reason why the airplane, after a missed approach, cannot obtain landing priority at its alternate airport and why it cannot be permitted to execute any hold necessary at that airport at optimum altitude rather than at 457 meters (1500 feet). Both of these measures would result in significant fuel saving. Major additional savings can be obtained, of course, if the airplane can be redirected to an alternate airport prior to the time when it begins its deceleration and descent to its original destination. Such supersonic diversion becomes more practical as air traffic control, navigation, and communications systems are improved and as the basic range capability is increased so that more alternates tend to become available.

It is anticipated that these factors together with airframe and engine changes aimed at improving off-design operating efficiency eventually will permit the use of fuel reserves no larger than today's SST payload fractions. As illustrated in figure 13, a fuel reserve of 23 000 kilograms (50 000 pounds) or $6\frac{1}{4}$ percent of the assumed 363 000 kilograms (800 000 pounds) take-off weight has been assumed in the following projection of the capabilities of the advanced technology second-generation SST.

PROJECTED SECOND-GENERATION SST CAPABILITIES

A sketch of a possible $M = 2.7$ second-generation SST based on the preceding discussion is presented in figure 14 along with some of its major specifications and advanced characteristics. The arrangement shown features twin pod nacelles underslung at about the midsemispan with large high-aspect-ratio flaps located between the nacelles and the fuselage.

As illustrated in figure 15 a number of alternate configurations are possible. The one in the upper left of the figure features over-under engines which increase the extent of the wing trailing edge available for flaps and controls compared with the preceding design, but also introduces some difficult inlet design problems. In the two designs at the bottom, the engines are split into fore and aft pairs. Such arrangements clear up wing trailing-edge clutter, improve the airplane's balance, and offer the possibility of increased take-off weight capability (due to increased circulation lift) at the expense of pod design, pod support, and pod-wing interference problems of as yet undefined difficulty. The configuration at the top right is a modernized version of the 3-engine NASA SCAT 16 configuration of nearly 10 years ago. In this arrangement the pitch-up which plagued the original configuration with a high tail of the type shown is reduced in severity or avoided by a major reduction in the size of the inboard wing strake, with residual pitching-moment nonlinearity problems handled by the assumed advanced SAS system. This configuration would not be expected to be as efficient as the others shown for classical long range supersonic missions but has the potential of good performance in mixed supersonic-subsonic missions and good take-off noise characteristics.

The projected range/payload performance of the second-generation SST of figure 14 is presented in figure 16. With the assumptions made, the 363 000-kilogram (800 000-pound), 370-passenger aircraft is estimated to have a full-payload range capability of 5290 nautical miles as indicated by the solid symbol. The performance assumption of figure 14 most open to question is the ratio of operating empty weight to take-off weight. The sensitivity of the range to this parameter is indicated by the shaded band. If the weight ratio is increased from the assumed value of 0.39 to 0.42, the full-payload range capability is reduced to 4670 nautical miles which still is ample for most anticipated route structures.

Range, payload, and airplane gross weight can be traded against each other to an acceptable degree of accuracy in the analytical design process. With the take-off weight held constant, it is estimated that a 500-passenger airplane can be developed with full-payload range capability of 4000 nautical miles and that a 234-passenger airplane can be developed with a full-payload range capability of 6580 nautical miles. If some of the range of the latter machine is traded for reduced aircraft weight, it appears that full-payload range of 3470 nautical miles (N.Y. to Paris + 300 miles) can be attained at a gross weight of only 242 000 kilograms (533 000 pounds). The three configurations to the left of the figure progressing from the bottom to the top have payload fractions of 9.2 percent, 9.7 percent, and 13.0 percent. These payload fractions are $1\frac{3}{4}$ to $2\frac{1}{2}$ times those projected for the first-generation SST's and, consequently, should provide very good economic potential.

CONCLUDING REMARKS

A number of areas have been identified in which anticipated technology progress offers the potential of major advances in SST performance, economics, and social acceptability compared with the characteristics of first-generation aircraft. When considered together, it is evident that the technical prognosis is good for the development of a well balanced second-generation aircraft that should be able to more than hold its own in the intercontinental air-transportation market of the 1980's.

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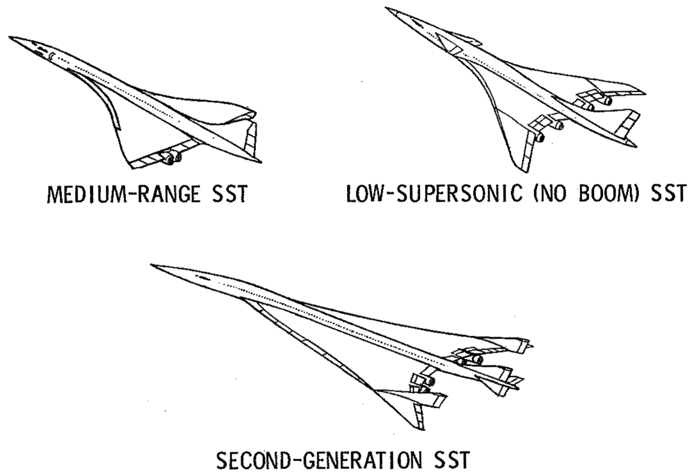


Figure 1.- Classes of supersonic transport aircraft.

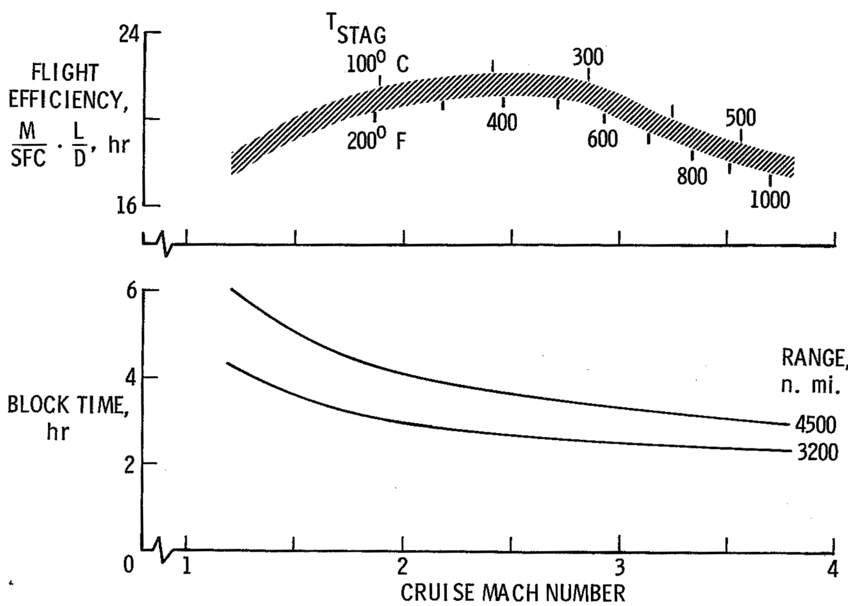


Figure 2.- Factors influencing choice of cruise speed.

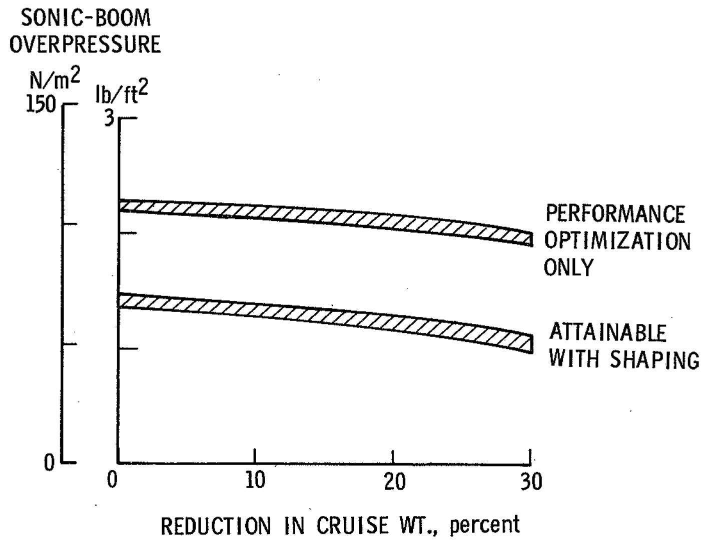


Figure 3.- Cruise sonic boom for an advanced supersonic transport.

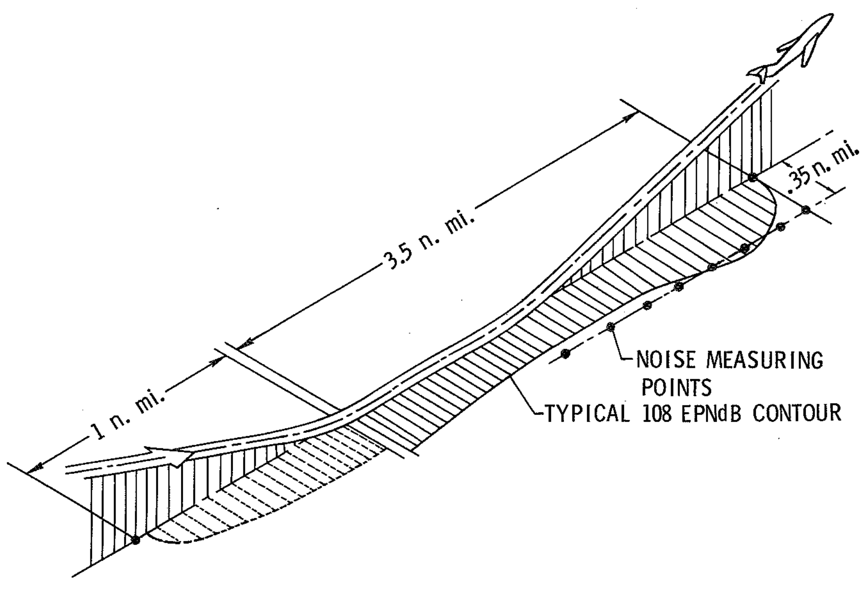


Figure 4.- Airport and community noise.

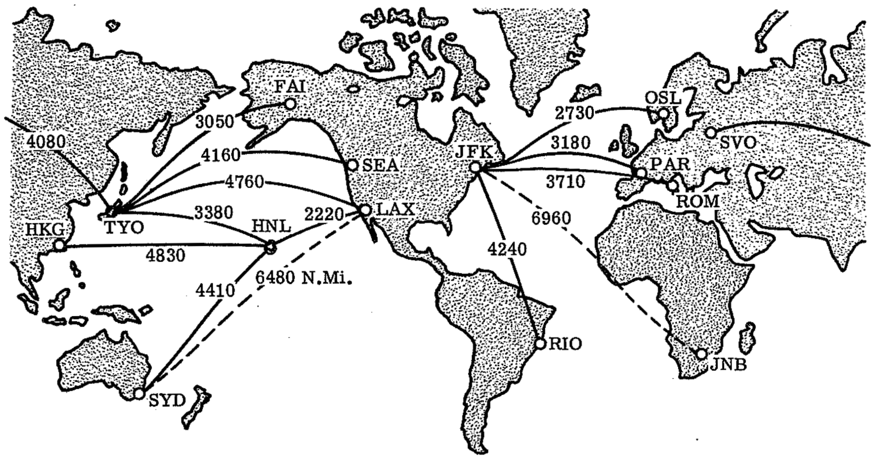


Figure 5.- Range requirements for potential supersonic transport routes.

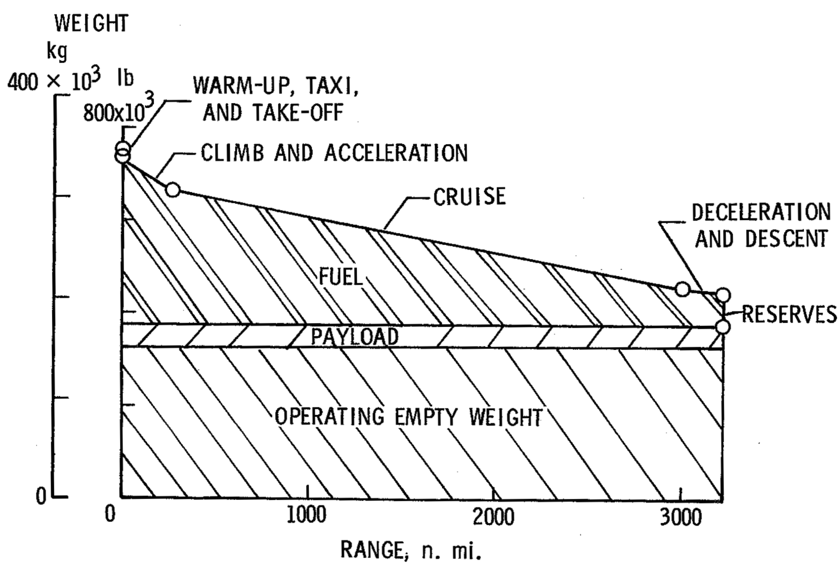


Figure 6.- Weight history of 340 000-kilogram (750 000-pound) current-technology supersonic transport on a New York to Paris mission.

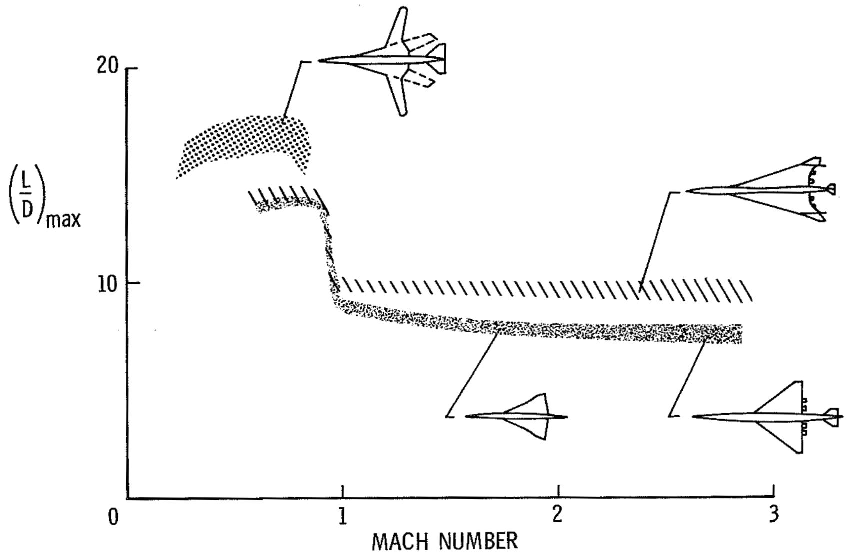


Figure 7.- Improvement of airframe efficiency.

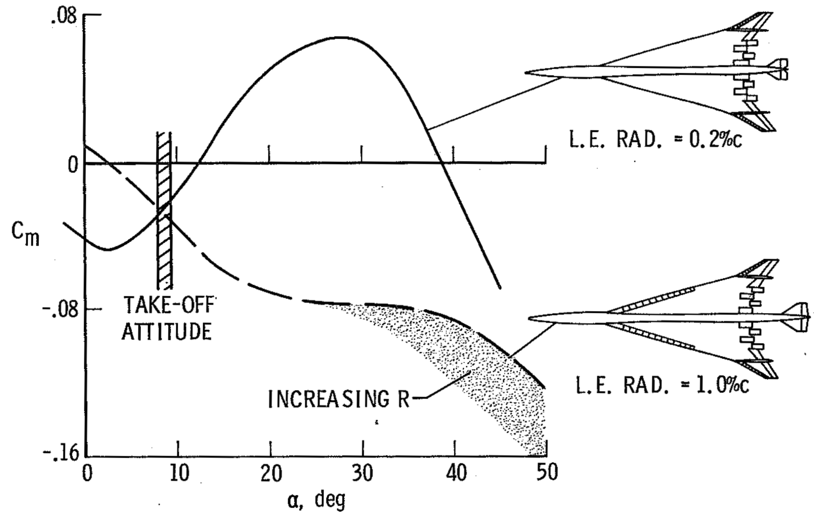


Figure 8.- Improvement in low-speed stability. $R = 3 \times 10^6$.

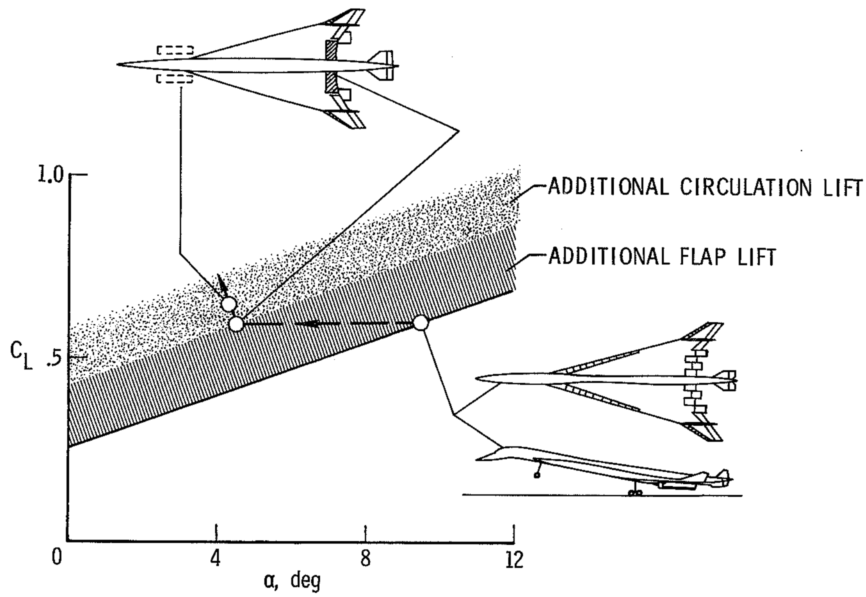


Figure 9.- Improvement of low-speed lift. $R = 3 \times 10^6$.

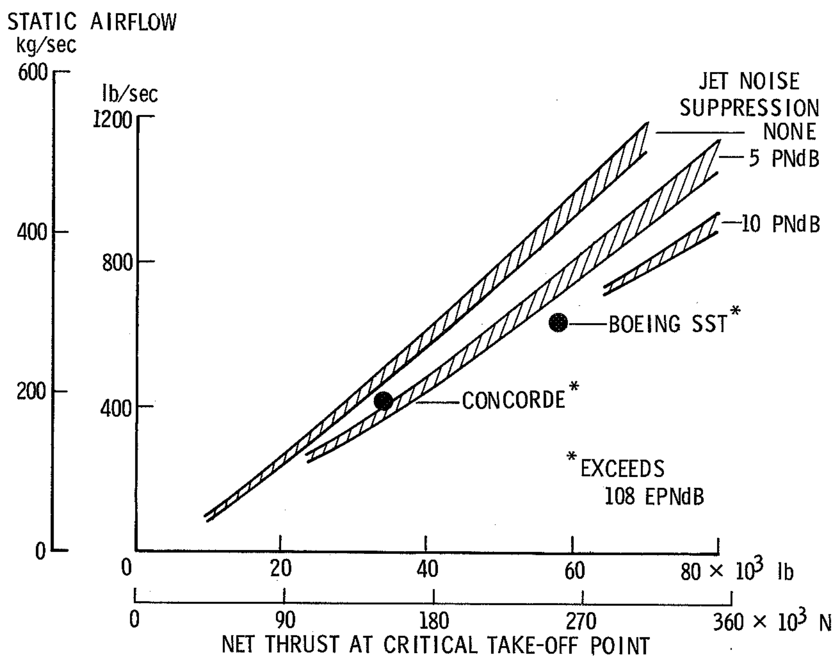


Figure 10.- Minimum dry turbojet airflow for 108 EPNdB sideline noise. $M = 0.3$; altitude, 152 meters (500 feet); 10° C hot day.

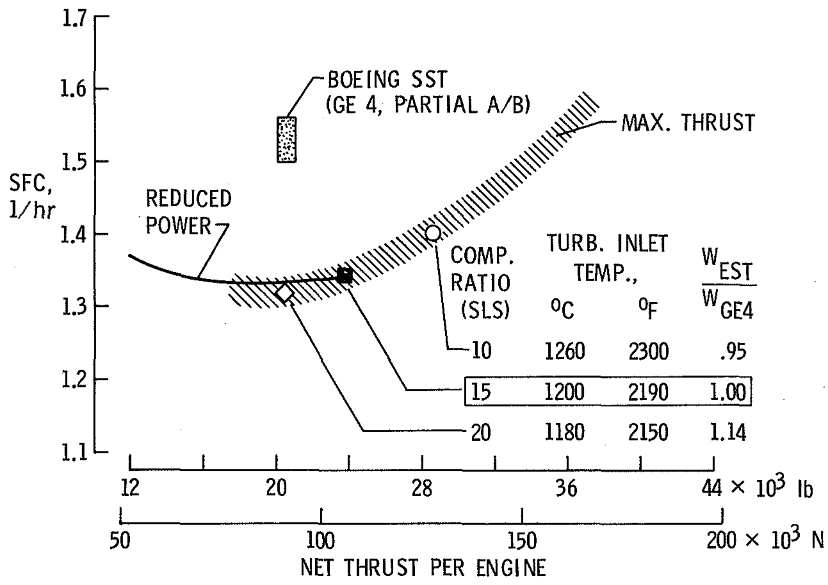


Figure 11.- Cruise performance at $M = 2.7$ and 18-kilometer (60 000-foot) altitude of dry turbojets sized for take-off requirements of advanced 363 000-kilogram (800 000-pound) SST with 5-PNdB noise suppressors.

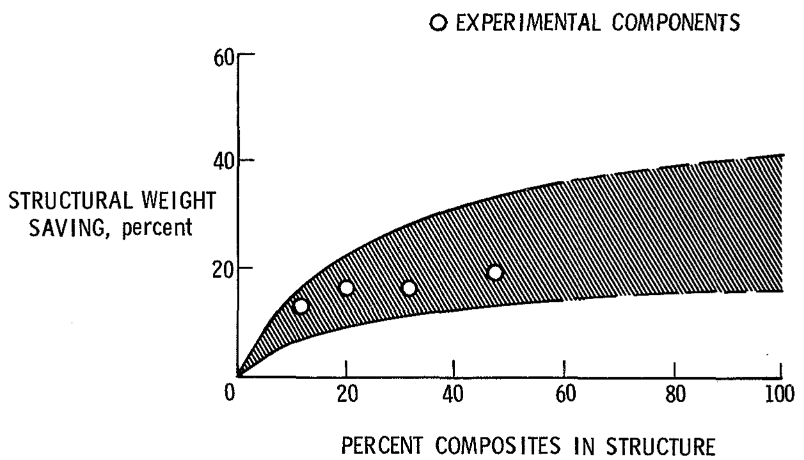


Figure 12.- Potential saving in structural weight as a function of composite material utilization.

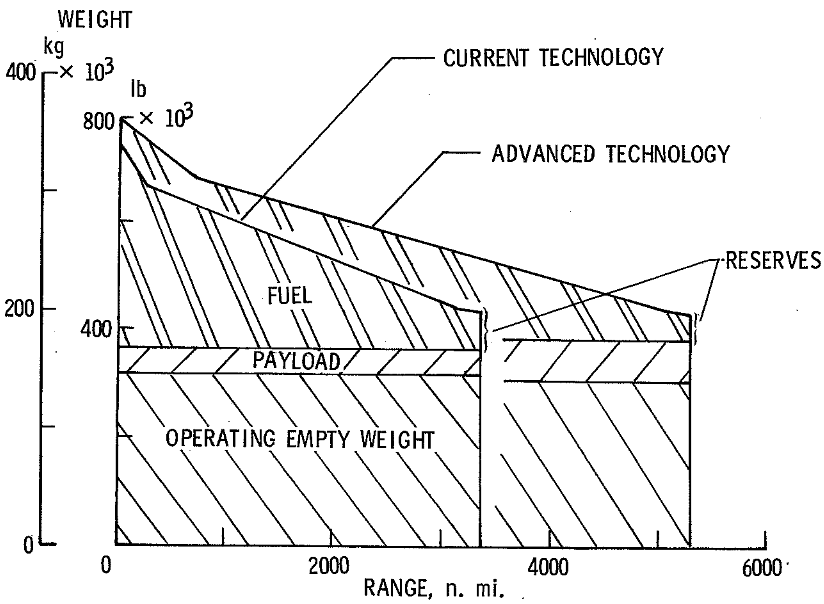
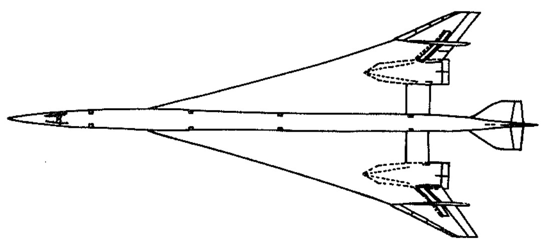


Figure 13.- Supersonic transport weight history.



W_{TO} = 363 000 kg
(800 000 lb)

370 PASSENGERS

4 DRY TURBOJETS

MAX. AIRPORT AND COMMUNITY NOISE, 108 EPNdB

ADVANCED CHARACTERISTICS	
$(L/D)_{CRUISE}$	8.7
SFC_{CRUISE}	1.35 hr^{-1}
OEW	141 000 kg (310 000 lb), 0.39 W_{TO}
$W_{RESERVES}$	23 000 kg (50 000 lb), 0.06 W_{TO}

Figure 14.- Projected $M = 2.7$ second-generation supersonic transport.

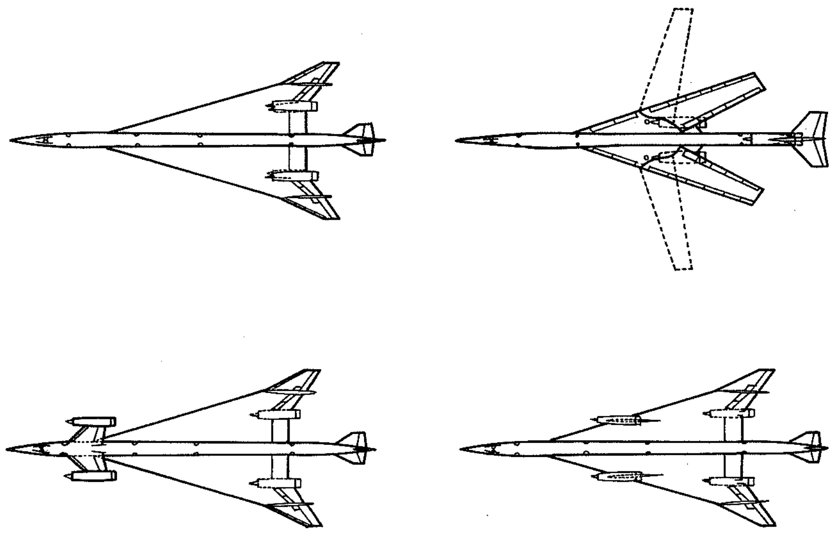


Figure 15.- Alternate second-generation supersonic transport concepts.

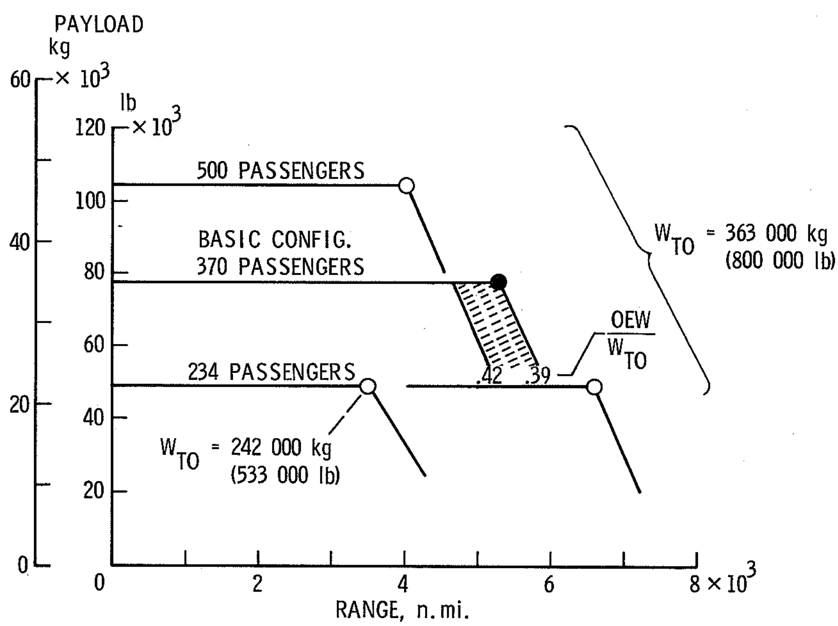


Figure 16.- Projected range/payload for $M = 2.7$ second-generation supersonic transports.

HYPERSONIC TRANSPORTS

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INTRODUCTION

The previous applications papers were concerned with refinements for vehicle types which have already enjoyed massive development. There has been no such large-scale support for hypersonic aircraft, but nevertheless steady progress is being made in the disciplines, as seen from the earlier hypersonics papers in this conference compilation. In the present paper an attempt is made to assess the implications of these disciplinary advances for the hypersonic transport. In projecting some 20 years into the future, it is obviously appropriate that these assessments be uninhibited.

OPERATIONAL CONSIDERATIONS

The anticipated long-range air traffic of 1990 (refs. 1 and 2) is indicated in figure 1. About 90 percent of this traffic involves ranges of less than 6000 nautical miles. Increasing exchange with Russia and China could substantially augment the traffic in the 5000- to 7000-nautical-mile range. Beyond that range there is only a small 9000-nautical-mile requirement and no projected traffic in the semiglobal area.

25

On the lower scales, the nominal capabilities of the candidate long-range systems are compared. The solid lines denote what is possible with today's technology. The original semiglobal system is the rocket-glider, an outgrowth of Sänger's proposal of the thirties. This system is now enjoying a revival because some of its prime technology requirements are being developed in the Space Shuttle program. One rather extreme proposal suggests that the shuttle might be directly adapted to carry some 200 passengers for earth transportation. (The shuttle referred to here is the so-called "fully reusable" or ultimate version of the Space Shuttle.)

Some of the features of such a shuttle-type transportation system are illustrated in figure 2. This system has all the exciting aspects of space flight. Although the sonic-boom levels for the upper stage would be very low for most of this trip, there would be a boom problem during the latter part of the overwater descent. There would also be a boom problem for the returning launch stage. Often advertised as a 1-hour trip, the actual trip time could be more like 2 hours if the low-speed travel to the remote launch pad and final subsonic overland approach phases are included.

The basic problems of such a rocket system are extreme launch noise necessitating very remote launch sites, high acceleration, and high operating costs. Fuel costs constitute most of the 12¢/seat-mile direct operating cost (DOC) shown in figure 3. The true costs would be higher than this value if the actual shuttle lifetime (100 flights) and turnaround time (order of weeks) were used here instead of assumed typical current airline values.

A more likely eventuality is the use of shuttle-derived rocket system technology in more modest and less energetic systems like the horizontal take-off (HTO) rocket shown in figure 3, which is representative of several recent studies that have appeared in the literature. However, all the rocket systems display the characteristics of very great noise and high acceleration (by airline standards).

Plausible schemes for avoiding these basic difficulties suggested in the literature center on elimination of the launch rocket and substitution of some form of airbreathing propulsion in the first-stage vehicle. (See, for example, refs. 3 and 4.) In the scheme depicted on the right-hand side of figure 3, a hypersonic airbreathing launch vehicle was assumed. This scheme would of course involve new technology, beyond the shuttle.

The application of airbreather technology to cruise vehicle systems in the Mach 6 to Mach 8 category (fig. 4) is in many ways more attractive than the mixed airbreather-rocket systems. Operationally the cruise airbreather will fit into present-day airline practice and will have characteristics and problems generally similar to those of the supersonic transports (SST's). Direct operating costs of about 1.5¢/seat-mile are projected for hydrogen fuel costs of 22¢/kg (10¢/lb).

A possibly important advantage of the hypersonic transport, in addition to its high speed, is its relatively low cruise sonic-boom overpressure level (fig. 5), which might permit it to fly overland. There would be tremendous economic advantages and other traveler advantages to such overland flights, and eventually it is hoped that these advantages will be logically evaluated and weighed against the environmental implications of the reduced booms with the result that such flights will prove permissible. In assessing the consequences of these booms, their total impulse and signature shapes as well as the peak overpressures will have to be considered.

Figure 5 also shows that there is little trip-time saving if the cruise Mach number increases beyond about 8, even for the 4860-nautical-mile Los Angeles to Paris trip. Interestingly, the Los Angeles to Paris trip time at Mach 8 is only about one-half hour longer than the rocket-glider trip depicted in figure 2.

By fortunate happenstance, hydrogen-fueled cruise vehicles also tend to optimize in the Mach 6 to Mach 8 region from the standpoint of weight (fig. 6). If current advanced-state-of-the-art "hot hypersonic structure" weights (ref. 5) and nominal current estimates of aerodynamic and propulsion parameters are used, a 6000-nautical-mile range is achievable with a take-off gross weight of 340 000 kg (750 000 pounds), which is about the same as the gross weight of the current nominal 3500-nautical-mile Mach 2.7 JP-fueled SST design.

The Mach 3 and Mach 4 vehicles, whose range performance is seen in figure 6 to be only slightly inferior to the range performance of the Mach 6 aircraft, can be thought of as state-of-the-art titanium vehicles like the Mach 2.7 SST except for changes incident to the use of hydrogen fuel (and the use of ramjets at Mach 4) which were accounted for in some detail in the present study. Their performance would seem to warrant open-minded consideration of hydrogen fuel for future SST's.

HYDROGEN FUEL

The limited-petroleum-energy outlook for the far future (25 to 50 years hence) coupled with increasingly stringent pollution-control requirements indicates an inevitable change to new fuels for transport vehicles. Hydrogen is the most likely candidate from a performance and pollution viewpoint. It is already being used in space vehicles, experimental automobiles, and elsewhere (ref. 6). There are major problems of production and logistics for hydrogen; however, there are two problems which are more imaginary than real. First is the question of costs. Figure 7 shows the general picture (derived from refs. 7 and 8 and from personal communications with fuels specialists). Large increases in the production of liquid hydrogen (LH_2) coupled with expected increases in JP costs due to diminishing reserves will result in about equal costs by the end of the century. If major environmental and conservational pressures develop, the cost of JP could rise faster than shown.

Despite the exceptional safety record of hydrogen in the space program, hydrogen safety is another question clouded by fears and misunderstanding possibly originating from the widespread impact of the Hindenburg Disaster. Figure 8 summarizes the general view of the specialists: overall comparability with gasoline or methane safety (ref. 9).

STRUCTURES DEVELOPMENTS

The traditional approach to cruise vehicle structures is to utilize superalloy or refractory metals, protected extensively with lightly loaded exterior heat shields (fig. 9). This "hot structure" approach is enjoying at present the prospect that heat-shield technology will benefit from the Space Shuttle program. The other approach, actively cooled structures of conventional materials (including the load-bearing exterior skin), has only recently evolved as a valid possibility (ref. 10). It has its roots in the advanced scramjets which leave a large residual part of the fuel heat sink available for airframe cooling (as described by John R. Henry and H. Lee Beach in paper no. 8 and by Melvin S. Anderson and H. Neale Kelly in paper no. 9). With this approach there is the exciting ultimate potential also of making the boron-aluminum composites usable in the hypersonic environment.

The bulk of heat-shield research and development so far has been concerned with critical heating situations, for the most part with relatively short-lived space reentry systems. The design life goal for the Shuttle heat shields is 100 flights, or only about 1 percent of what is required for a hypersonic transport. The type of design which has evolved (fig. 10) uses thin (0.025 to 0.051 cm (0.010 to 0.020 inch)) shields connected with delicate clips to the underlying structure (ref. 5). Sliding joints, sealed by flexible bellows against water and hot boundary-layer air leakage, are required to allow for the large expansions which occur as the shields attain white heat. It is a moot question whether this complex and delicate surface structure can successfully endure the rigors of airline handling, weather, and repeated exposure to intense heating over many years of airframe life. Maintaining precision airtight seals not only between the sliding joints but in such major problem areas as landing-gear doors is an example problem of great difficulty. The consequences of a boundary-layer leak were well illustrated in the failure of a seal in the nose-gear cover of the X-15 at Mach 5 (fig. 11). Aluminum tubing was melted in this accident.

In the cooled structure, there is the possibility of either eliminating or alleviating all the difficulties of the hot structure, but at the expense of solving a wholly different set of research and development problems. The cooled Mach 6 vehicle shown in figure 12, which is the ultimate goal of the active cooling approach, would differ little in materials of construction or skin thickness from the SST's. In flight, instead of the red-hot/white-hot heat-shielded exterior of the hot structure, one would see a vehicle identical to the SST's in appearance. And presumably the lifetime of the cooled system would be the same as that of the SST's. A rather detailed first exploration of the cooled structure has recently been completed with the aid of several contractors. The panel design shown in figure 13 has been analyzed in some detail, and the coolant-tube spacings and sizes, the temperature gradients, and the coolant flow rates have been found to be within practical limits. A study of the entire system (fig. 14) revealed practical mechanical component dimensions and weights. Although a system covering the entire airframe surface, such as the one studied here, presents many questions, it is well to realize that the so-called hot structural approach must also incorporate active cooling in passenger cabins, in baggage and avionics bays, and in the propulsion ducting. Thus a major commitment to active cooling must be accepted in any case.

The cooling system weights in the present study were found to be more than offset by the reduction in airframe weights made possible by cooling (fig. 15). The coolant (water-glycol) and its piping constitute the principal cooling-system weight. Future studies will consider design criteria and system reliability in detail and determine the optimum panel concepts and secondary coolants for a variety of airframe materials, including composites.

These results for the cooled structure permit realistic speculation on its potential advantages for future vehicles. With cooling, a 15-percent structural weight reduction due to composites and a major reduction in cryogenic-tankage-insulation weight can be postulated. In addition, a modest 10-percent gain in lift-drag ratio (L/D) through refinements such as area ruling, twist, camber, and filleting has been included. A 12-percent increase in specific impulse is also believed to be obtainable from increases in component efficiencies over the nominal values used previously. And thus the 6000-nautical-mile range obtained with the hot vehicle could be extended to over 10 000 nautical miles through future technology, as seen in figure 16. For comparison, the lesser gains estimated on a comparable basis for a JP aircraft are also shown.

Actually, translating these future probabilities into terms of reduced vehicle size (fig. 17) is of more interest. Going back to the 6000-nautical-mile range with 300 passengers as a likely design goal, one sees that this goal could be achieved with a cooled 181 000-kg (400 000-pound) vehicle having a 16-percent payload fraction. Or, looked at another way, these cooled hydrogen vehicles have a tremendous weight margin which can be utilized for take-off-noise suppression, shape changes in the interest of low sonic boom, payload growth, solution of aeroelastic problems, and so forth. Because of the inherent larger structural-weight fraction of hydrogen aircraft, the area for potential improvement is of course many times larger than that shown for the JP aircraft. And thus the overall future prospect combines unique operational advantages associated with very high-speed, high-altitude cruise with the advantages of minimal vehicle weight and high payload fractions. These foreseen advances in hypersonic technology together with the projected reductions in the cost of liquid hydrogen result in direct operating costs (fig. 18) of hypersonic airbreathing transports that are competitive with those of JP-fueled aircraft.

A HYPERSONIC RESEARCH AIRPLANE

Although promising new approaches are being pursued in all the disciplines, there are of course several major deficiencies, chief among which are the lack of a proven power plant and the lack of a proven practical structural concept. Probably the most serious deficiency is the absence of any real-flight-vehicle development. Past experience suggests that progress beyond the present stage will be slow until the development of an actual vehicle is undertaken. In previous situations of this kind where it is obviously too soon for a full-scale prototype, the research airplane has been used to great advantage to provide the necessary focus, stimulus, and resource levels. The X-15 program, for example, provided the first great impetus to hypersonics and manned-space-flight technology.

Figure 19 presents a concept and specifications for a small research airplane which can be thought of as a 1/3-scale version of the hypersonic transport. Airbreathing research scramjet engines and wing panels which could embody a variety of structural concepts are principal features. The vehicle would be capable of about 5 minutes cruise at Mach 8 either on its primary rocket propulsion or with the research scramjets. Present technology would fully support the development of such a vehicle. Both the analytical and the experimental tools are available. No new national facilities would be needed.

The technology base developed with a hypersonic research airplane would make it possible to proceed with confidence to a full-scale prototype hypersonic transport or other applications including airbreathing launch systems.

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PROJECTED 1990 INTERCONTINENTAL AIR TRAFFIC

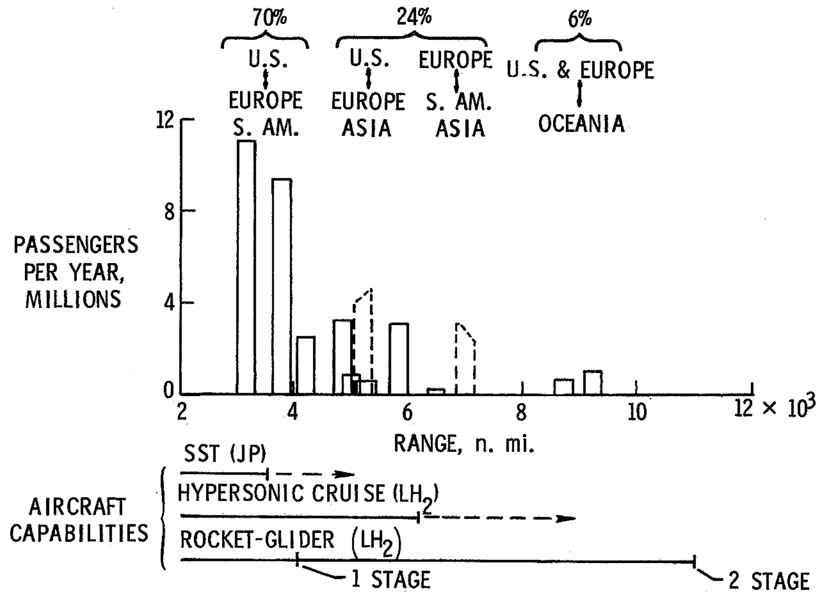


Figure 1

LOS ANGELES – PARIS SHUTTLE

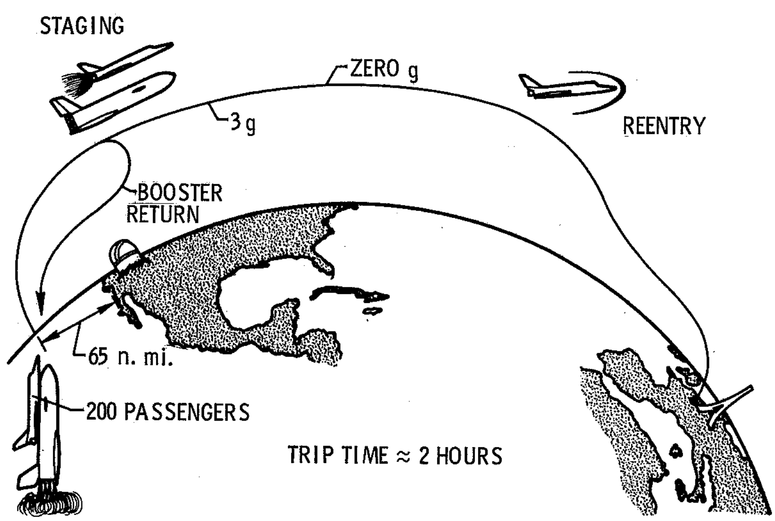
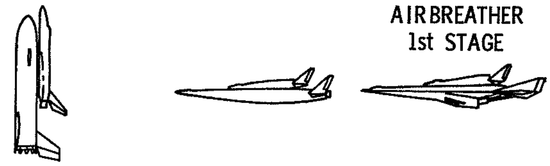


Figure 2

ROCKET-GLIDER TRANSPORTATION SYSTEMS



	SHUTTLE	HTO ROCKET	HTO AIRBREATHER 1st STAGE
L/D (GLIDER)	2	4	4
RANGE, n. mi.	11000	5500	5500
MACH NUMBER	26	18	16
GROSS WT., 10^6 kg	2.2	1.2	0.6
(10^6 lb)	(4.9)	(2.7)	(1.4)
MAX. ACCEL., g units	3	3	<1
NOISE, PNdB at 50 n. mi.	111	105	—
DOC, ¢/seat-mi. (5500 n. mi.) (AIRLINE-TYPE UTILIZATION)	12.4	5.8	4.7

Figure 3

HYPERSONIC AIRBREATHER

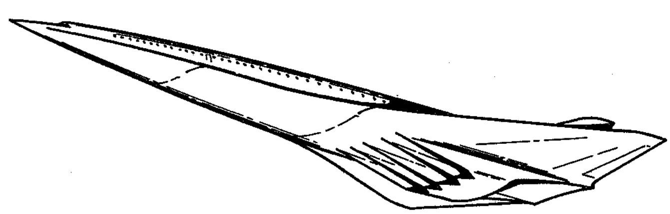


Figure 4

OVERLAND OPERATION

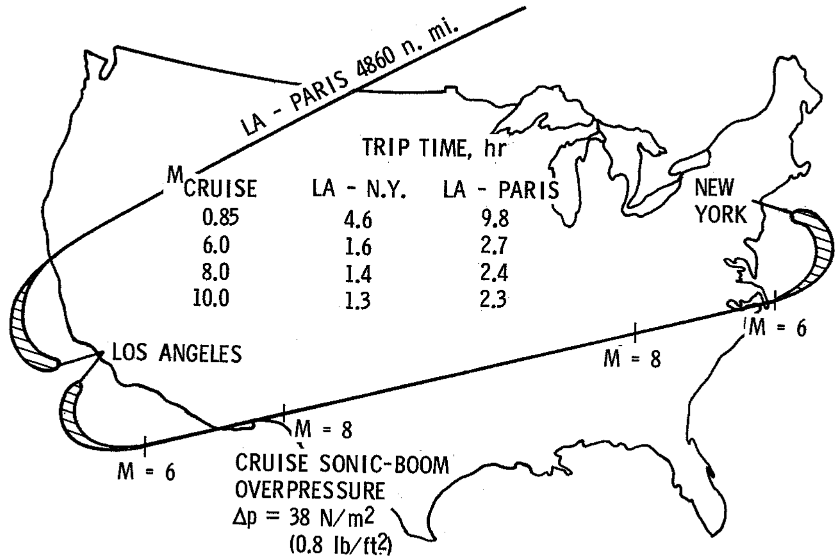


Figure 5

RANGE CAPABILITIES OF LH_2 FUELED TRANSPORTS

340 000 kg (750 000 lb); 300 PASSENGERS; CURRENT TECHNOLOGY

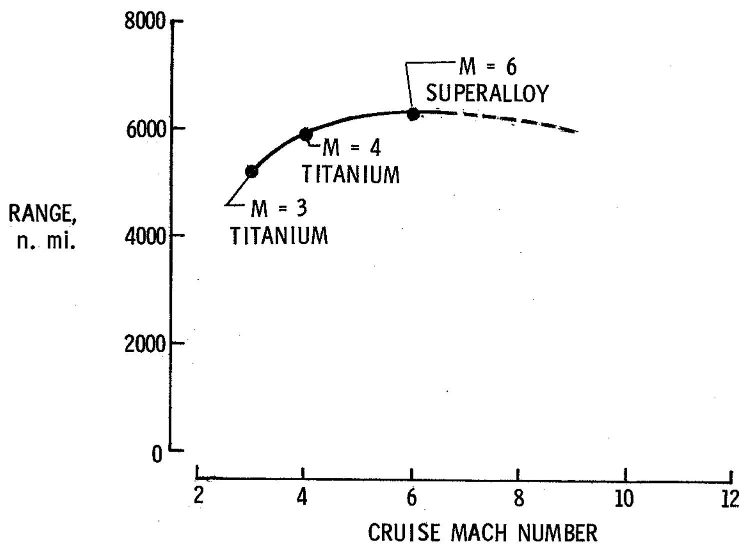


Figure 6

FUTURE FUEL COST

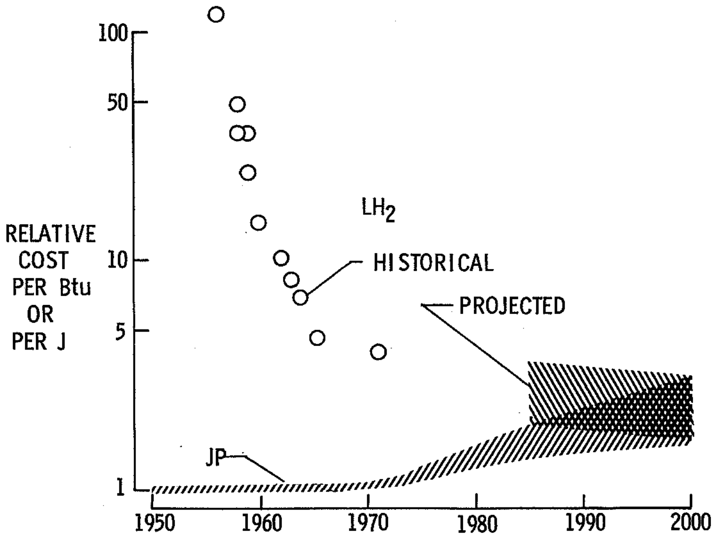


Figure 7

HYDROGEN FUEL SAFETY

NEGATIVE

- WIDE FLAMMABILITY LIMITS
- LOW IGNITION ENERGY
- DETONATION OF A CONFINED H_2-AIR MIXTURE MAY OCCUR

POSITIVE

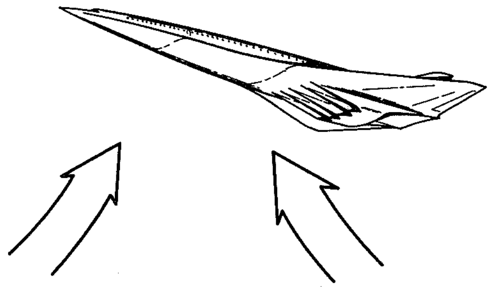
- DETONATION OF SPILLED FUEL IS HIGHLY UNLIKELY
- RAPID EVAPORATION
- LOW HEAT RADIATION
- NO SMOKE TO CAUSE ASPHYXIATION
- HYDROGEN VAPORS ARE NONTOXIC

<u>OVERALL</u>

COMPARABLE TO METHANE AND GASOLINE

Figure 8

STRUCTURAL APPROACHES



HOT SYSTEMS

- PASSIVITY
- HEAT-SHIELD TECHNOLOGY FROM SHUTTLE

COOLED SYSTEM

- CONVENTIONAL MATERIALS AND OPERATING TEMPERATURES
- COMPOSITES

Figure 9

CRUISE-VEHICLE HOT WING STRUCTURE

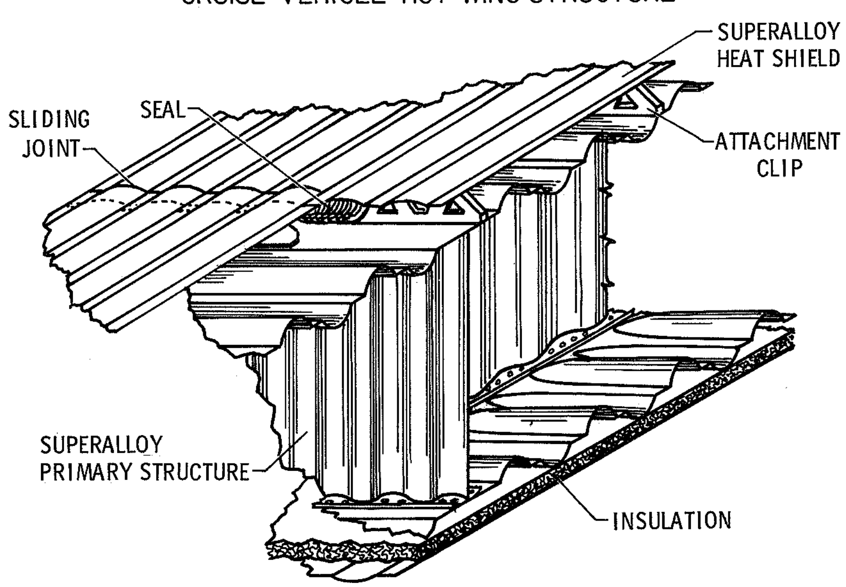


Figure 10

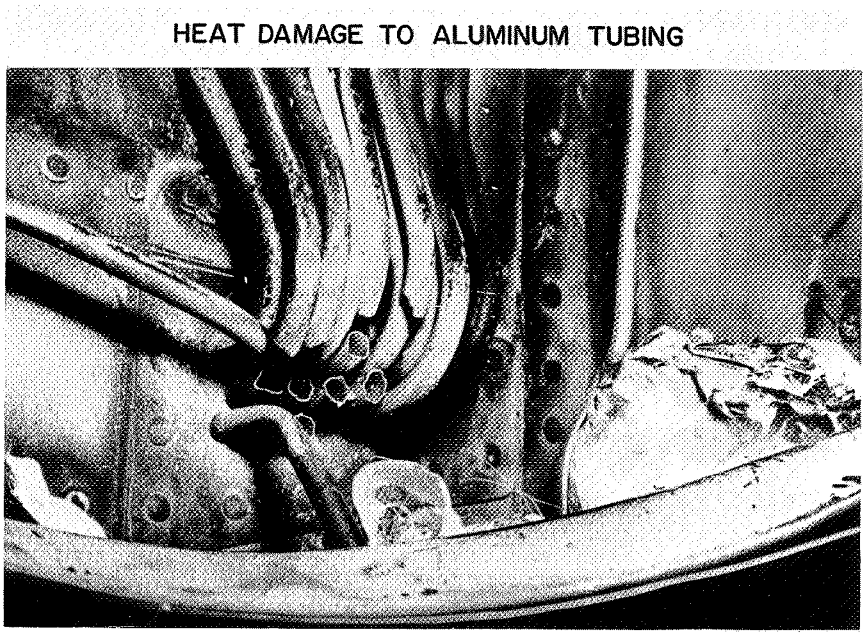


Figure 11

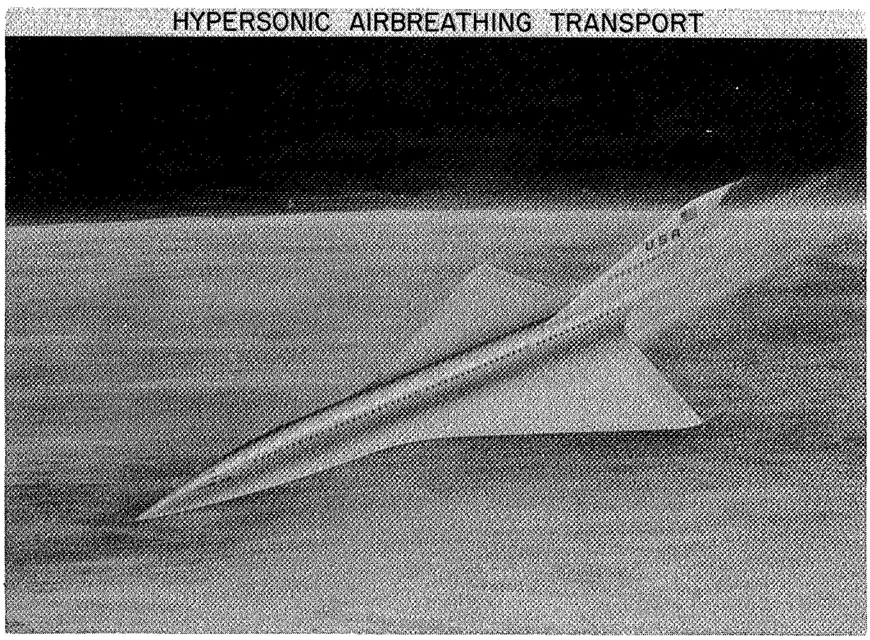


Figure 12

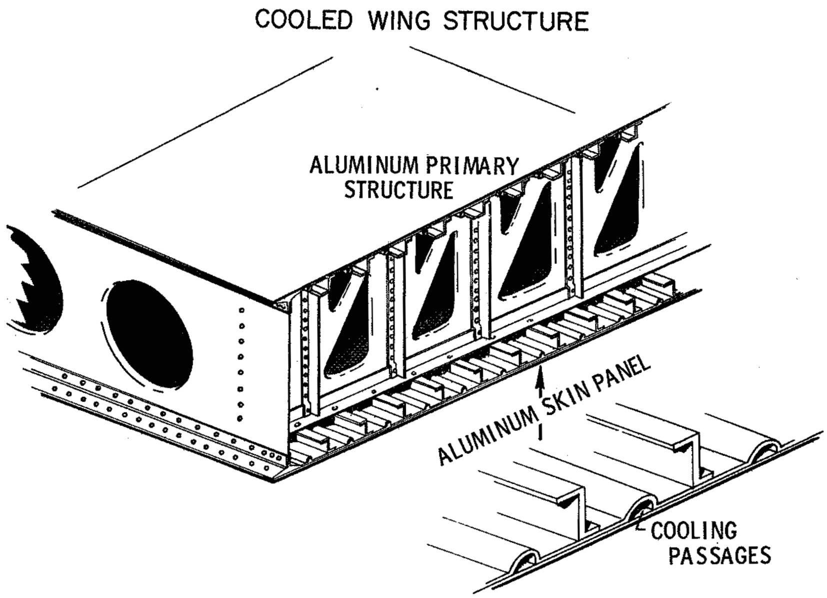


Figure 13

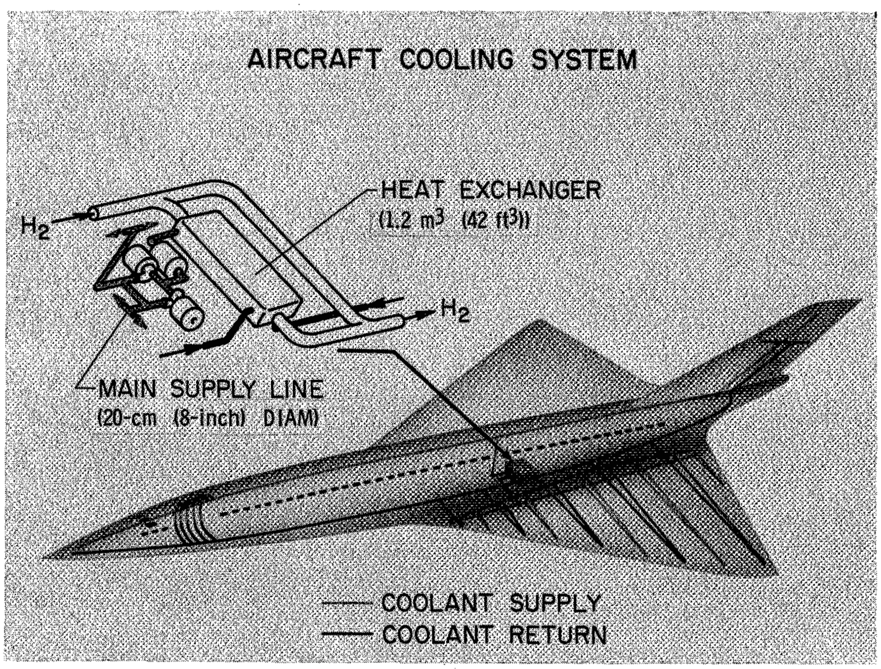


Figure 14

COOLED VS HOT STRUCTURE—TYPICAL WEIGHT DIFFERENCES

272 000-kg (600 000-lb) GROSS WEIGHT

	$W_{COOLED} - W_{HOT}$, kg	(lb)
• PIPING AND COOLANT	+4 400	(+9 800)
• PUMPS, FUEL, MISC.	+1 100	(+2 500)
• HEAT EXCHANGER	+700	(+1 600)
• AIRFRAME	-10 100	(-22 300)
• HEAT SHIELDS AND INSULATION	-6 500	(-14 400)
NET CHANGE:	-10 400	(-22 800)

Figure 15

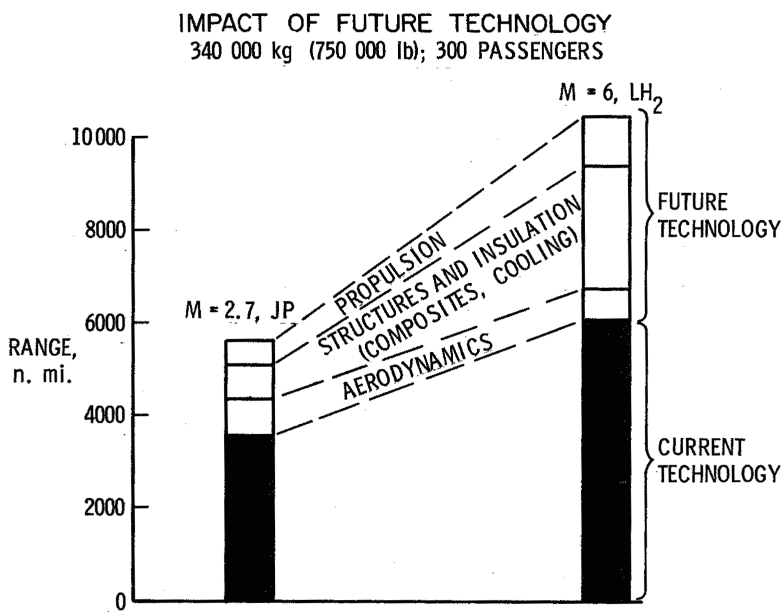


Figure 16

PROJECTED RANGE–WEIGHT CHARACTERISTICS
300 PASSENGERS

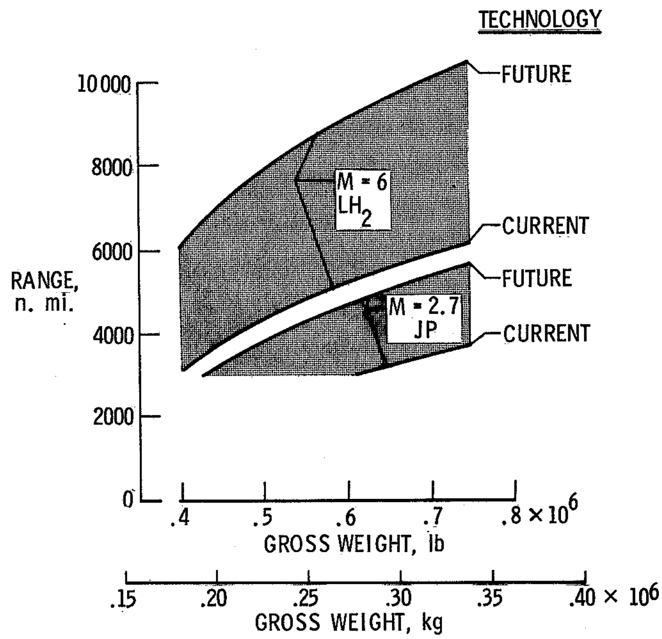


Figure 17

PROJECTED DOC FOR HYPERSONIC AIRBREATHERS
300 PASSENGERS; 6000 n. mi.

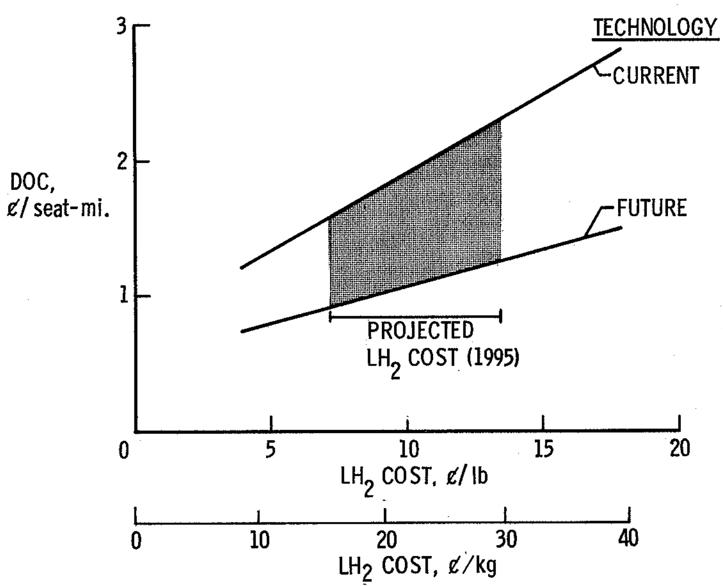
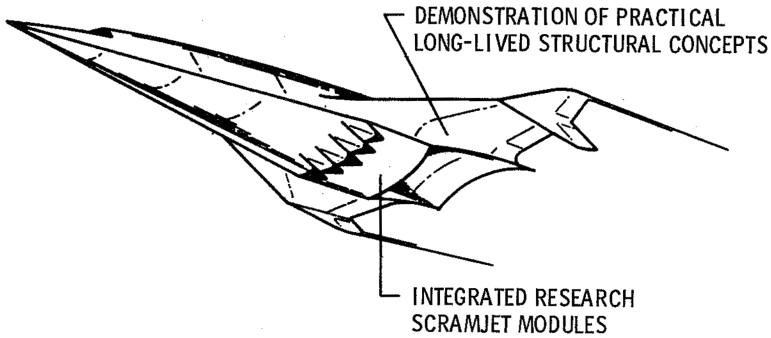


Figure 18

RESEARCH AIRPLANE CONCEPT AND SPECIFICATIONS



- GROSS TAKE-OFF WEIGHT ≈ 36 000 kg (80 000 lb)
- LENGTH ≈ 24 m (80 ft)
- CONVENTIONAL TAKE-OFF AND LANDING
- MAXIMUM SPEED, M = 8 TO 12
- EXISTING ROCKET (PRIME PROPULSION)
- MODULAR RESEARCH AIRBREATHING ENGINES
- 5-MINUTE CRUISE AT MAXIMUM SPEED

Figure 19

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