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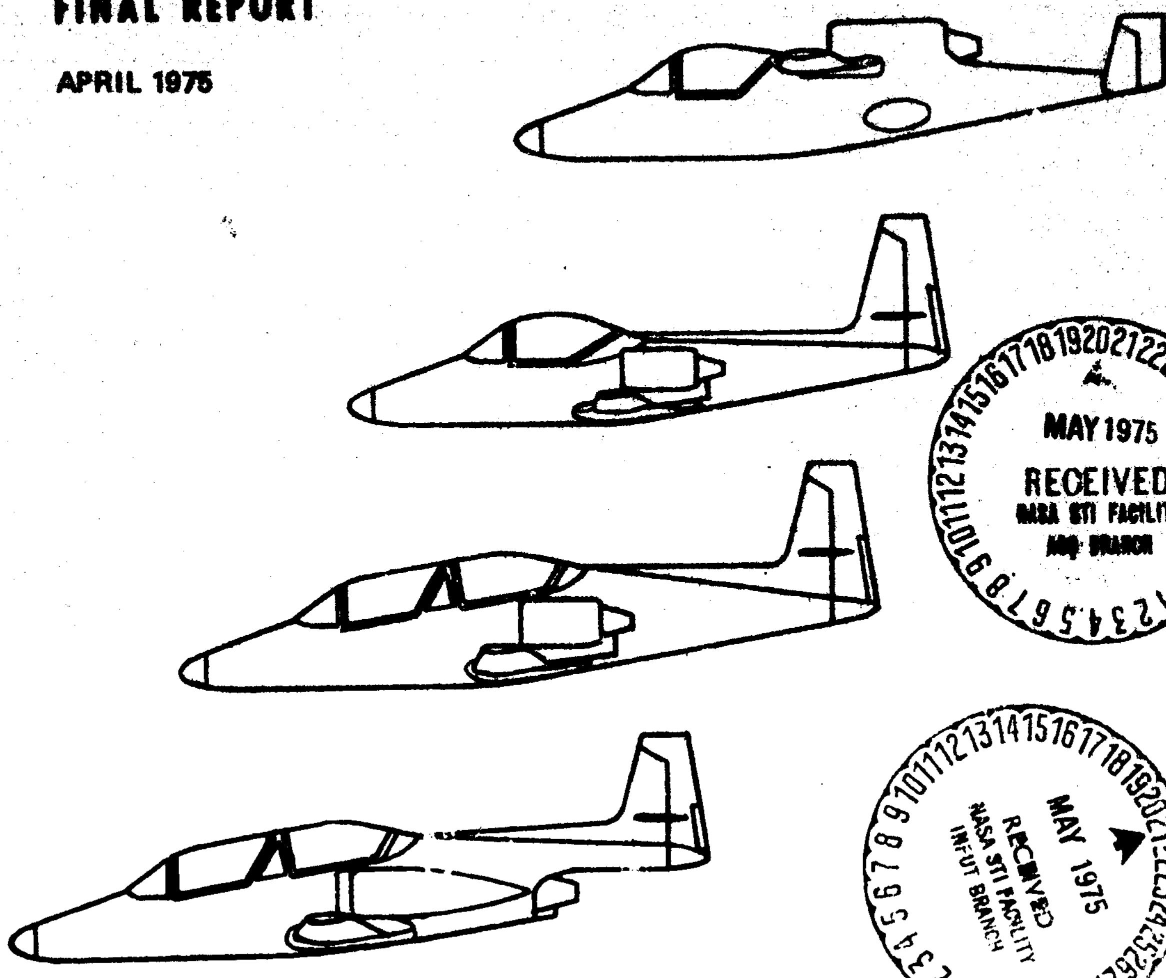
STUDY OF SMALL CIVIL TURBOFAN ENGINES APPLICABLE TO MILITARY TRAINER AIRPLANES

FINAL REPORT

APRIL 1975



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STUDY OF SMALL CIVIL TURBOFAN ENGINES APPLICABLE TO MILITARY TRAINER AIRPLANES FINAL REPORT

By R. W. Heldenbrand, G. L. Merrill, G. A. Burnett

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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

Systems Studies Division

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STUDY OF SMALL CIVIL TURBOFAN ENGINES APPLICABLE TO MILITARY TRAINER AIRPLANES

By R. W. Heldenbrand, G. L. Merrill, and G. A. Burnett

SUMMARY

This report presents the results of a study sponsored by the NASA Ames Research Center, Systems Studies Division (Contract NAS2-6799, Mod. 2), regarding the applicability of small turbofan engines to military primary trainer airplanes.

Earlier efforts accomplished under the original contract work statement showed that efficient turbofan propulsion systems could be designed for and extended successfully to smaller and lower speed civil airplanes than have been considered heretofore. This follow-on study by NASA-Ames and AiResearch expands that work to include the application of these small turbofan concepts to military trainer airplanes, and to establish the potential for commonality between civil and military engines. With the aid of the NASA General Aviation Synthesis (Computer) Program, four primary trainer configurations were defined and studied. A "best" engine was defined for the trainer mission, and sensitivity analyses were performed to determine the effects on airplane size and efficiency of wing loading, power loading, configuration, aerodynamic quality, and engine quality.

The principal conclusion drawn from the results of this investigation is that a turbofan propulsion system for a small civil aircraft is also applicable to military trainer airplanes. Aircraft designed with these engines to meet military requirements for basic trainers are smaller, less costly and more efficient than existing basic trainer aircraft or basic trainer aircraft that have been conceptually designed with high subsonic turbojets or turbofans. In addition, substantial benefits may accrue to both military and civil sectors if this commonality is exploited.

INTRODUCTION

Turbofan propulsion, as applicable to smaller, lower speed general-aviation airplanes than have been designed and produced to date, was thoroughly investigated in the initial program conducted under this contract (Reference 1). The significant results of that program are given in Table I, and the specific airplane design addressed in that study is shown on Figure 1.

Historically, military aircraft engine developments have provided the genesis and economic impetus for nearly all civil aircraft engine developments. Thus, a logical continuation of the effort to define advanced turbofan propulsion for general-aviation airplanes would be to identify military applications for such engines. It was determined that a follow-on study should be undertaken to quantify airplane performance and cost advantages for a new, turbofan-powered military primary trainer. The objective of the study would be to identify the technical requirements of a new military aircraft engine, and to establish the commonality of this engine design with civil engine requirements. Because the performance envelope of a primary trainer is typically consistent with that of many general-aviation airplanes, a "best" turbofan for a trainer should be directly applicable to potential general-aviation airplane designs. Following discussions with United States Air Force and Navy training headquarters personnel, the program was revised to permit a more comprehensive study of military trainers. The investigation of the civil airplane was consequently deferred to a later date and will be addressed in a follow-on program.

The military trainer design and mission criteria that were selected as guidelines for this study were defined in the USAF Mission Analysis Report noted in Reference 2 (referred to in this report as the "Randolph study"). These criteria were developed for a primary trainer designated TA-2 in the USAF report. Use of the USAF data has permitted the definition and investigation of the following four trainer configurations:

- o Single-engine, side-by-side seats
- o Twin-engine, side-by-side seats
- o Twin-engine, tandem seats
- o Single-engine, tandem seats

Although work was begun with the single-engine, side-by-side configuration, the study tasks were designed to give primary

TABLE I. SUMMARY RESULTS OF INITIAL PROGRAM: "A STUDY OF SMALL TURBOFAN ENGINES APPLICABLE TO GENERAL AVIATION AIRCRAFT."

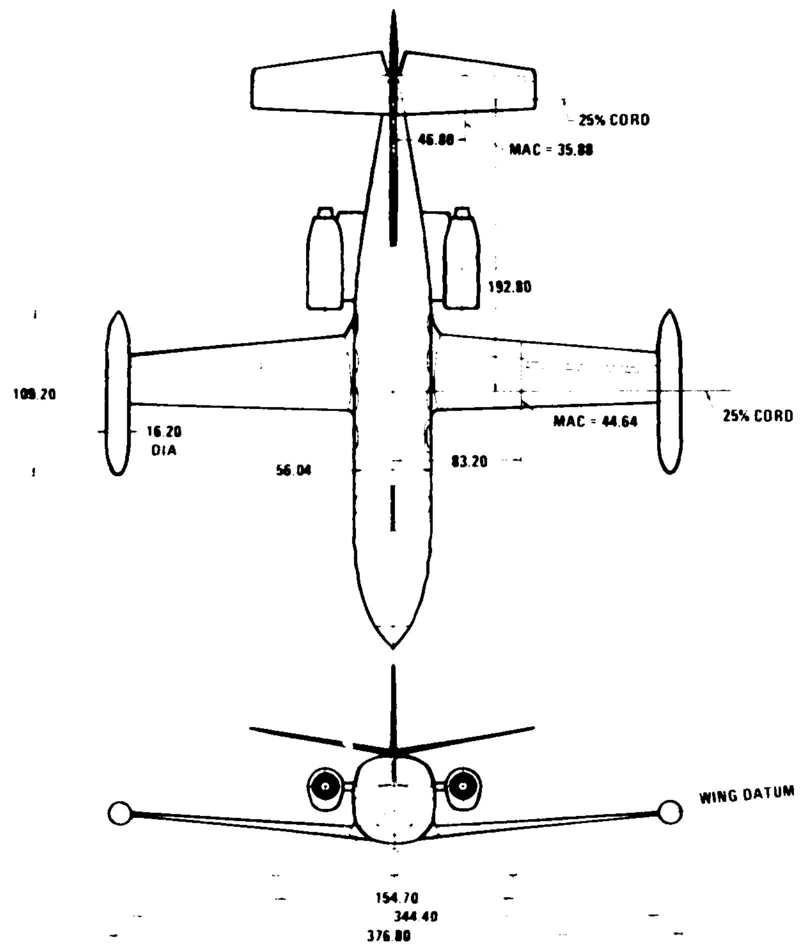
The initial study, completed in 1973, investigated the applicability of turbofans to high-performance civil light twin engine aircraft. Significant results achieved were:

- Definition of engine/airplane design, performance and cost interrelationships, using NASA-AMES general aviation synthesis program (GASP).

- Credible preliminary design of an attractive airplane, demonstrating the applicability of turbofans to much smaller, lower cost airplanes than previously thought possible.

- Better understanding of propulsion principles for lower speed, lower cost airplanes.

- Understanding that military sponsorship of development and procurement of turbofans in this class would expedite availability.



GROSS WEIGHT, LB	6230
EMPTY WEIGHT, LB	3363
USEFUL LOAD, LB	2867
MAX FUEL WEIGHT, LB	1931
MAX CABIN LOAD (6 PAX + BAGS), LB	1200
LENGTH, FT	32.13
WINGSPAN, FT	28.1
WING AREA, FT^2	100.5
WING LOADING, PSF	62
ASPECT RATIO	7.86
FOWLER FLAP SPAN, %	100
HORIZONTAL TAIL AREA, FT^2	44.8
VERTICAL TAIL AREA, FT^2	23.7
TURBOFAN SEA LEVEL STATIC THRUST, LB	1300
DESIGN CRUISE SPEED, KNOTS	350
DESIGN CRUISE ALTITUDE, FT	24,000
SERVICE CEILING, FT	36,000
TAKEOFF DISTANCE (TO 35 FT), FT	2421
MAX FUEL RANGE, N. MILES	1000
CALCULATED DRAG POLAR (CRUISE CONFIGURATION)	$C_D = 0.0301 + 0.0484\ C_L^2$
CALCULATED BASIC PURCHASE PRICE	$306,000

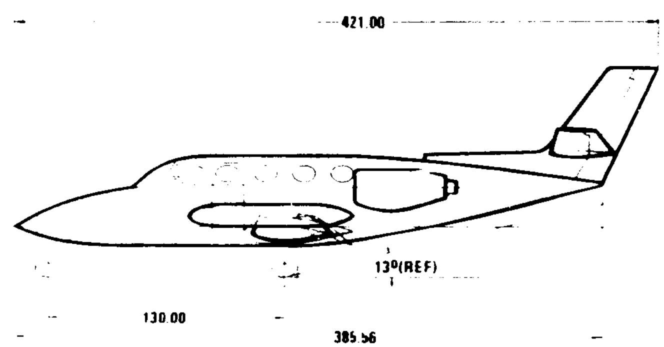


Figure 1. General Aviation Light-Twin Three-View Schematic and Design Results from Initial Study.

emphasis to the twin-engine, side-by-side airplane. The NASA-Ames General Aviation Synthesis (computer) Program (GASP) was used throughout the investigation for airplane design definition, and for the evaluation of the effects of various design parameters on airplane size and cost. With the GASP program, an initial sensitivity study was performed with the twin-engine, side-by-side airplane to establish criteria for selection of wing loading and thrust loading appropriate to the airplane performance requirements. These criteria were considered applicable to the other three airplane configurations. The majority of the airplane and engine sensitivity studies and trade-off analyses were performed on the twin-engine, side-by-side airplane.

Conceptual designs of the four airplane configurations were originated by AiResearch; however, in order to ensure that these designs met certain military trainer requirements, informal discussions were held with personnel of the USAF Air Training Command Headquarters, Randolph Air Force Base, Texas, and of the USN Air Training Center, Corpus Christi Naval Air Station, Texas, as well as various other military offices. Further, to ensure their overall credibility, all the designs were reviewed by the Cessna Aircraft Company, under subcontract, for equipment fit, balance, weight, performance, stability, control, and other design considerations. The specific study guidelines agreed to by AiResearch and NASA-Ames are given in Table II. Under these guidelines, only the Air Force TA-2 airplane requirements given in the Randolph study were addressed. Each of the four airplane configurations was designed to meet these requirements, with airplane size, efficiency, and engine size "solutions" varying.

A baseline engine cycle for the trainers was defined by addressing and separately quantifying the elements of overall propulsion system efficiency as applicable to the trainer performance and mission requirements.

The airplane cruise speed and initial estimates of the airplane's fuel and engine weight fractions are the fundamental parameters evaluated in the determination of a credible baseline engine cycle by the methods described in Reference 1.

The mechanical arrangement, aerodynamic component selection, and detail design concepts for the baseline engine were chosen to provide for low manufacturing cost, high reliability, and maintainability. The basic engine design philosophy was to achieve the simplest possible high-bypass-ratio turbofan configuration, having two spools, no reduction gears, only two frames,

TABLE II. GUIDELINES FOR THE STUDY OF SMALL TURBOFANS APPLICABLE TO MILITARY AND CIVIL AIRCRAFT.

Airplane designs to be totally responsive to the TA-2 (primary trainer) performance, mission, and configuration requirements reported in--

"Mission Analysis on Future Undergraduate Pilot Training 1975 through 1990," by mission analysis study group, Randolph AFB, Texas (Jan. 1972).

o	Define four airplanes	Single-Engine, Side-by-Side Seats Twin-Engine, Side-by-Side Seats Single-Engine, Tandem Seats Twin-Engine, Tandem Seats
o	Define a "Best" engine for	463 km/hr (250 kt) Cruise Speed 4572 m (15,000 ft) Cruise Altitude And Other Mission Requirements
o	Do parametric sensitivity analyses	Wing Loading/Thrust Loading Design/Configuration Engine Quality
o	Identify engine and airplane with lowest operating cost.	

four bearings in two bearing cavities, castable subsonic aerodynamic components, low rotor speed per unit of airflow, low stresses, modest temperatures, and modular assemblies. The conceptual design layout for the baseline engine is shown in Figure 2. This engine design provided the basis for later, in-depth, parametric cycle optimization studies, and for the definition of engine candidates for aircraft synthesis evaluation of engine cycle quality.

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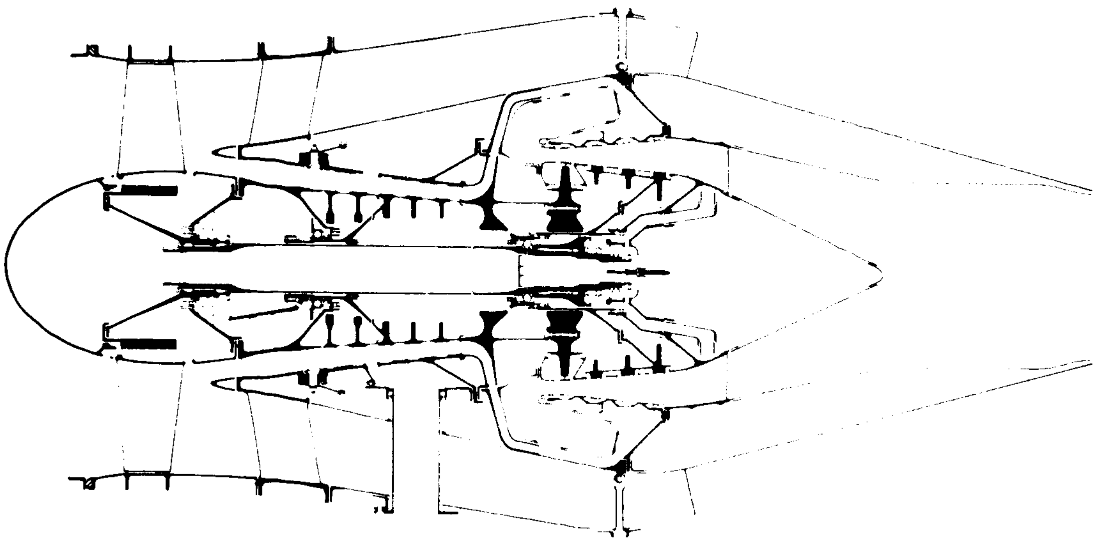

FOR 250 KT/15,000 FT. CRUISE

- FAN PRESSURE RATIO 1.3
- CORE PRESSURE RATIO 7.0
- TURBINE INLET TEMPERATURE 1228°K (1750°F)
- BYPASS RATIO ~9

Figure 2. Illustration of the Baseline Engine Design for Military Trainer Study.

SYMBOLS

AR	Aspect ratio
BPR	Bypass ratio
Btu	British thermal unit
°C	Degrees centigrade
C_D	Drag coefficient
C_{Di}	Induced drag coefficient
$C_{D\ wet}$	Drag coefficient referenced to the wetted area
C_L	Lift coefficient
C_P	Specific heat of constant pressure
CU	Customary units
EPNdB	Effective perceived noise level
e	Oswald efficiency factor
°F	Degrees Fahrenheit
F	Engine thrust, N (lbf)
f_a	Fuel-air ratio
FAR	Federal Air Regulations
F_N	Net thrust, N (lbf)
fpm	Feet per minute
ft	Feet
F_{SLS}	Sea level static thrust, N (lbf)
F/W_a	Engine specific thrust, or thrust per unit airflow N-s/kg [lbf/(lbm/sec)]
g	Acceleration of gravity
gal	Gallon
hp	Horsepower
hr	Hour
in.	Inch(es)

SYMBOLS (Contd)

ISA	International Standard Atmosphere
J	Joules and work conversion factor 778/550
°K	Degrees Kelvin
K	Thousand
kg	Kilogram
kt	Knot
L	Length
lb	Pound(s)
m	Meter
mm	Millimeter
mph	Miles per hour
N	Newton
N_G	Gas generator speed, rpm
n. mi.	Nautical miles
p	Pressure, lb per sq ft
PR	Pressure ratio
psf	Pounds per square foot
psi	Pounds per square inch
q	Dynamic pressure
°R	Degrees Rankine
S	Wing area, sq m (sq ft)
SI	Systeme International
SLF	Sustained load factor
SLS	Sea level static
sec	Second

SYMBOLS (Contd)

S_{wet}	Wetted area, sq m (sq ft)
T	Temperature, °K (°F or °R)
TAS	True airspeed, knots
TSFC	Thrust specific fuel consumption, kg/N-hr [(lbm/hr)/lbf]
ΔT	Temperature change
U	Rotational velocity, m/sec, (fps)
V_a	Axial velocity
V_s	Airplane stall speed, km/hr (mph)
W	Weight, kg (lbm)
W/S	Wing loading, kg/m^2 (lbm/ft^2)
η	Efficiency (actual work/ideal work)
λ	Turbine work factor ($gJc_p \Delta T/U^2$)
ϕ	Flow coefficient (V_a/U)
ψ	Compressor work coefficient ($gJc_p \Delta T/U^2$)

GENERAL AVIATION SYNTHESIS PROGRAM (GASP) DESCRIPTION

The airplane and engine performance and design parameters were combined in the NASA-Ames General Aviation Synthesis Program (GASP) to aid in performing preliminary design studies of the various trainer configurations. This computer program was designed by the NASA-Ames Systems Studies Division, and is described in Society of Automotive Engineers Paper 73033 (Reference 3). It was utilized extensively by AiResearch during the study of general-aviation turbofan engines reported in Reference 1. During the study of military trainers, the GASP program was refined to permit the direct input of improved, separately calculated engine performance maps. It was found that for aircraft of this size, the performance penalties for engine bleed and shaft power extraction were more significant than for larger aircraft. Therefore, a revised and streamlined method was derived for inputting complete engine performance data from a separate, off-design, cycle matching program. Throughout these studies, GASP proved to be a valuable tool for conducting:

- Airplane configuration comparisons
- Comparative assessments of aircraft performance and economics
- Performance trade-off studies and parametric analyses
- Assessments of advanced technology

GASP was described briefly in Reference 1; however, it is pertinent to review its principal features here. The following description, taken from Reference 3, defines the calculation flow paths through the various airplane analyses. As illustrated in Figure 3, the control module directs the computational flow through the other modules of the synthesis with module sequencing determined by parameter input to the control module, as well as the normal mode of operation. Input for each module consists of quantities generated internally by other modules, or design variables that are input directly, or both. The integrated approach established in the program methodology ensures that the multiple effects of design variables are continuously accounted for in the aircraft sizing procedures.

The airplane geometry module, Figure 4, computes the sizing of the wing, fuselage, empennage, and engine nacelles. The wing geometry is characterized by the aspect ratio, taper ratio, airfoil thickness-chord ratio, quarter-chord sweep, etc. The fuselage shape and volume are related to the number of passengers, seating arrangement, and fuselage configuration.

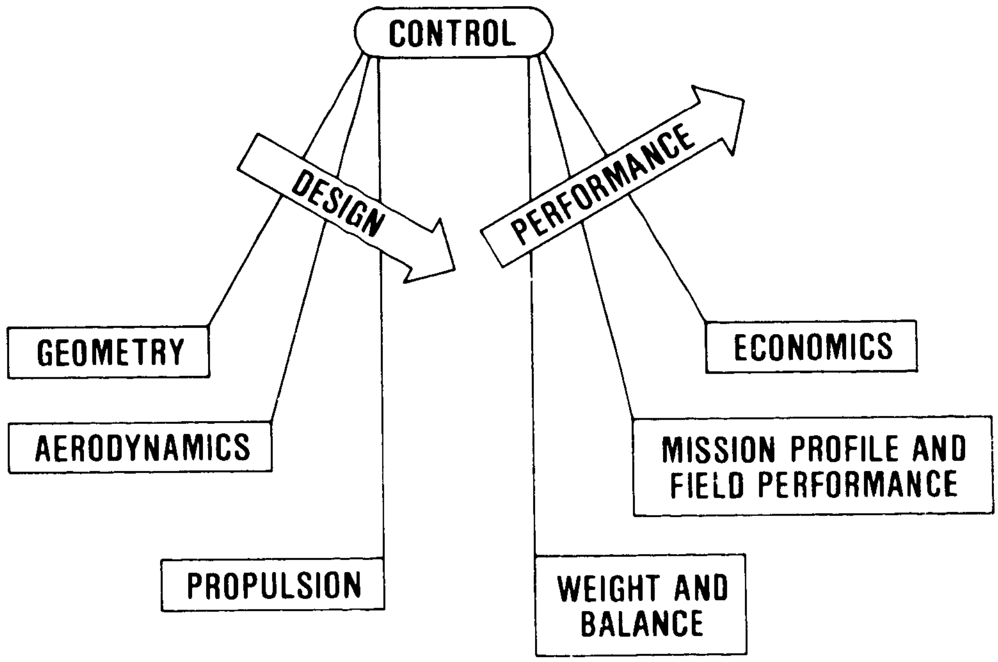

Figure 3. Primary Program Modules of GASP.

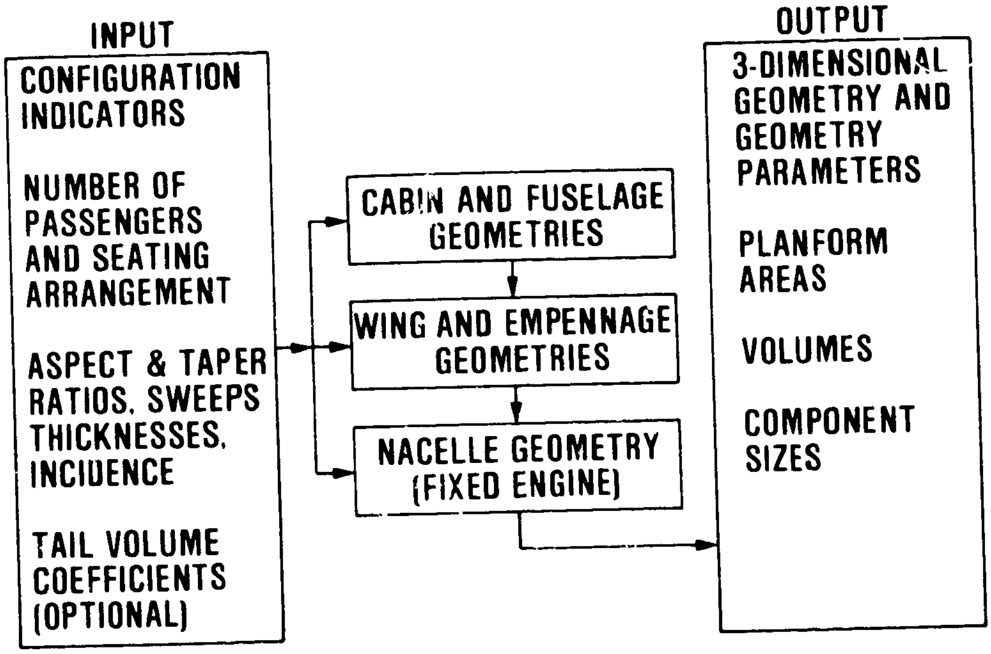

Figure 4. GASP Geometry Module Description.

The aerodynamic module, Figure 5, computes the airplane lift and drag characteristics on a point-by-point basis during takeoff, climb, cruise, and landing. The cruise drag is determined for each aircraft component based on Reynolds number and Mach number. Form factors are used to account for body shape and component interference, or for duplicating the drag of an existing aircraft. Cruise lift is based on an input value of angle of attack for zero lift and a semi-empirical method for computing the lift curve slope. The effects of plain, split, slotted, and Fowler-type trailing-edge flaps are simulated for high-lift increments in optionally selected takeoff and landing configurations. The methodology accounts for flap deflection, span and chord, wing sweep, thickness, and aspect ratio. Nacelle drag is accounted for as either an aircraft drag or as a propulsion system drag, reducing uninstalled thrust, and increasing specific fuel consumption.

The propulsion module, Figure 6, computes the engine performance, dimensions, weight, and volume required for airplane synthesis definition. Complete engine data including thrust, fuel flow, and airflow maps are input to the program with installation losses included. The propulsion system is initially sized to match the cruise drag and a rate of climb requirement at the end of climb. Program options permit engine sizing for specified takeoff distance, or sizing such that the climb requirements of FAR Part 25 are satisfied. Engine diameter and weight can be internally calculated as a function of engine front-face, design-point Mach number, the hub-tip diameter ratio, and engine airflow required, or a separately derived engine specific weight may be input to the program.

A weight and balance analysis, Figure 7, is completed on the airplane after the configuration geometry is defined and the engine size and weight are calculated. Weights for the various airplane components are estimated from trend equations derived from the general-aviation airplane class correlations. Available fuel is determined from the empty airplane weight, which is computed by summarizing the subsystem weights and the input gross weight and payload.

The airplane mission module, Figure 8, computes the airplane performance during taxi, takeoff, climb, cruise, and landing. Options are available in this module for calculating engine-out and accelerate/stop distance, best rate of climb, best lift-to-drag ratio, and additional airplane and engine operating characteristics. The effects of gear and flap retraction and ground effect are accounted for during the takeoff segment. Fuel reserve inputs are accounted for in the cruise segment. Range is

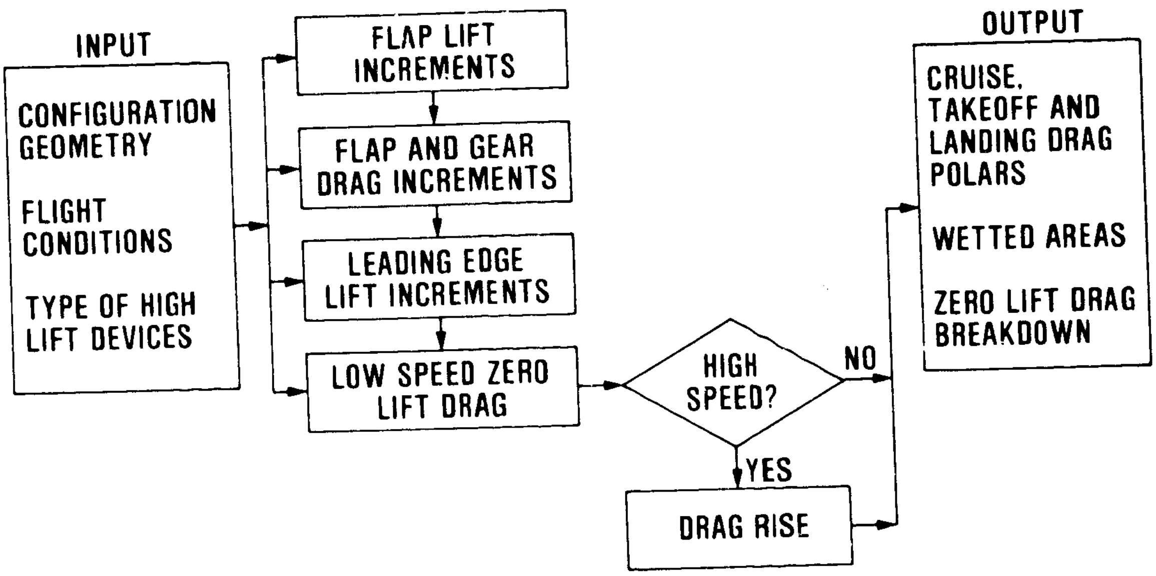

Figure 5. GASP Aerodynamic Module Description.

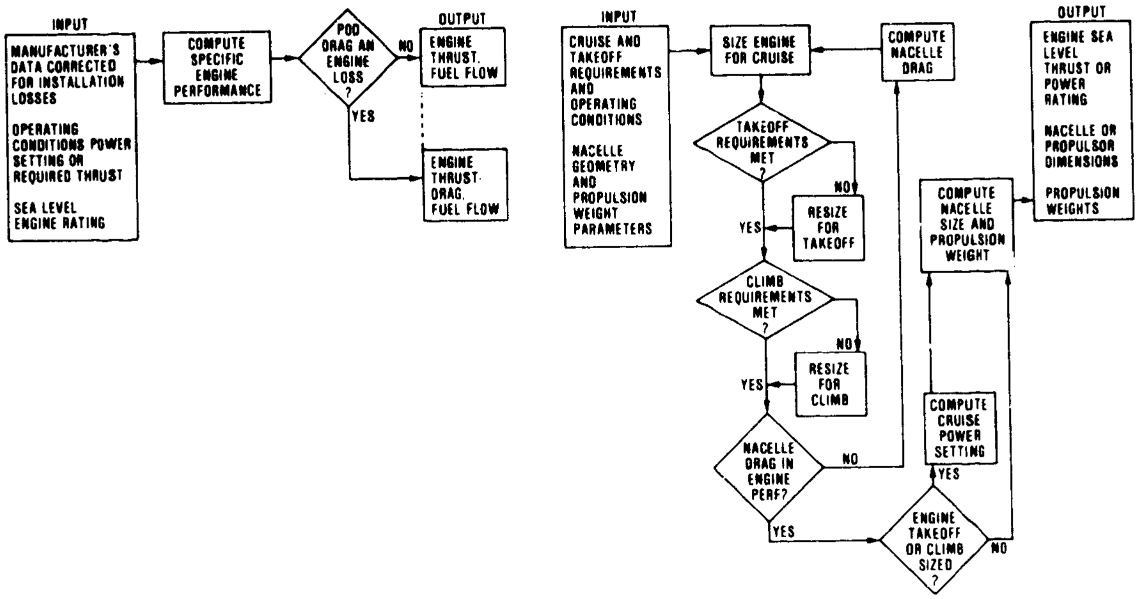

Figure 6. GASP Propulsion Module Description.

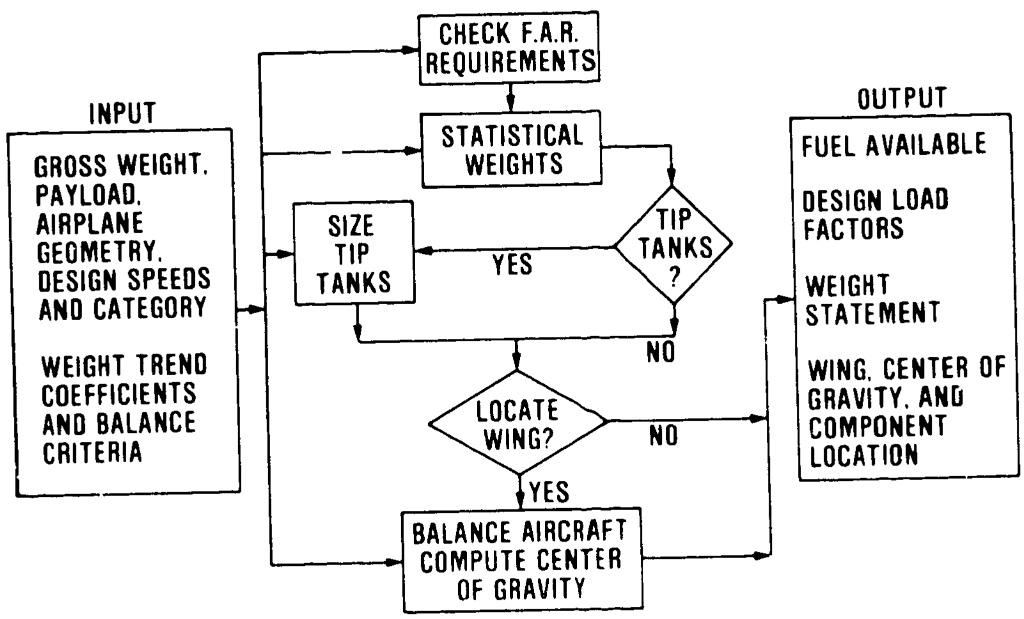

Figure 7. GASP Weight and Balance Module Description.

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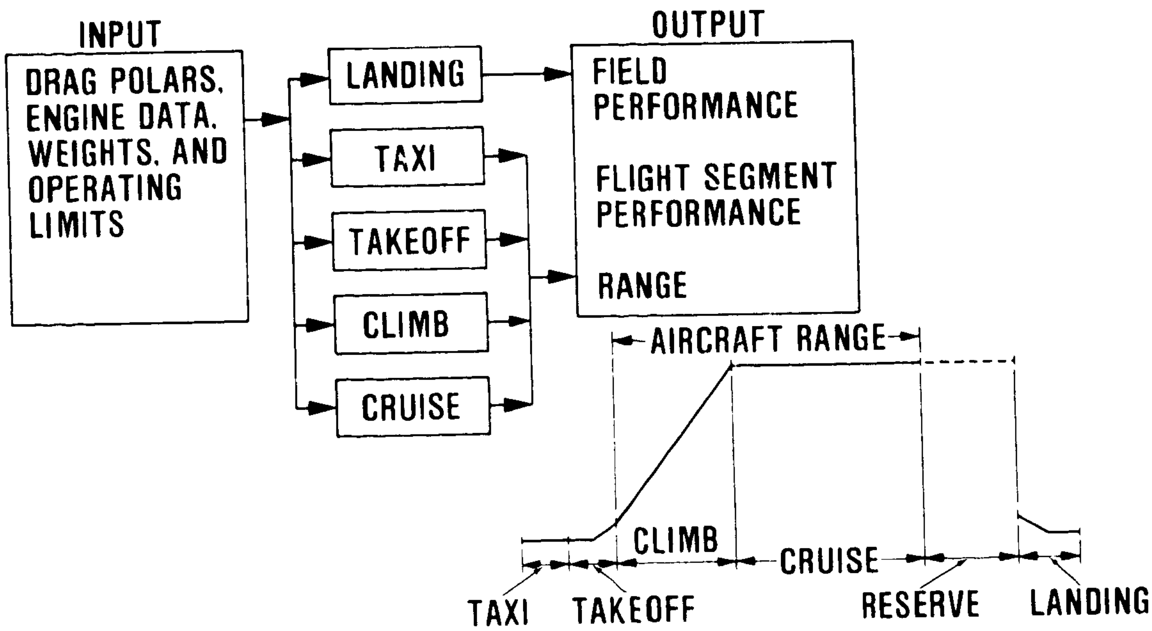


Figure 8. GASP Mission Profile and Airfield Performance Module Description.

accounted for during the climb and cruise segment. When a specific range definition is required, a program option is utilized that iterates on the airplane size until the calculated range is within a specified tolerance of the required range.

The economics module, Figure 9, is used for civil aircraft cost evaluations with good correlation. However, for the military trainer airplanes, it was considered necessary to include life-cycle cost data that incorporated cost factors and calculation formats consistent with USAF experience. These data were not available within the time span of this study, thus, the economics module was not utilized to its full capability.

The synthesis program calculation sequence, Figure 10, has been designed to provide an iterative, integrated method which ensures that the results contain the effects of design interaction within each calculation module and between modules. For example, a change in specified wing loading affects wing area, tail size, lift, drag, structural weight, aircraft performance, and finally, engine and airplane "solution" size. Some of the effects are minor while others are significant; however, all effects that impact the airplane "solution" are iteratively accounted for.

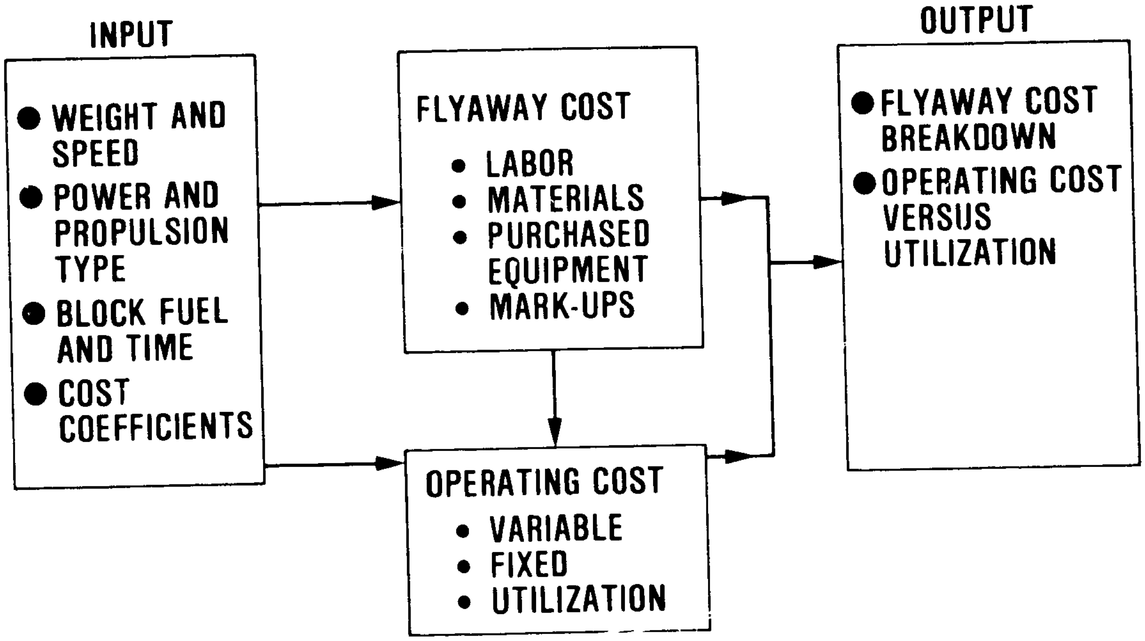


Figure 9. GASP Economics Module Description.

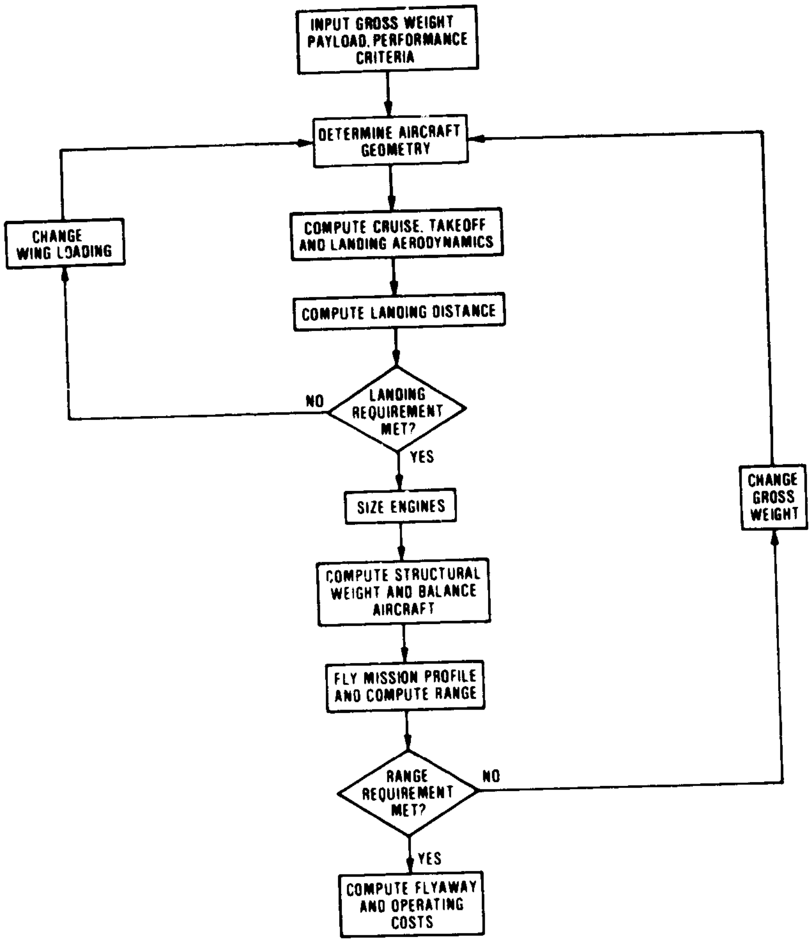


Figure 10. GASP Normal Calculation Flowpath Description.

BASELINE ENGINE DEFINITION

General Design Considerations

In the final report on the first investigation conducted under this contract (Reference 1), methods were discussed whereby a "nearly best" turbofan engine cycle can be defined directly and quickly, without recourse to extensive cycle analysis, parametric trade-off analyses, and comprehensive preliminary design exercises. By separately quantifying the elements of overall propulsion system efficiency (propulsive efficiency, thermal efficiency, airplane performance and mission-related installed drag, and installed weight and drag), an engine that nearly minimizes airplane size and cost can be readily defined. When the airplane required-cruise-thrust sizes the engine, the cruise flight speed and altitude may be assumed to be the engine design point. Then the fan pressure ratio and core jet velocity that maximize propulsive efficiency at the design point can be calculated directly. For maximum thermal efficiency, the highest practical cycle pressure ratio is chosen. For minimum core size and weight, the highest practical turbine inlet temperature is chosen. However, maximum propulsive efficiency must be traded-off against fan system weight and nacelle drag, and maximum thermal efficiency must be traded-off against core weight to achieve the "best" engine that minimizes airplane size, initial cost, and operating cost. While a "nearly best" engine can be defined readily, an optimum engine can be determined only from aircraft synthesis sensitivity and trade-off analyses. Because the interrelationships between engine and airplane performance qualities are complex, synthesis analyses are vital in defining the most cost-effective engine. This has, of course, been the procedure used in this investigation of primary trainers.

The airplane design point for this study was set at 463 km/hr (250 kt), at 4572 m (15,000 ft) altitude. For airplane optimization studies, a baseline engine was defined that would provide near maximum net propulsive and thermal efficiencies at this design point. The 1.3 fan pressure ratio chosen was estimated to give the best balance between propulsive efficiency, fan system size and weight, and nacelle drag. The 7.0 core pressure ratio chosen was judged to provide the best balance between thermal efficiency and core weight. The 1255°K (1800°F) turbine inlet temperature was selected to minimize the cost of the engine core; that is, the highest temperature that would not require expensive turbine blade cooling. These principal determinants of cycle quality, plus the additional efficiency and loss assumptions made for the baseline engine, are listed in Table III. In later modeling for engine performance mapping, many of the values were adjusted based on further evaluation of component design

TABLE III. BASELINE ENGINE CYCLE AT SEA LEVEL STATIC DESIGN POINT

Fan Pressure ratio	1.3
Core Compressor Pressure Ratio	7.0
Core Turbine Pressure Ratio	2.99
Fan Turbine Pressure Ratio	2.83
Core Jet Nozzle Pressure Ratio	1.2
Bypass Ratio	7.81
Turbine Inlet Temperature	1255°K (2260°R, 1800°F)
Inlet Pressure Loss	0%
Inter-Compressor Pressure Loss	0%
Combustor Pressure Loss	4%
Core Exhaust-Duct Pressure Loss	2%
Bypass-Duct Pressure Loss	2%
Fan Efficiency	89%
Core Compressor Efficiency	82%
Combustor Efficiency	100%
Core Turbine Efficiency	86%
Fan Turbine Efficiency	89%
Nozzle Velocity Coefficient (Both Nozzles)	0.985
Mechanical Efficiency (Both Spools)	100%
Shaft Power Extraction	0
Bleed-Air Extraction	0
Overboard Leakage Loss (CPD)	1.9%
Thrust Specific Fuel Consumption	0.0370 kg/N-hr (0.363 lb/hr/lb)
Specific Thrust	207.41 N-sec/kg (21.15 lb/lb/sec)

and the determination of component performance maps. Further adjustments were made to accommodate shaft power extraction and compressor bleed for airframe needs.

Design studies were conducted on overall engine and component configurations for the baseline engine. In accordance with the contemporary engine design principles discussed in the final report of the initial investigation (Reference 1), the baseline engine design was to exhibit an understanding that by proper choice of aerodynamic configurations, costly parasitic machinery could be avoided. The basic two-frame, four-bearing configuration with direct fan drive was considered essential, even at bypass ratios as high as 10:1. With this configuration, bearings, gears, seals, splines, couplings, fasteners, shaft elements, expe expensive lubrication system plumbing, pumps, and cooling devices could be minimized. If the engine design were subjected to a design-to-cost exercise, this configuration would permit a greater emphasis on aerodynamic and thermodynamic quality. Engine performance for a specified cost would thereby be maximized.

Component Configurations and Design Parameters

Core compressor. - The core compressor is the key component in achieving the desired engine configuration. In turn, the key compressor design parameter is the rotational speed per unit airflow. If this parameter has a low value, the engine core may have a large center hole to accommodate the fan driving shaft when supported on just two bearings. Table IV lists the principle parameters chosen for a preliminary compressor design that was one of several examined for the baseline engine. This compressor design is all subsonic. With thicker airfoils permitted by subsonic design, the castability of the compressor stages is enhanced, which can result in significant cost savings. With low axial velocity and the flow-path configuration chosen to maximize the height of the flow-path annulus, the potential for high efficiency is inherent in the design. The six-stage, low-speed compressor design selected addresses all the criteria developed in the initial investigation for cost-effective engine design.

Fan stage. - In detail engine design, the fan and its turbine must be the subject of extensive trade-off analyses to ensure that the fan spool makes the maximum contribution to overall propulsion system efficiency and engine cost-effectiveness. The size, weight, and drag of the nacelle are, in large part, a function of the fan design. In addition, the weight of the fan spool, the inlet and bypass duct pressure losses, the efficiencies of the fan and the turbine, the core exhaust-duct pressure loss, and the cost of the fan spool are all important trade-off parameters affected by fan design. High through-flow velocity, low

TABLE IV. BASELINE ENGINE CORE COMPRESSOR AND STAGE CHARACTERISTICS

Airflow	2.83 kg/sec (6.24 lb/sec)	Adiabatic efficiency	82 percent
Corrected airflow	2.27 kg/sec (5.00 lb/sec)	Speed	27,627 rpm
Pressure ratio	7.0	Inlet hub-tip ratio	0.78

Stage	Pressure Ratio	ΔT °K (°F)	$U^{(1)}$ mps (fps)	$\phi^{(1)}$	$\psi^{(1)}$	η (%)
1	1.323	31 (56)	288 (945)	0.480	0.38	87
2	1.323	32 (57)	280 (919)	0.465	0.41	87
3	1.296	33 (60)	272 (393)	0.450	0.45	87
4	1.253	31 (56)	264 (866)	0.450	0.45	87
5	1.218	29 (52)	256 (840)	0.450	0.45	87
6	2.13	123 (222)	374 (1228) (2)	0.450	0.90	83

(1) At mean flow-path radius

(2) Centrifugal tip speed

(3) ϕ = Flow coefficient
ψ = Compressor work coefficient
η = Efficiency

hub-tip radius ratio, and low tip work coefficient serve to decrease fan diameter and increase rotational speed. In turn, nacelle weight and drag are reduced, the diameter or number of stages and the weight of the fan-driving turbine are decreased as is the cost of these elements. Offsetting these gains are the increases in internal duct losses and an accompanying decrease in fan efficiency which reduces thrust, propulsive efficiency, and finally fuel economy. The interrelationship between these parameters and those of the fan aerodynamic design is complex, and their detailed evaluation was beyond the scope of this study. However, the intent was to provide a fan design for the baseline engine that balanced these parameters based on previous experience. Thus a reasonable and compatible set of losses, efficiencies, component sizes, and weights would be achieved. Some of the pertinent fan preliminary design parameters are given in Table V.

TABLE V. BASELINE ENGINE FAN STAGE CHARACTERISTICS

Airflow	25 kg/Sec (55 lb/sec)
Pressure ratio	1.3
Adiabatic efficiency	89 %
Speed	14,114 rpm
Inlet hub/tip ratio	0.45

Design Parameter	Units	Hub	Tip
Pressure Ratio	-	1.3	1.3
ΔT	°K(°F)	25.2 (45.3)	25.2 (45.3)
U	m/sec (fps)	159 (522)	354 (1160)
[illegible]	-	1.0	0.45
[illegible]	-	1.0	0.2025
η	%	89	89

<u>Core turbine.</u> - A high-specific-work [65,220 J/kg, (28.04 Btu/lb)] single-stage turbine was selected for the baseline engine core. Preliminary stress analysis indicated that an integrally cast, tip-shrouded stage was possible if the stage were designed with high work coefficient and substantial outlet swirl.

Although less efficient than a two-stage design, the single-stage turbine would be lighter and substantially less costly. Again, in-depth turbine design and trade-off analyses would be needed to prove the best choice. Table VI lists several design parameters for the baseline core turbine.

TABLE VI. BASELINE ENGINE CORE TURBINE CHARACTERISTICS

Specific work	65,220 J/kg (28.04 Btu/lb)
Flow	2.83 Kg/sec (6.24 lb/sec)
Pressure ratio	2.83
ΔT	242°K (436°F)
η, Efficiency	86 %
Speed	27,627 rpm
Hub/tip ratio (exit)	0.89
U, Tip speed	380 m/sec (1246 fps) (1)
ϕ, Flow coefficient	0.59 (1)
λ, Turbine work factor ($gJc_p\Delta T$)	1.97 (1)
Exit swirl angle	30 deg. (1)

(1) At mean radius

Fan-driving turbine. - The fan-driving turbine for the baseline engine is a three-stage unit close-coupled to the core turbine. It was designed with a symmetrical flow path with the mean radius equal to that of the core turbine. Each of the three stages have tip shrouds with labyrinth seals for improved efficiency. Approximately 1.5 percent turbine efficiency could be gained over the baseline by adding an interturbine diffuser duct. However, decreased velocity through the turbine would increase the turbine weight and cost. This design option further illustrates the need for trade-off analyses to ascertain the total effects on airplane size and costs. Design parameters for the baseline fan-driving turbine are given in Table VII.

TABLE VII. BASELINE ENGINE FAN-DRIVING TURBINE CHARACTERISTICS

Specific work	63730 J/kg (27.4 Btu/lb)
Flow	2.83 kg/sec (6.24 lb/sec)
Pressure ratio	2.83
ΔT	193 °K (347 °F)
η, Efficiency	87 percent
Speed	14,114 rpm
Hub/tip ratio (exit)	0.76
U, Tip speed	194 m/sec (636 fps) (1)
ϕ, Flow coefficient	1.10 (1)
λ, Turbine work factor	1.97 (1)
Exit swirl angle	0 deg. (1)

(1) At mean radius

Combustor. - The baseline engine combustor is a reverse-flow, annular configuration, sized for a heat-release rate of 1.1 million Joules/hour/atmosphere/cubic meter (3 million Btu/hour/atmosphere/cubic foot). Although the configuration has a high surface-to-volume ratio, the moderate turbine inlet temperature chosen for the cycle should permit a low turbine-inlet pattern factor and adequate liner cooling with little difficulty. The definition of a fuel-admission system has little impact on engine preliminary design or performance, since comparable performance can be expected from any of several alternate systems. However, to meet the chemical emissions requirements applicable to small civil turbofans after 1979 will probably require development of a hybrid system combining the desirable characteristics of both atomizers and vaporizers. Such a system is known by the generic term "air-blast atomizer," a form of which is depicted on the layout drawing of the baseline engine.

Exhaust ducts. - The engine exhaust ducts are conventional. They provide diffusion to 0.3 Mach number and terminate in jet nozzles that have equivalent convergence angles of 15 degrees. The planes of the jet nozzles are located sufficiently aft to provide a maximum nacelle boattail half-angle of 15 degrees.

Accessories. - The engine/airframe accessories are mounted on a gear case that is integral with the engine front frame. A radial "tower" shaft transmits power from the forward end of the core compressor shaft to the accessory gear case through bevel gear sets. The accessories systems complement consists of: a 150-ampere, 30-vdc, 12,000-rpm starter-generator; a 12,000-rpm fuel pump; an electronic/hydromechanical fuel control; a dual-igniter continuous-ignition system; a lubrication system consisting of a three-element pump, integral oil tank, and a fuel-oil heat exchanger. The gear case also has provision for an optional engine-driven hydraulic pump.

A representative engine mechanical design was executed in sufficient detail to determine component configurations, stress levels, manufacturing methods, material selections, and finally to make a credible engine weight estimate.

The baseline engine basic layout is shown in Figure 11. As stated previously, the intent of the design was to achieve a cost-effective engine by maximizing performance quality while minimizing parasitic machinery. The drawing illustrates the results of the careful adherence to these principles. The engine was sized for a sea-level static thrust of 5204 N (1170 lb). At this thrust level, the calculated engine weight is 125 kg (275 lb) including all accessories, bypass duct, and final jet nozzles. The maximum bypass duct diameter is 58 cm (23 in.), and the overall engine length from inlet flange to the plane of the primary jet nozzle is 114 cm (45 in.).

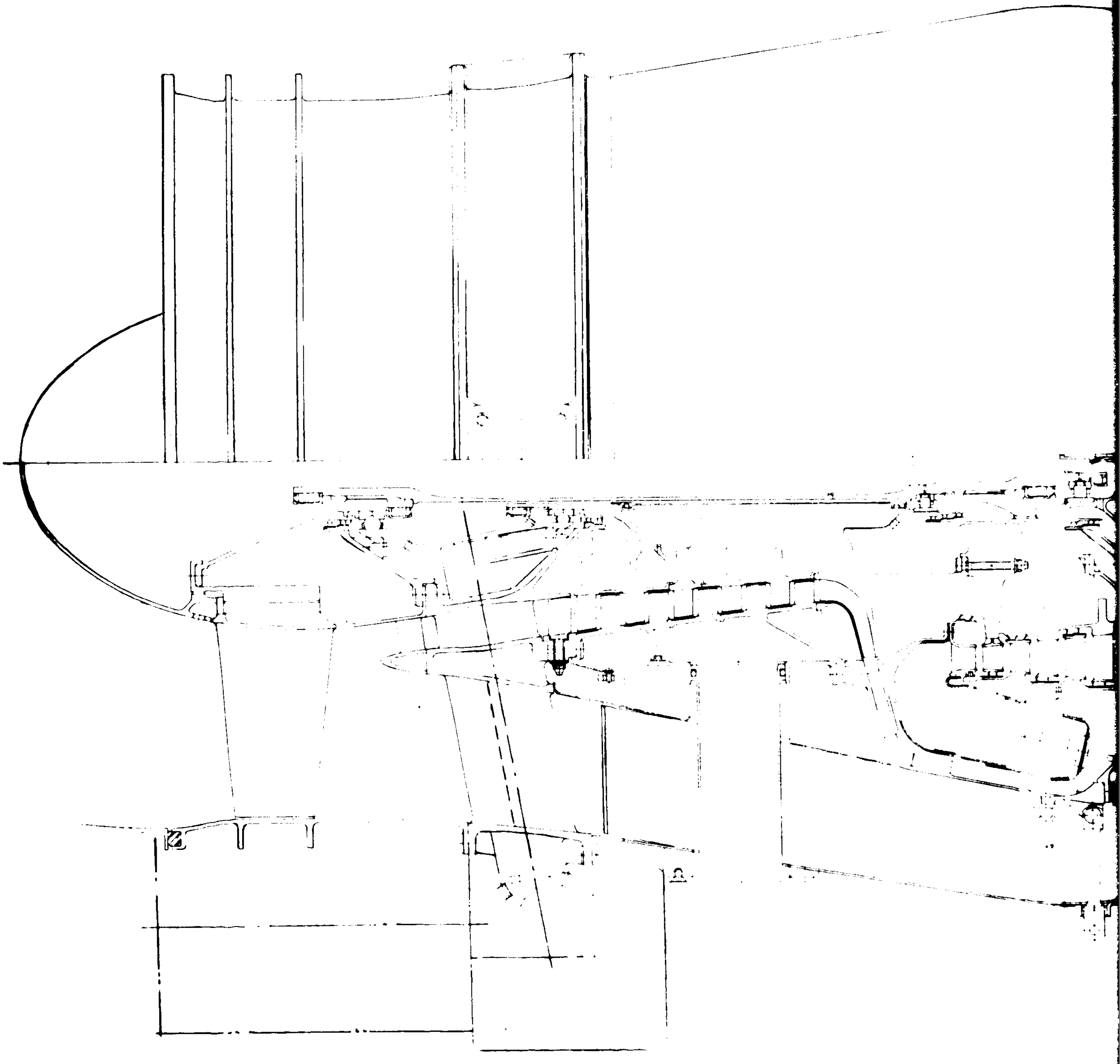
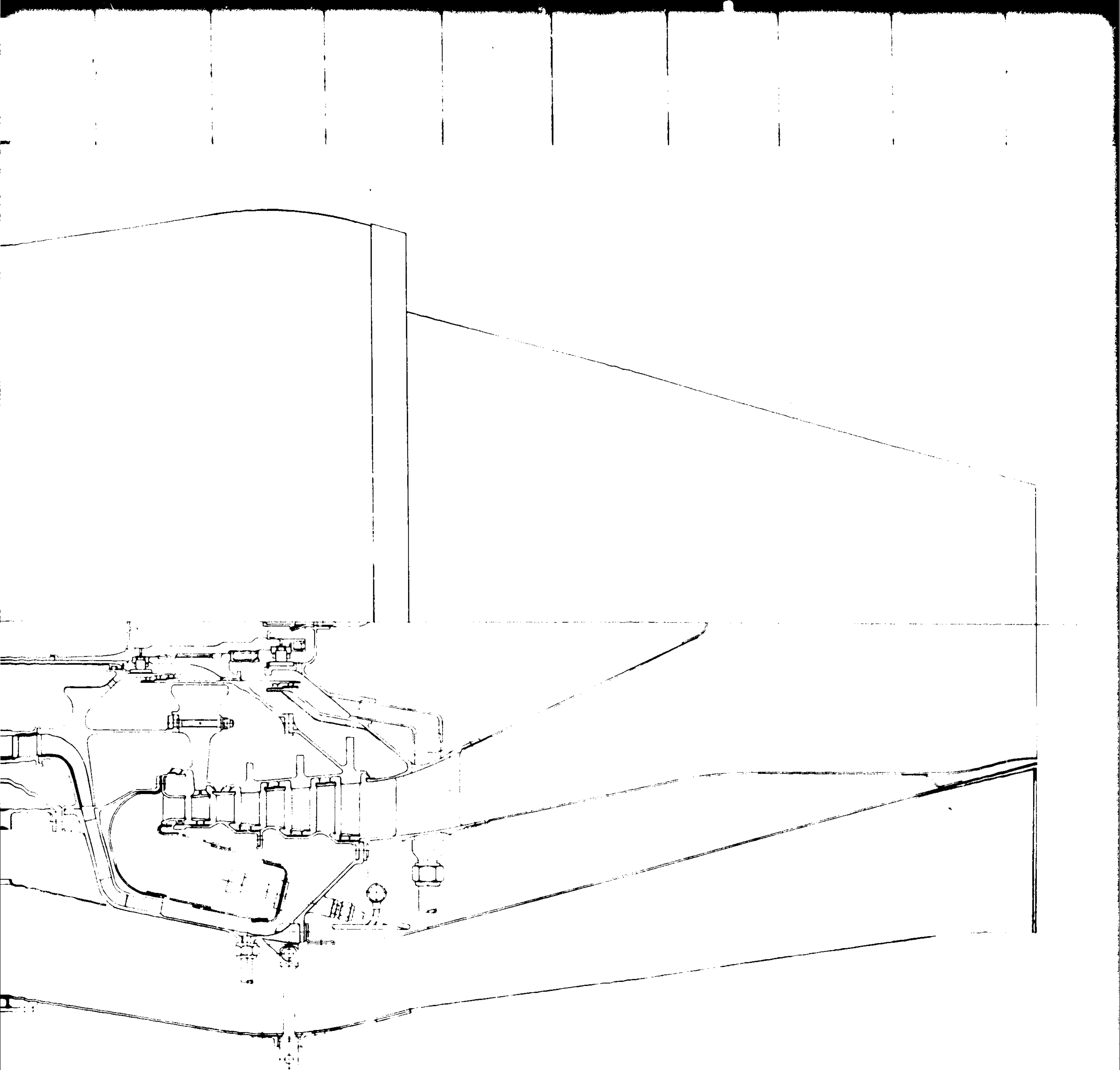
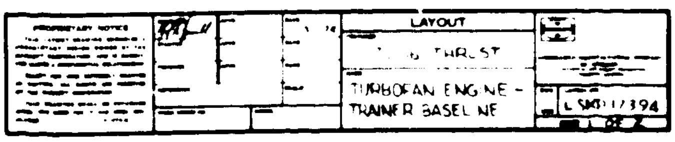


Figure 11. Baseline Engine Cross Sect

FOLDOUT FRAME 1



aseline Engine Cross Section.



FOLDOUT FRAME

From this nominal size, it was determined that the baseline engine could be scaled over the range of 3114 to 7562 N (700 to 1700 lb) thrust with negligible change in specific performance. The linear scale factor over this range of thrust varies approximately plus and minus 20 percent. This variation in scale factor was found to be sufficient to cover the thrust requirements of the various single- and twin-engine airplanes derived in the study.

[redacted]

BASELINE AIRPLANE DEFINITION

General Design Considerations

Based on a summary knowledge of the USAF primary training mission profile, early studies showed that a candidate military primary trainer would exhibit a strong resemblance in size, cost, and operating cost to a high-performance, civil, light airplane. It was learned that the typical mission consisted of flights of 1-1/2-hour duration, at 370 to 463 kilometers (200 to 250 kt) airspeed at altitudes up to 6096 m (20,000 ft) usually under visual flight rule (VFR) conditions. The Cessna T-37B (the present USAF definitive primary trainer) has elementary instrument flight rule (IFR) capability, no cockpit pressurization, and has short-range capability that is consistent with the nominal 1-1/2-hour primary training mission. However, it was designed to be fully aerobatic and capable of 648 km/hr (350 kt) airspeed. With an empty weight over 1860 kg (4100 lb) and gross weight over 2994 kg (6600 lb), the T-37B consumes fuel at the rate of 719 liters (190 gallons) per hour in typical use. If it were in production today, it would cost the USAF over $300,000.

In comparison, contemporary civil, high-performance, single-engine light airplanes carry four to six people and baggage at speeds up to 352 km/hr (190 kt), over ranges of 1296 to 1667 km (700 to 900 n.mi.), yet weigh about half as much as the T-37B, and consume fuel at a rate one-tenth as great. These airplanes, with a comprehensive set of IFR avionics, sell to consumers for less than $80,000. This includes markups to cover the cost of design, development, and commercial distribution.

This comparison illustrates anomalies that can only partly be explained by the fact that a sturdy military training airplane would weigh more and consume more fuel than a civil airplane, which was designed to less stringent standards. In fact, it can be shown that the disparities in this comparison result from the differences in overall propulsion-system efficiencies between the types compared. At the time the T-37 was designed, there were no small turbofan engines in production and the propulsive system had to be a turbojet. At the low airspeeds flown in the training syllabus, this engine cycle is very inefficient due to low propulsive efficiency. This inefficiency far out-balances the inherent light weight and low volume of turbojet engines. In addition, the state of the art in attainable aircraft gas turbine cycles is markedly improved today. With this combination of factors, the thrust specific fuel consumption (TSFC) at 463 km/hr (250 kt) airspeed of the turbofan engines is less than half the TSFC of the T37B turbojets.

The aircraft design studies described later in this report resulted in approximately a 25 percent increase in both gross weight and fuel consumption over these preliminary results.

In the work leading to the definition of engine and airplane baselines for this study, the turbofan engine was found to have overall propulsion system efficiency that compared favorably with that of contemporary light airplanes. Preliminary airplane sizing analysis of a baseline airplane showed that with performance and operational capabilities similar to those of the T-37B, but with less equipment than the Randolph study TA-2, a new turbofan-powered trainer could be designed to have a gross weight of 1270 kg (2800 lbs) and a cruise fuel consumption of 117 liters/hr (31 gal/hr) at 463 km/hr (250 kt), 4572 m (15,000 ft) altitude.

Initial configuration study. - From these initial results, a three-view drawing was prepared for a baseline single-engine airplane configuration. Military standard cockpit sizing, engine location, landing gear stowage, wing vertical location, equipment volume requirements, and the estimated center of gravity location were the major considerations in the preparation of this drawing. Crew visibility requirements, initial wing and tail size and plan form options, ground clearance angles, and seat-ejection path clearance were other items addressed in the configuration study.

Together with baseline engine performance data, the geometry of the baseline airplane configuration was input to the GASP computer program to obtain a baseline program model. Several iterations were required for a balanced solution with wing, engine, tail, and equipment locations and sizes that were representative of the three-view drawing. With this model completed, the next task was to "calibrate" the program with structural component weights that were based on actual attainments in similar airplanes. Ultimate load factor, pressurization level, and design speed requirements were input to the program weight and balance module. At this point the specific requirements chosen for the TA-2 Randolph study airplane were reviewed and input, accommodated, or achieved in successive synthesis analyses. Table VIII lists the mission and performance requirements, and Table IX lists the configuration and equipment requirements that were met.

The first three-view and synthesis results were then submitted to Cessna Aircraft Company for evaluation and comment. Equipment fit and weights, and general configuration considerations were of primary concern in Cessna's initial review.

TABLE VIII. PRIMARY TRAINER MISSION AND PERFORMANCE REQUIREMENTS ADOPTED FROM THE RANDOLPH STUDY.

	Mission
o Takeoff	10 min. idle + 5 min. MIL power
o Climb	MIL power climb to 4572 m (15,000 ft)
o Cruise	1.5 hr at 463 km/hr (250 kt) at 4572 m (15,000 ft)
o Landing	15 min. MIL power at sea level
o Reserves	20 min. loiter at sea level
	Performance
o Takeoff Ground Run	≤ 1220 m (≤ 4000 ft)
o Takeoff Time	10.15 seconds
o Landing Roll	≤ 1220 m (≤ 4000 ft)
o Approach Speed	167-204 km/hr (90-110 knots)
o Rate-of-Climb	≥ 610 m/min at 4572 m (≥ 2000 ft/min at 15,000 ft)
o Single-engine Hot-Day Takeoff Configuration	≥ 122 m/min (≥ 400 ft/min) at sea level
o Cruise Endurance	1.5 hour at 463 km/hr (250 kt) at 4572 m (15,000 ft)
o Cruise Ceiling	7620 m (25,000 ft)
o Sustained Load Factor	≥ 2.5 g's at 4572 m (15,000 ft)
o Instantaneous Load Factor	≥ 4.0 g's at 4572 m (15,000 ft)
o Maximum Speed	463 km/hr (250 kt, 0.399 Mach)

TABLE IX. CONFIGURATION AND EQUIPMENT REQUIREMENTS ADOPTED FROM THE RANDOLPH STUDY.

Configuration	
o Seating	2-place side-by-side
o Cockpit Geometry	MIL-STD-133 or equivalent
o Visibility	MIL-STD-850
o Propulsion	Two engines in flight, restart capability
o Flight Controls	Conventional primary controls, flaps, deceleration devices, lift spoilers
o Landing Gear	Retractable, nose wheel steering, antiskid brakes
Equipment	
o Avionics	Communications - UHF, hot-mike intercom
	Navigation - TACAN or VOR-DME area nav. IFF/SIF(AIMS). ILS marker beacon
	Special - Collision avoidance
o Instruments	Engine - State-of-the-art round dial
	Flight - Attitude, heading ref. system, flight director, angle of attack indicator
o Status Monitoring	Conventional light warning
o Student Performance-Measuring Equipment	Audiovideo recording system, audio tape recorder
o Air Conditioning	
o Bird-Proof Windshield	
o Windshield, Engine Inlet Anti-Ice	
o Oxygen and Pressurization	
o Zero/Zero Escape System	
o Standard Emergency System	

Personnel of the DCS Operations staff at USAF Air Training Command Headquarters were solicited for comments on the configuration and projected performance capabilities. Similarly, U.S. Navy Training Command personnel were consulted on the configuration.

Although it was the original intent of the study to define one most cost-effective airplane having a single engine applicable to civil use, it became increasingly clear during discussions with military personnel that twin-engine configurations should be included in the study. Furthermore, while side-by-side seating was selected for the initial baseline, certain advantages of the tandem seating arrangement were pointed out by military personnel, and it was subsequently decided to include this alternative. Finally, the basic configuration sensitivity study included a complement of four airplanes--single-engine and twin-engine, each with side-by-side and tandem seating.

Three-view drawings were prepared, and synthesis definition of baseline models was completed for each of the four configurations. Except for their inherent differences, these baselines were executed in such a manner that their performance and operational characteristics were nearly identical. Of course, the twin-engine airplanes required extra propulsion-related equipment and instrumentation to be included in the weight, and tandem seating required extra cockpit instrumentation.

The analysis of single-engine climb capability of the twins provided the greatest difference in the performance analysis. Federal Aviation Regulations (FAR), Part 25 were applied to the single-engine climb analysis of the twins. With respect to engine failures on the single-engine airplanes, it was considered imperative that the stall speed be less than 113 km/hr (70 mph), which would usually permit emergency landings without destruction of the aircraft or serious injury to the crew. This is apparently the intention of the 70-mph maximum stalling speed rule of FAR Part 23 that is applicable to single-engine civil airplanes.

Complete descriptions of the baseline airplanes are provided in the following sections, Tasks I through IV, together with the results of an engine sizing study and the synthesis sensitivity analyses conducted on the baseline configurations.

TASK I - EVALUATION OF SIDE-BY-SIDE, SINGLE-ENGINE TRAINER

Initial Airplane and Engine Sizing Analyses

The baseline airplane geometry was defined by a three-view drawing and a compatible set of synthesis results. The first drawing was prepared with a 9.29 sq m (100 sq ft) wing, anticipating that the "solution" airplane would weigh about 1067 kg (3500 lb), resulting in a wing-loading approximately the same as that of the T-37B. The 10-aspect-ratio wing was located in a "shoulder" configuration, at eye level, behind the cockpit. Wing-tip fuel tanks were sized to accommodate more than half the total fuel, thus providing a large wing relieving load that would ensure a light wing structure. However, the tip tanks were removed in subsequent design analyses when Cessna Aircraft Company advised that, in a fully aerobatic trainer, it is considered imperative to minimize the moment of inertia about the roll axis to enhance recovery from spins. Synthesis analysis was performed with the assumption of full span, 100 percent Fowler-action flaps, and the drawing incorporated Mitsubishi-type spoilers for roll control. Two vertical tails were located at the tips of the horizontal tail, and tail surfaces were sized to provide volume coefficients of 0.075 and 1.36, for the vertical and horizontal, respectively. The engine was located on the top of the fuselage tail cone, with the rectangular inlet located at about 50 percent of the wing root chord. The landing gear was assumed to be a conventional tricycle configuration. The oleo-spring nose gear retracted forward into the fuselage nose, and the main gear, with spring steel struts, retracted about a single pivot hinge into the fuselage tail cone. Crew accommodation was provided within a fully glazed canopy, similar in size and shape to that of the T-37B.

Because the GASP mission module is based on a typical general aviation mission format of takeoff, climb, and cruise, plus reserve fuel, it was necessary to rationalize the Randolph TA-2 airplane mission into this format. Based on initial synthesis results and separate calculations, it was determined that a mission nearly equivalent in fuel consumption to that of the TA-2 could be represented by takeoff, climb to 4572 m (15,000 ft), 250 kt cruise for 740 km (400 n.mi.), plus reserve fuel for 45 minutes at that speed and altitude. All further synthesis sizing work done in the program was based on this mission.

With baseline engine and airplane sizing results obtained from initial synthesis analysis, the required airplane performance envelope was examined for engine sizing criteria other than those analyzed in GASP. The TA-2 airplane requirements specified in the Randolph UPT study report were adopted for this investigation. Initial synthesis results showed that when the engine was sized by the thrust required at start-of-cruise it would provide adequate takeoff and climb performance. However, a separate analysis of the thrust required to provide a sustained load factor (or maneuver rate) of +2.5 g's at 4572 m (15,000 ft) showed that this performance requirement probably sized the engine at all reasonable values of wing-loading. Because the twin-engine airplane to be evaluated in Task II was to be the subject of extensive wing-loading and thrust-loading studies, it was decided to forego this work on the single-engine airplane until optimized values were obtained in Task II.

The calibrating values of airplane component weights for refining the GASP calculations were taken from the 1724-kg (3800-lb) Cessna Turbo Centurion II. From these calibrating values GASP has the capability of calculating new weights based on the different structural load criteria, pressurization requirements, and component sizes. For example, the GASP-calculated wing weight was based on a 10-g ultimate load factor, resulting in a substantially heavier wing than that of the more lightly loaded Centurion. Fixed equipment weights were estimated by Cessna Aircraft Company, based on the Randolph "fit" and currently available equipment lists. The breakdown of equipment weights is given in Table X.

Final Design results and Evaluation

Sensitivity and trade-off analysis results obtained in Task II were incorporated in a final side-by-side, single-engine design. The three-view drawing, Figure 12, and GASP printouts supplied in Appendix B describe this design in detail. The most notable aspect of these design results are the large reductions in airplane size and fuel consumption over those obtained in the Randolph study. The reduction in empty weight amounted to over 227 kg (500 lb); in gross weight, about 363 kg (800 lb); and in mission fuel, about 30 percent. This achievement is attributed to the use of a "best" engine, which is designed and optimized to meet the stipulated mission and airplane performance requirements.

Compared with the other three configurations evaluated in this study, the side-by-side single-engine trainer is unquestionably the most cost-effective. With side-by-side seating, there is a minimum of duplication of cockpit instrumentation and equipment. With a less costly single-engine installation (nacelle, instruments, etc.), and lower specific engine cost (dollars per unit of thrust), the development and flyaway costs of this configuration should be substantially less than those of the others. With the lowest fuel consumption, lower maintenance costs permitted by the single engines and least amount of equipment, the airplane operating costs should also be lower by a significant margin. Finally, with the easier operation inherent in single-engine airplanes, training effectiveness should be improved by eliminating the requirement for teaching "primary" students the more complex multi-engine piloting tasks.

TABLE X. CESSNA-PROVIDED FIXED EQUIPMENT WEIGHT ESTIMATE

	kg	lb
INSTRUMENTS:		
Engine Instruments, Transmitters (0.5 of Citation)	6.8	15.0
Cabin Pressure Instruments	1.0	2.1
Flight Instruments, Dual + Dual Flight Director [11 kg (25 lbs) ea]	42.9	94.5
Angle-of Attack (Citation)	4.1	9.1
Accelerometer (T-37)	0.7	1.5
VGH Recorder (T-37)	1.1	2.5
Total	56.6	124.7
ELECTRICAL (Except Starter-Generator)		
Battery - 22 amp-hr, 24-volt, & case	26	58
Solid-State Inverters (2)	4	9
Cutouts and Voltage Reg. (0.5 of T-37)	3	6
Switches, Rheostats, Panels, Boxes	2	4
Circuit Breakers	0.5	1
Junctions, Distribution Boxes	1	3
Plugs	1	3
Relays	2	4
Wiring	9	20
Conduit	0.5	1
Miscellaneous	4	8
Lights (Incl. Strobes), Horns	10	22
Supports	3	7
Total	66	146
FURNISHINGS & EQUIPMENT		
Rocket Zero-Zero Ejection Seats with Chutes	50	110
Cushions	2	5
Oxygen System (High-Pressure, Limited Size, for Decompression Only)	9	19
Pins, Plates, Mirror, Rugs, Trim Insulation, First Aid	5	12
Fire Detect. & Extinguish. & Portable Extinguisher	8	18
Ventilation System	7	15
Heat System	0.5	1
Cooling System (Air Cycle)	25	56
Defog & Windshield Anti-Ice & Rain Removal (All Weather)	5	10
Auxiliary Gear	1	2
Total	112.5	248

TABLE X. (Contd.)

AVIONICS	Side-by-Side		Tandem	
	kg	lb	kg	lb
UHF	4	9	5	12
Hot Mike Intercom & Audio System	2	5	3	6
VOR-ILS-MB	3.2	7.1	5	12
DME	8	18	9	20
Area NAV	5	11	7	15
IFF/SIF (AIMS)	3.4	7.5	3.4	7.5
Collision Avoidance (Proximity Only)	3	7	4	8
Audiovideo Recording System and Audio Tape Recorder	18	40	18	40
Sub Total	46.6	104.6	54.4	120.5
10% For Installation	4.7	10.5	5.4	12.1
TOTAL	51.3	115.1	59.8	132.6

PERFORMANCE

CRUISE SPEED (15,000 ft ALT.)	250 kts	(288 mph - 0.399 Mn)
STALL SPEED (FULL FLAPS)	~60 kts	(57.6 mph)
RANGE (WITH 2 CREW)	~700 nm	(+45 min RESERVE)
TAKEOFF DISTANCE (S.L. STD DAY)	~1564 ft	(OVER 35 ft)
LANDING DISTANCE (S.L. STD DAY)	~1665 ft	(OVER 50 ft)
SUSTAINED MANEUVER RATE	2.5 g	(AT 200 kt/15,000 ft)
MAX RATE OF CLIMB (S.L. - 15,000 ft)	3800-2660 fpm	

WEIGHTS

GROSS	3497 lb
EMPTY	2453 lb
OPERATING (INC ONE CREW)	2663 lb
FIXED USEFUL LOAD	210 lb
PAYLOAD	200 lb
FUEL (MAX)	635 lb (97 gal)
FIXED EQUIPMENT GROUP	669 lb
FLIGHT CONTROLS GROUP	76 lb
STRUCTURES GROUP	1246 lb
PROPULSION GROUP	459 lb

DIMENSIONS & AREAS

FUSELAGE	
LENGTH	26.65 ft
WIDTH	4.33 ft
HEIGHT	4.50 ft
WING	
AREA	77.7 sq ft
SPAN	27.9 ft
GEOMETRIC MEAN CHORD	2.81 ft
ASPECT RATIO	10
TAPER RATIO	0.50
THICKNES/CHORD (ROOT - TIP)	0.17 - 0.17
HORIZONTAL TAIL	
AREA	18.5 sq ft
SPAN	9.19 ft
CHORD	2.08 ft
ASPECT RATIO	4.57
THICKNESS/CHORD	0.07
VERTICAL TAIL (TWIN)	
AREA (TOTAL)	12.2 sq ft
SPAN (EACH)	3.02 ft
CHORD (EACH)	2.05 ft
ASPECT RATIO	1.50
THICKNESS/CHORD	0.087
ENGINE NACELLE	
LENGTH	4.5 ft
AVERAGE HEIGHT	1.67 ft
LANDING GEAR	
MAIN TIRE SIZE	6.00×6 in
NOSE TIRE SIZE	5.00×5 in
WHEEL BASE	8.75 ft
WHEEL TREAD	8.0 ft

ENGINE DATA

THRUST (SLS STD DAY)	1509 lb
AIRFLOW ( )	7.57 lb/sec
CRUISE THRUST (250 kt / 15,000 ft)	326 lb
WEIGHT DRY	322 lb

AERODYNAMIC DATA

WING LOADING	45 lb/sq
CRUISE DRAG POLAR	C_D = 0.0271
LANDING GEAR C_D INCREMENT	0.02816
EFFECTIVE FLAT PLATE AREA	2.108 sq ft
WETTED AREA	494.5 sq ft
MEAN SKIN FRICTION COEFFICIENT	0.00426
CRUISE REYNOLDS NO. / FOOT	1.843 × 10^6
C_L MAX (WITH FULL FLAPS)	3.67
HORIZONTAL TAIL VOLUME COEFFICIENT	1.123
VERTICAL TAIL VOLUME COEFFICIENT	0.075

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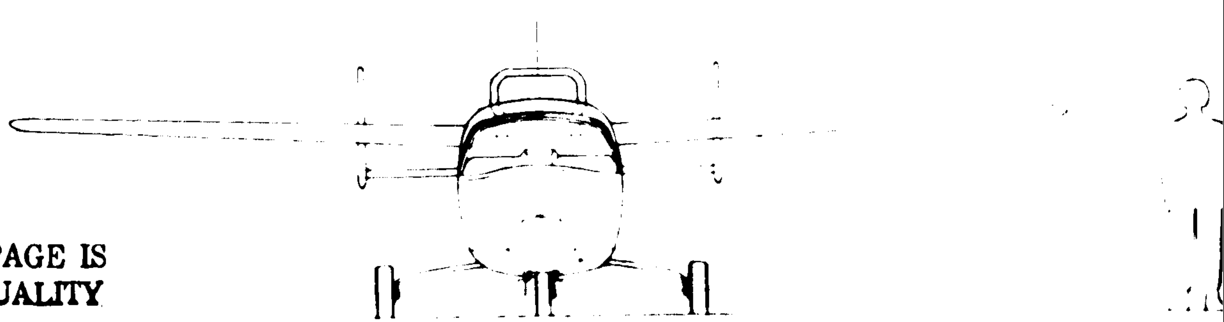


FOLDOUT FRAME

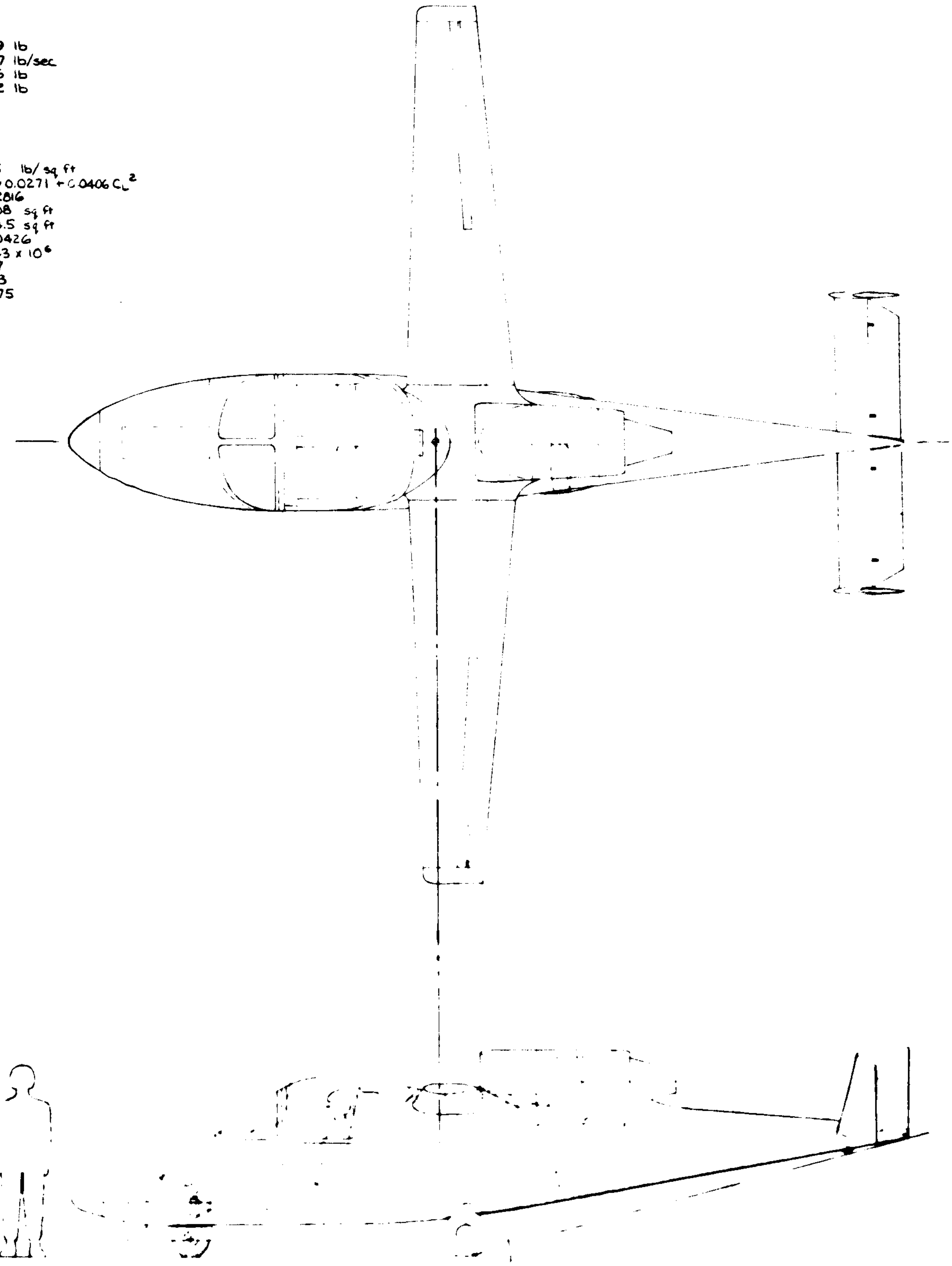
Figure 12. Three-View Drawing Side-By-Side, Single En

DATA

RUST (SLS STD DAY)	1509 lb
FLOW (" ")	75.7 lb/sec
SE THRUST (250 kt/15,000 ft)	326 lb
GHT DRY	322 lb

NAMIC DATA

G LOADING	45 lb/sq ft
SE DRAG POLAR	$C_D = 0.0271 + 0.0406\,C_L^2$
DING GEAR C_D INCREMENT	0.02816
ECTIVE FLAT PLATE AREA	2.108 sq ft
TED AREA	494.5 sq ft
N SKIN FRICTION COEFFICIENT	0.00426
SE REYNOLDS NO./FOOT	1.843×10^6
AX (WITH FULL FLAPS)	3.67
ZONTAL TAIL VOLUME COEFFICIENT	1.123
TICAL TAIL VOLUME COEFFICIENT	0.075



LAYOUT
SINGLE ENGINE SIDE BY SIDE SEATS
MILITARY TRAINER
THREE-VIEW

ide-By-Side, Single Engine Configuration.

TASK II - EVALUATION OF SIDE-BY-SIDE TWIN-ENGINE TRAINER

Initial Airplane and Engine Sizing Analyses

Wing-loading and thrust-loading. - Initial design efforts on the side-by-side, twin engine configuration were directed toward achieving a compatible set of results between the GASP output and the three-view drawing. The significant differences between this configuration and that of the single-engine airplane is the low versus shoulder wing location, and the relocation of the engines to the top of the wing. This engine position was chosen for two reasons. First, wing-mounted engines provide the lowest airframe structural weight. Heavy fuselage frames are eliminated, and wing bending moments are reduced by moving the "dead" weight of the engines outboard on the wing. This is particularly significant in a wing stressed for 10 g's ultimate load factor, as the case is for trainer airplanes. The second reason for wing-mounted engines is the elimination of the aerodynamic effects of the engine-to-fuselage pylon. In the deep-stall flight condition, the horizontal tail is less likely to be blanked by the combined wake of fuselage, pylons, and engine nacelles. With the consequent increased effectiveness of stabilizer and elevator, it should be possible to have a horizontal tail of shorter span and lower weight.

The original GASP definition of the airplane was performed with engines arbitrarily oversized to ensure conformance to the several airplane off-design performance requirements stipulated in the Randolph study. Because the interrelationships between wing loading (gross weight - wing area), thrust-loading (gross weight - installed thrust). and performance requirements are complex, an in-depth parametric analysis was undertaken. The results of this analysis facilitates the identification of the most cost-effective combination of these important design variables.

Figures 13 through 16 are plots of thrust available and thrusts required for 1 and 2.5 g's flight at 4572 m (15,000 ft), versus flight speed, for four values of wing-loading. The engine is "sized" in each case by the point of tangency between the thrust available and the thrust required for 2.5 g's load factor. With the engine "sized" by this method, the solution airplane thrust-loading is a fallout, varying with wing-loading. In further GASP analysis it was shown that with engines sized for 2.5 g's sustained maneuver capability at 4572 m (15,000 ft), all other performance requirements were met at all the values of wing-loading that were examined. Additional information

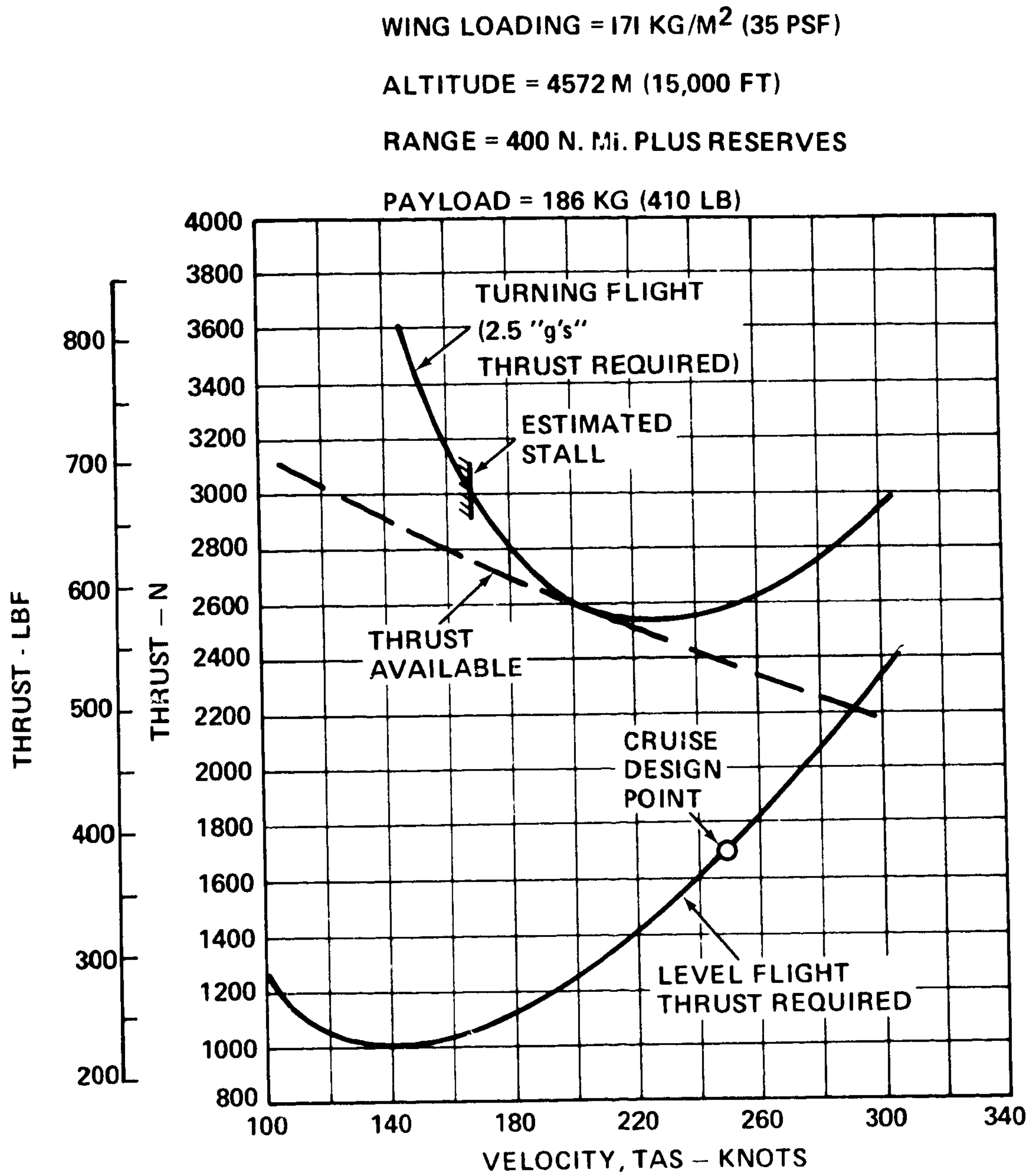


Figure 13. Engine Sizing Results at Wing Loading of 171 kg/sq m (35 lb/sq ft).

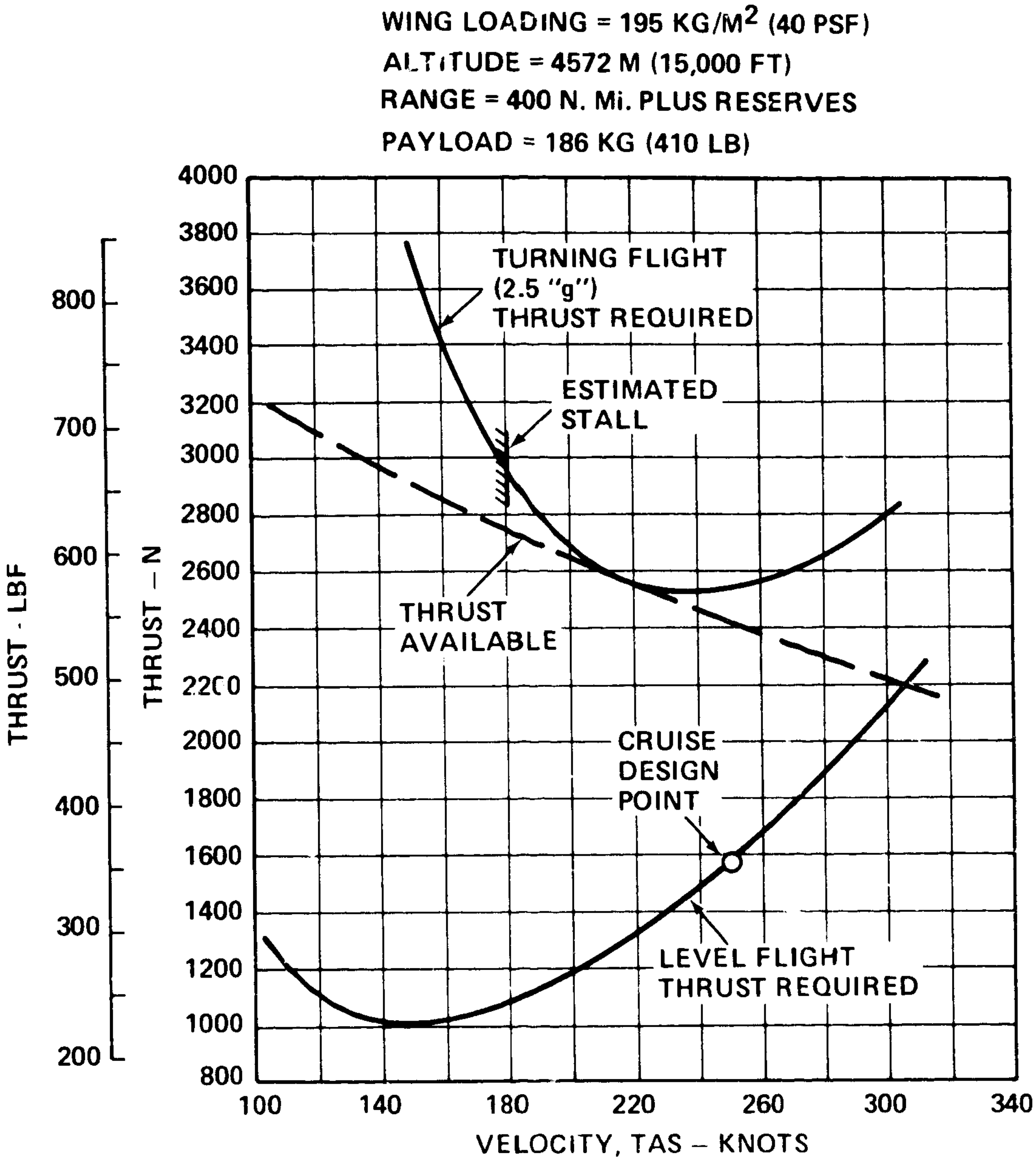


Figure 14. Engine Sizing Results at Wing Loading of 195 kg/sq m (40 lb/sq ft).

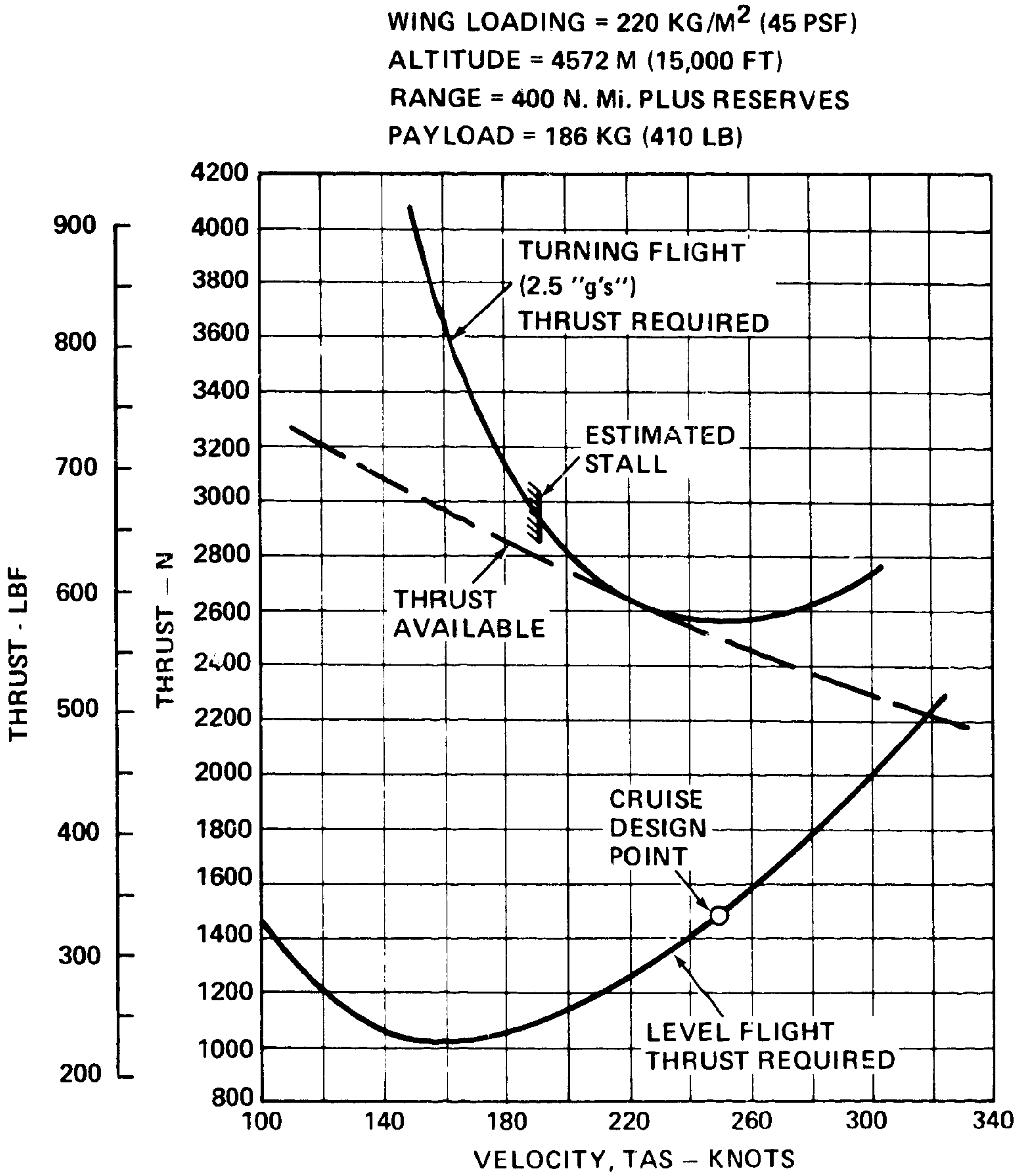


Figure 15. Engine Sizing Results at Wing Loading of 220 kg/sq m (45 lb/sq ft).

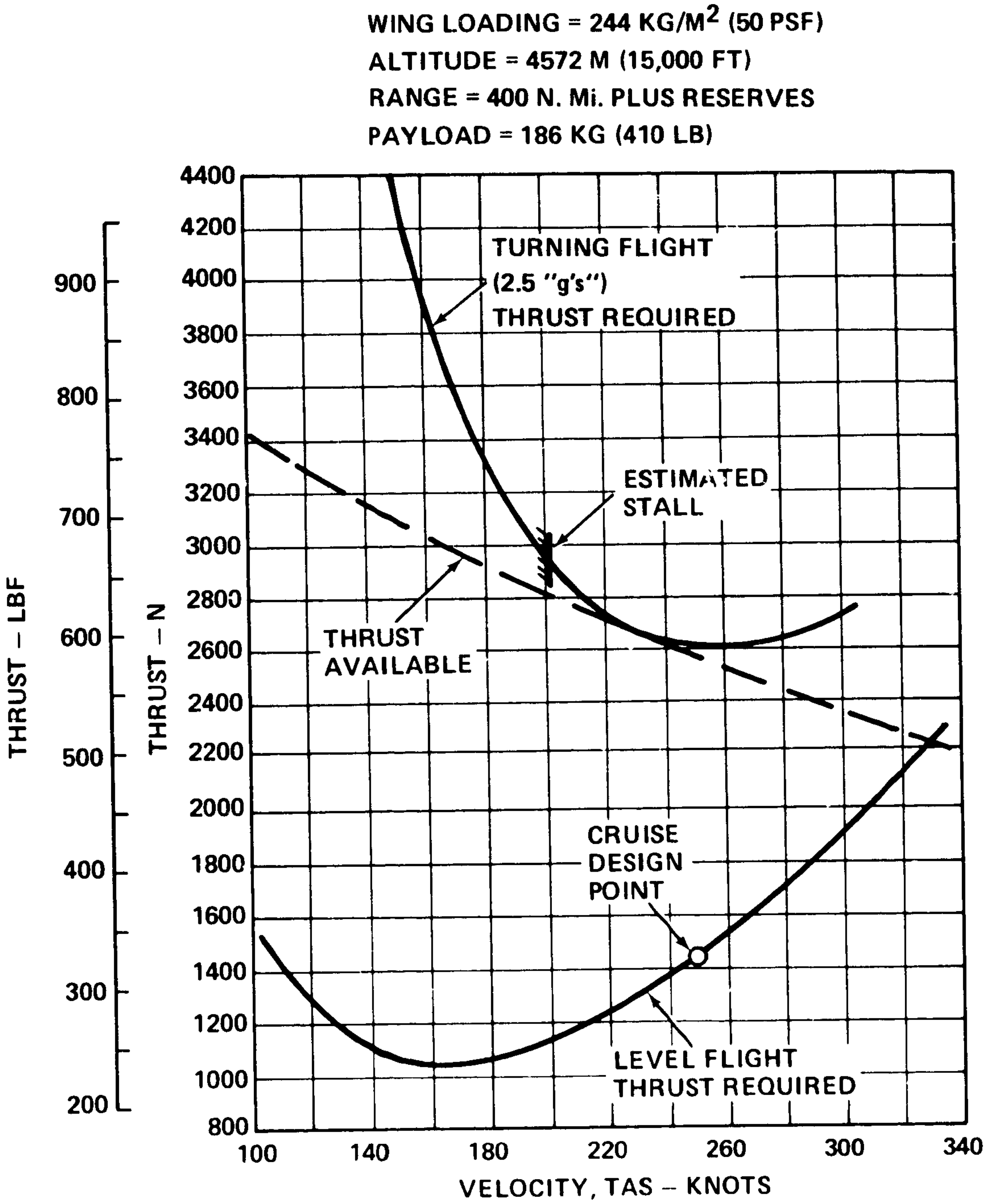


Figure 16. Engine Sizing Results at Wing Loading of 244 kg/sq m (50 lb/sq ft).

contained in Figures 13 through 16 includes the estimated stall or buffet limit velocity in a 2.5-g maneuver, best maneuver speed at 4572 m (15,000 ft), and the thrust required in level flight at the cruise design point, 463 km/hr (250 kt).

Continuing the wing-loading study, GASP results were used to generate the data contained in Figures 17 through 21. Here, the important design parameters affecting airplane unit cost and operating cost are plotted against wing loading. Three curves in each plot represent different engine sizing criteria; 2.5 and 2.0 g's sustained load factor, and 122 m/min (400 ft/min) single-engine, hot-day rate of climb. It should be reiterated here that with use of the GASP analysis technique, any point along these curves represents a unique airplane "solution" that meets all the stipulated performance, mission, and equipment requirements.

Examining Figures 17 through 21 reveals that over the range of wing loading investigated, 171 to 244 kg/sq m (35 to 50 lb/sq ft), airplane gross weight, empty weight, and cruise fuel consumption are reduced with increased wing loading. Only thrust loading and, consequently, engine sea level static thrust increases with increased wing loading. The immediate conclusion based on these results is that only life-cycle cost analyses would permit the selection of a wing-loading value that would maximize airplane cost-effectiveness. However, based on a preliminary evaluation of the relative values of the components of life-cycle cost, the highest wing loading examined in this study would result in the most cost-effective trainer. It was estimated that over the range studied the increased cost of the engine due to higher thrust would be approximately offset by lower airframe cost due to lower structural weight, and that the reduced fuel consumption would provide a net reduction in the life-cycle cost.

Coincident with the analytical efforts to select a best wing-loading for the baseline airplanes of this study, NASA personnel were engaged in the design of a computer program to simulate the dynamics of the conventional loop maneuver. Using this program, it was demonstrated that at 220 kg/sq m (45 lb/sq ft) the airplane would exhibit several desirable characteristics in the loop. Figure 22 illustrates that with a buffet limit lift coefficient of 1.12 and a 3.5-g maximum load factor, it would maintain 1.03 positive g's over the top, or apex, of the loop. Less than 1219 m (4000 ft) of vertical and horizontal air space would be required to accomplish the maneuver. This maneuvering capability, together with considerations of stall speed, landing speed and distance, and absolute wing dimensions on the "solution" airplane, led to the selection of a wing-loading of 222 kg/sq m (45 lb/sq ft). This value was used in all subsequent sensitivity analyses performed in the study.

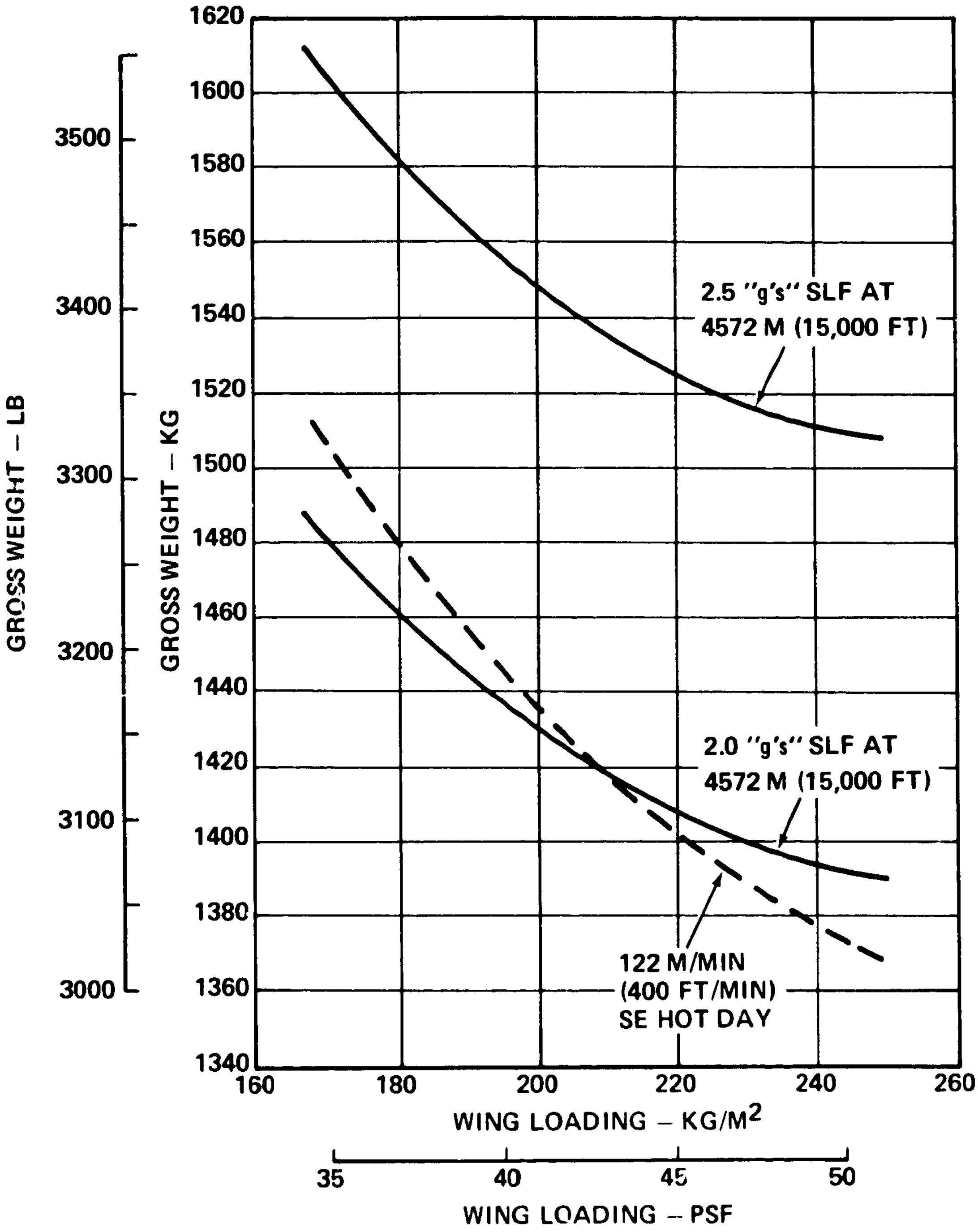


Figure 17. Gross Weight Variation With Wing-Loading.

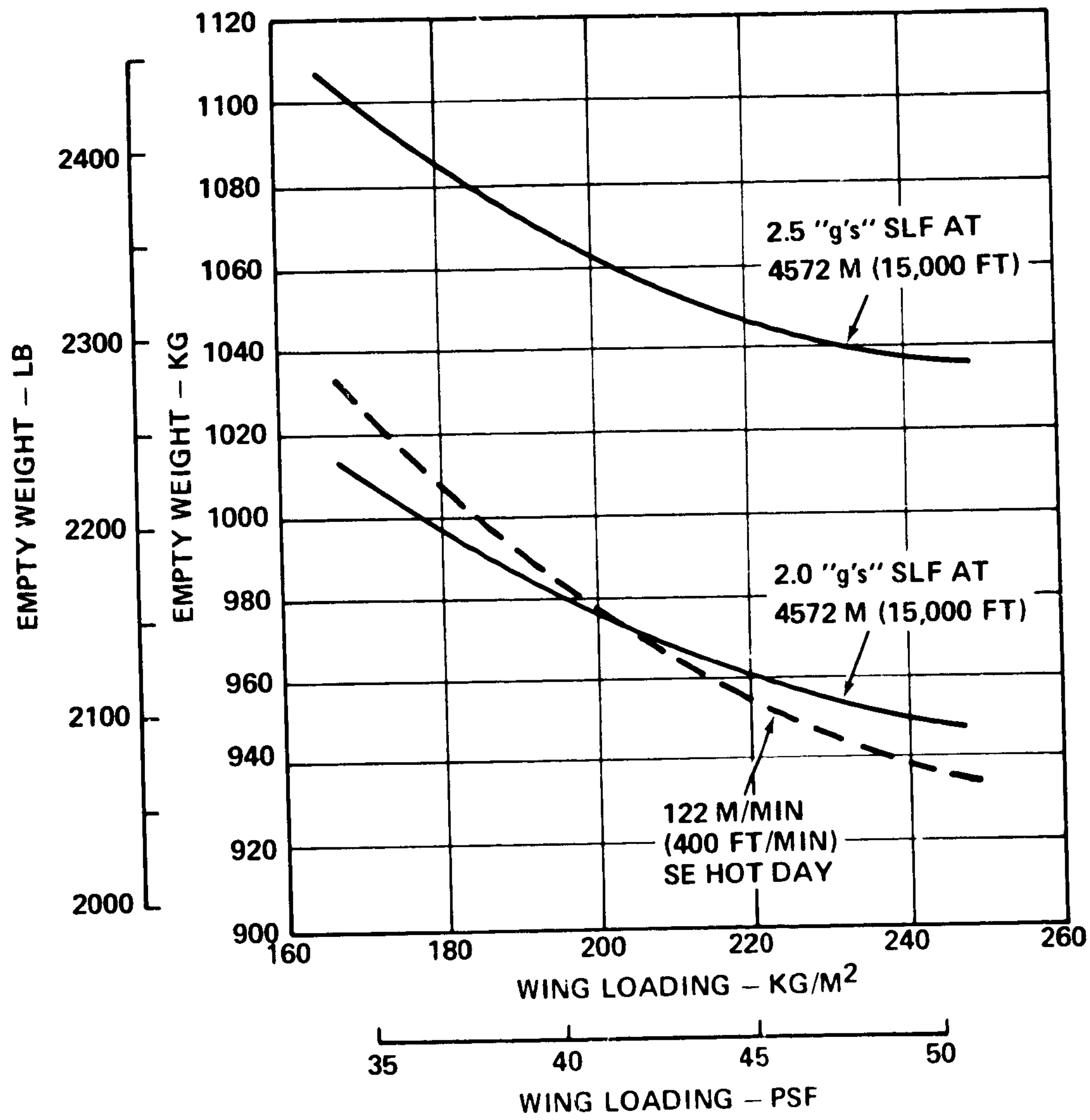


Figure 18. Empty Weight Variation With Wing-Loading.

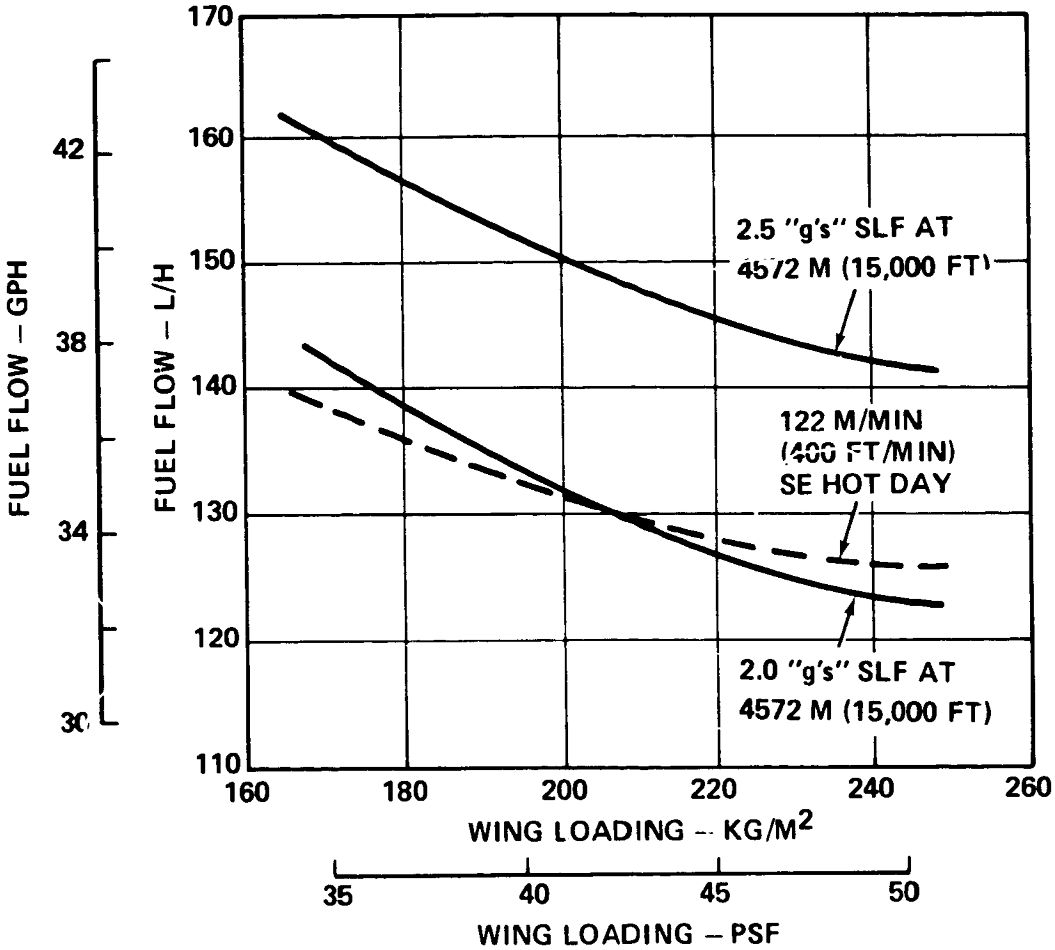


Figure 19. Cruise Fuel Consumption Variation With Wing-Loading

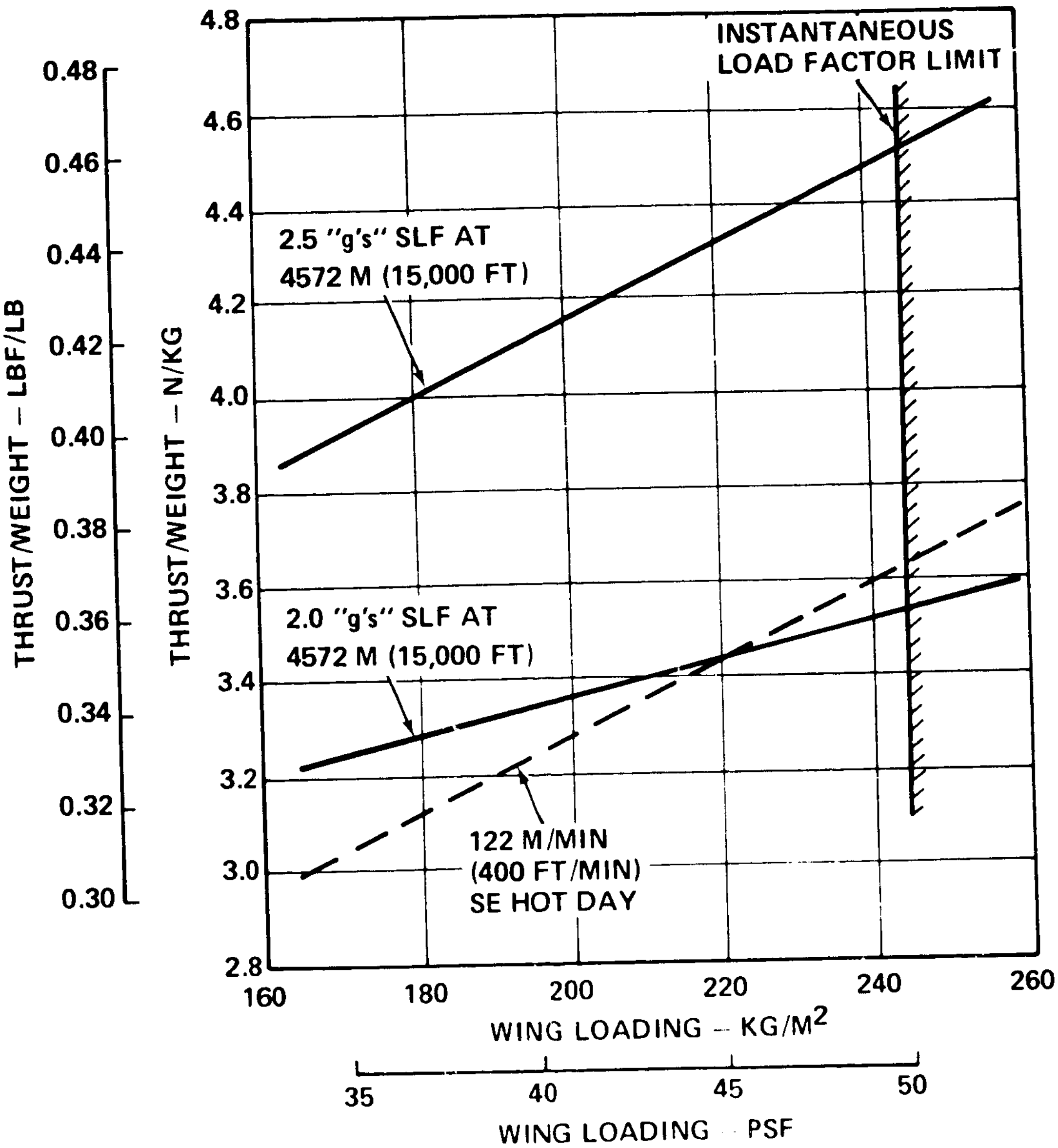


Figure 20. Takeoff Thrust-to-Weight Ratio Variation With Wing-Loading.

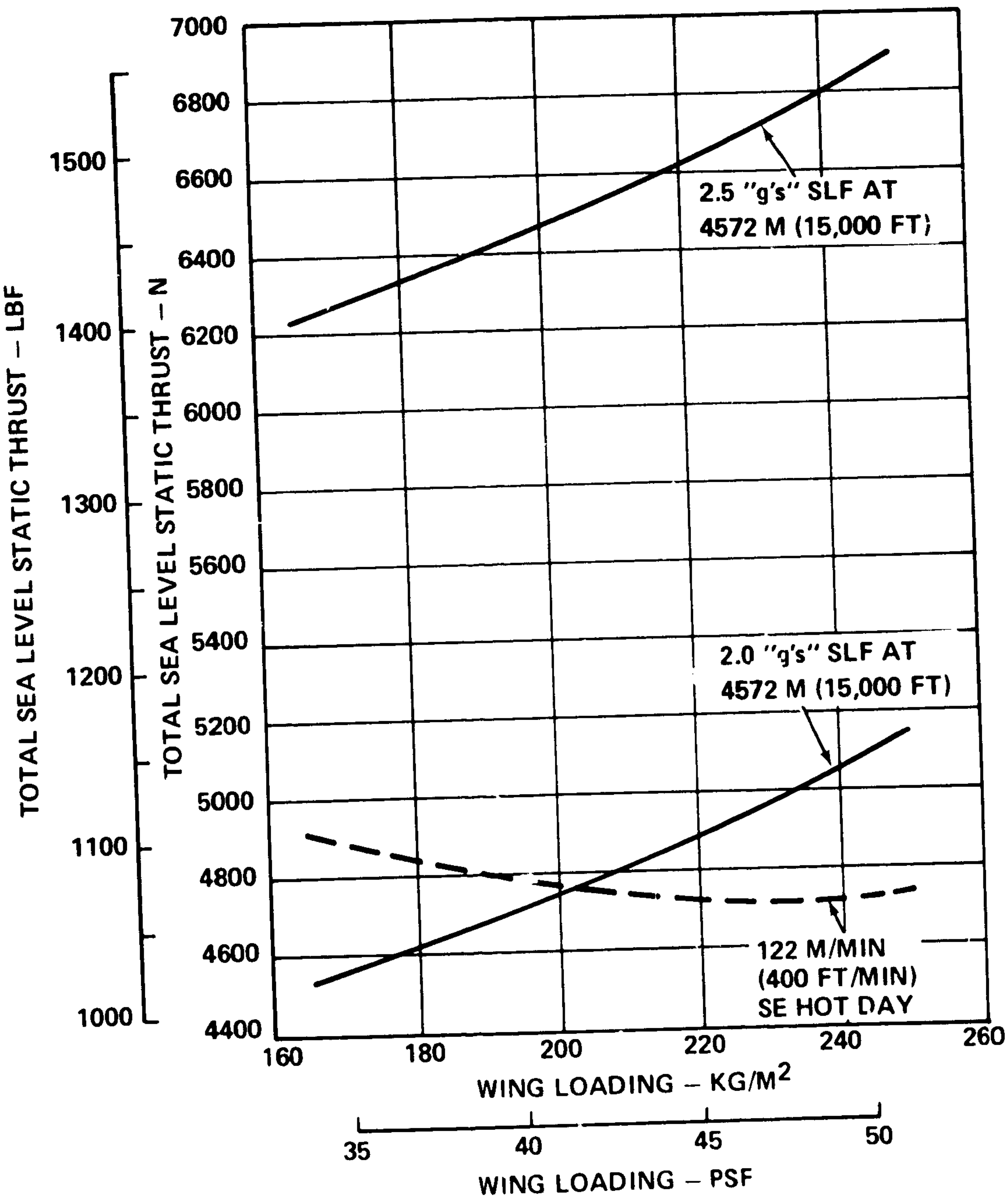


Figure 21. Total Sea Level Thrust Variation With Wing-Loading.

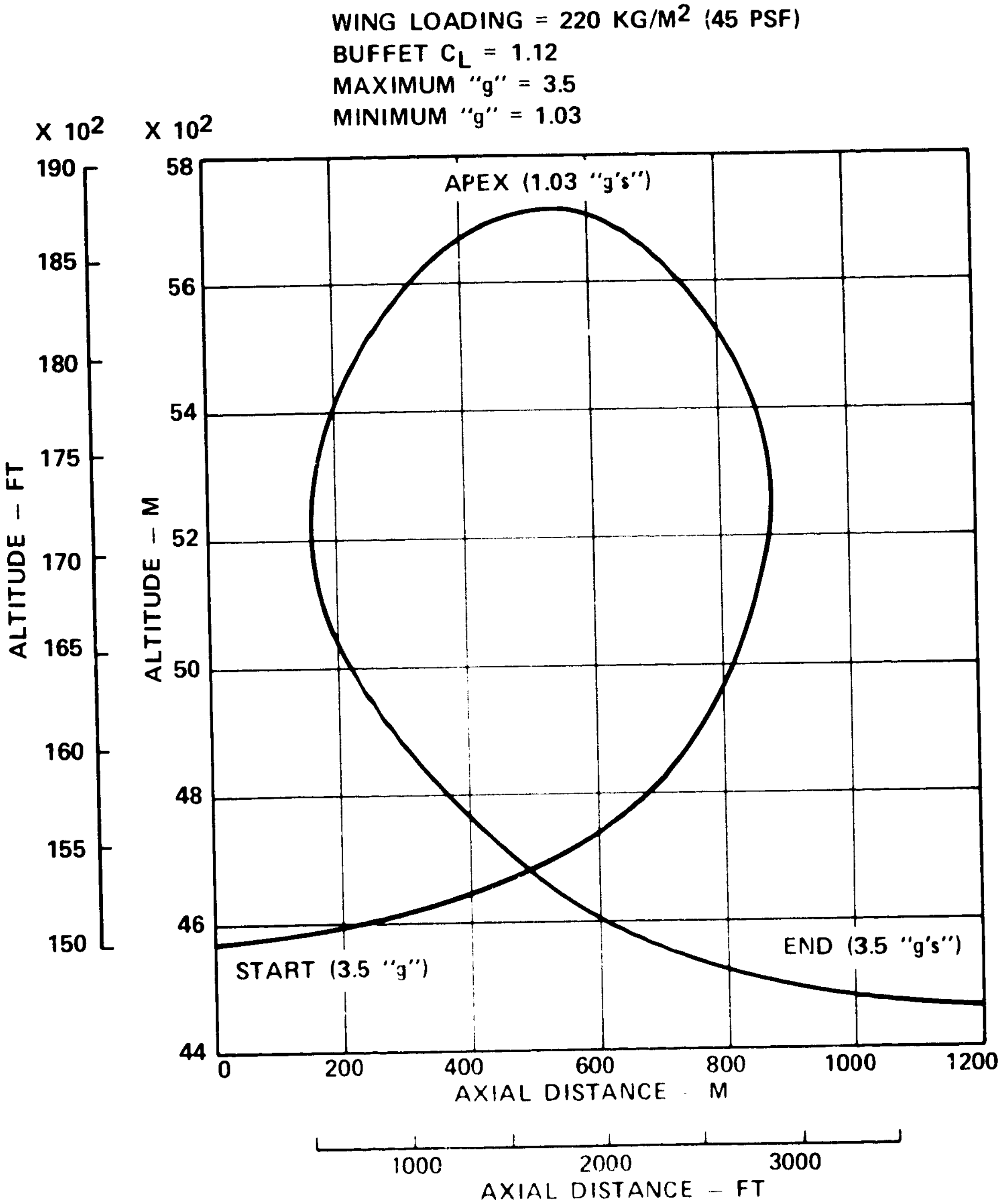


Figure 22. Loop Maneuver Characteristic of Wing-Loading of 220 kg/sq m (45 lb sq ft).

Final Design Results and Evaluation

Figure 23 illustrates climbing characteristics, which are functions of wing- and thrust-loading common to the four baseline airplanes. Time to climb to the assumed normal operation altitude of 4572 m (15,000 ft) is less than 5 minutes. Maximum rate of climb at this altitude exceeds the Randolph study requirement by about 152 m/min (500 ft/min). A service ceiling of about 10,060 m (33,000 ft) is indicated.

The operating envelope common to the four baseline airplanes is illustrated in Figure 24. In this plot of altitude versus flight speed, the maximum level flight speed, the speed for maximum rate of climb. the maximum altitude of 10,670 m (35,000 ft), and the maximum level speed of 598 km/hr (323 kt) are all shown.

With completion of the foregoing performance analyses of the baseline side-by-side twin configuration, it was concluded that the resulting airplane conceptual design was entirely responsive to the requirements defined in the Randolph study. The final design and performance quality varied only slightly from that of the Randolph conceptual TA-2. However, comparative analysis suggests that substantial gains had been made in areas that affect economics. The gross and empty weights were reduced over 20 percent, a factor that clearly indicates a potential for reducing flyaway cost from the Randolph study estimate of $319,000. Fuel consumption was estimated to have been reduced between 30 and 50 percent, indicating that operating cost could be lowered from the $99 per hour Randolph study estimate.

The three-view drawing presented in Figure 25 depicts the configuration details of the final side-by-side twin engine design. The GASP printouts given in Appendix B list additional dimensional, weight breakdown, and aerodynamic data. With use of these data, pertinent comparisons can be drawn between the design results and those of the current USAF primary trainer, the T-37B. For example, it is dimensionally smaller, with 85 percent of the fuselage length, and 80 percent of the wing span of T-37B. It is substantially lighter in weight--about half the gross weight and 60 percent of the empty weight of T-37B. With 30 percent of the fuel carried by T-37B, it has greater range and endurance in primary training missions. With cockpit pressurization and a superior avionics fit, it is able to perform training missions and meet training syllabus requirements that are not possible in the T-37B, as it is currently equipped.

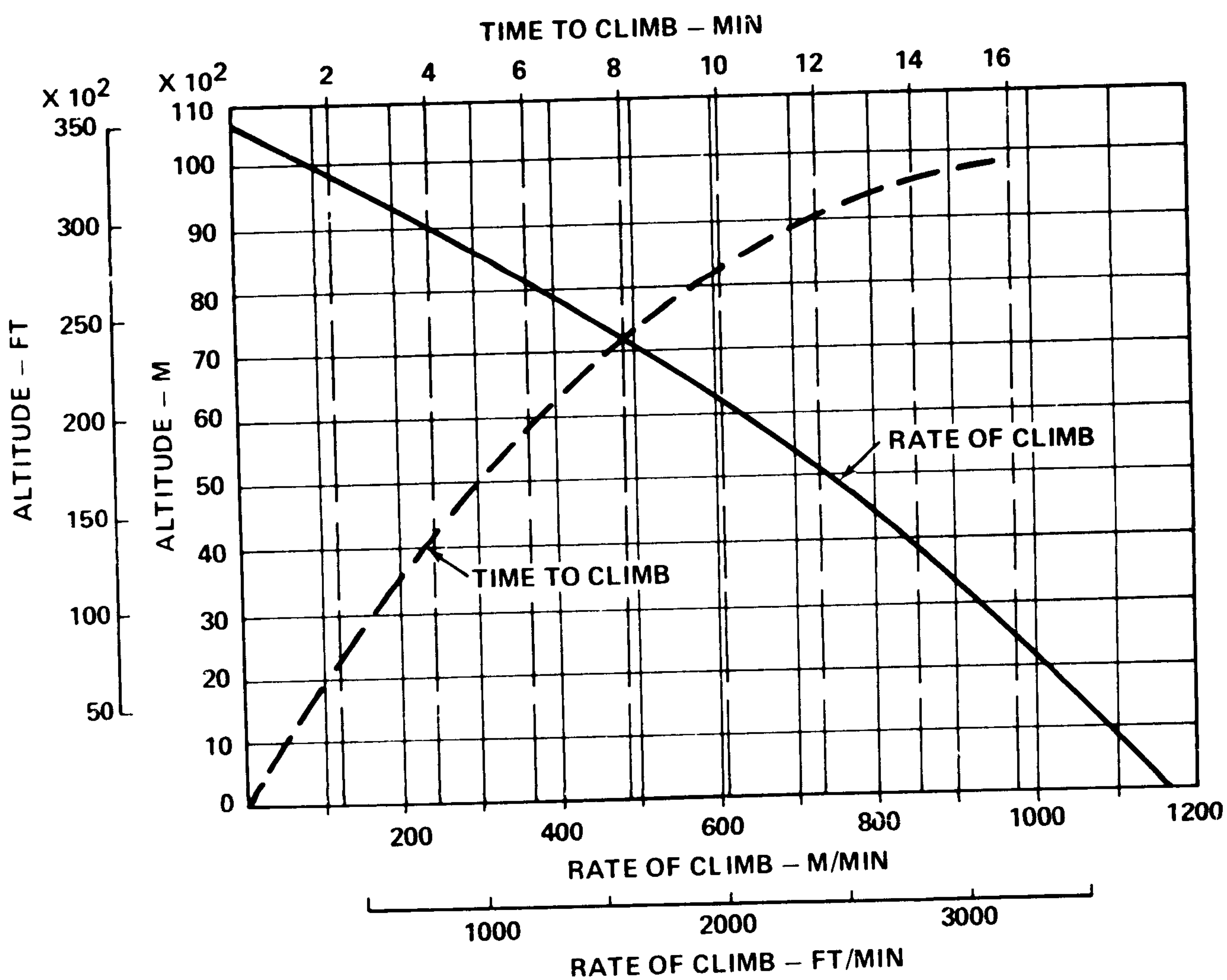


Figure 23. Airplane Climbing Characteristics at Wing-Loading of 220 kg/sq m (45 lb/sq ft).

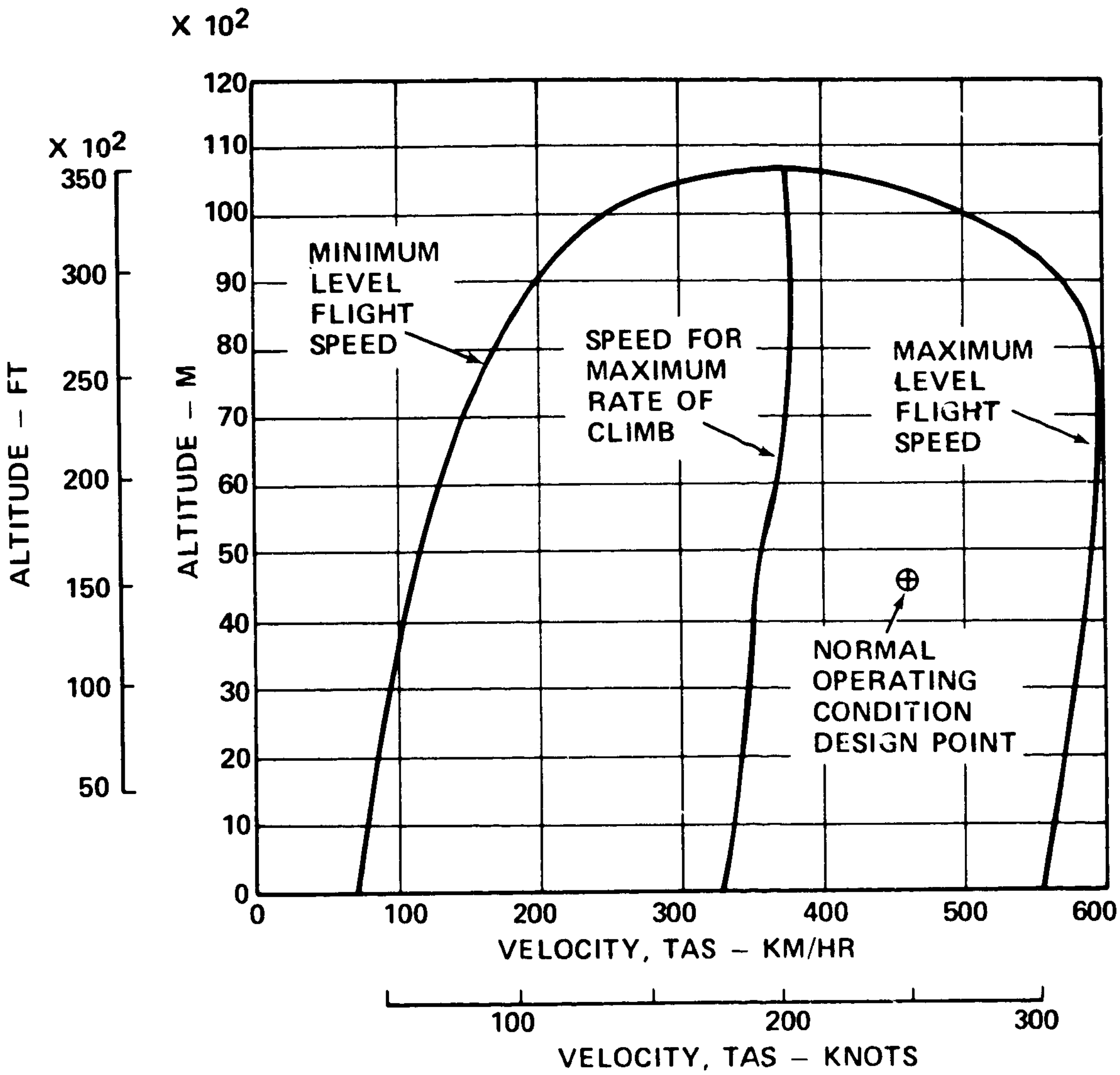


Figure 24. Airplane Performance Envelope at Wing-Loading of 220 kg/sq m (45 lb/sq ft).

Further comparisons show additional improvements in propulsion-related factors affecting operational use. It is estimated that the turbofan-powered trainer would have approximately 20 EPNdB lower noise level in takeoff (flyover), approach, and sideline measurements than the T-37B. With an exhaust jet velocity about 40 percent of that of T-37B, personnel safety on conjested flight lines would be improved. Chemical exhaust emissions would be reduced to newly proposed federal standards. Finally, with adherence to contemporary engine design practice, substantial improvements could be expected in reliability and maintainability. Modular engine component assemblies would facilitate on-the-wing, on-condition maintenance and, thereby, effect reductions in hourly operating costs.

The foregoing comparative evaluations of the side-by-side twin-engine configuration show that it is markedly superior to both the T-37B and the Randolph study TA-2 conceptual design. Only the side-by-side single-engine airplane defined in this study would have greater cost-effectiveness than this configuration.

Parametric Sensitivity Analyses

Engine definition for sensitivity analysis. - The baseline engine design described previously provided the basis for design point parametric cycle analysis, and for subsequent definition of candidate engines used to evaluate the effects of cycle quality on the airplane size and fuel consumption. In the parametric cycle analysis, all cycle variables affecting specific thrust and specific fuel consumption were examined. Fan pressure ratio, core compressor pressure ratio, turbine inlet temperature, and bypass ratio were varied over a sufficient range of values to permit the evaluation of performance trends and optimum conditions. Component efficiencies and cycle losses were varied or held constant appropriate to the cycle and component variations.

The effects on cycle quality of core compressor pressure ratio were examined by varying the pressure ratio from 4:1 to 10:1. The variation expected in adiabatic efficiency over this range is shown in Figure 26. In order to simplify the definition of compressor performance over this range, one axial-centrifugal compressor design was assumed, with design characteristics similar to those of the original baseline engine. Beginning with a 1.818 pressure ratio centrifugal compressor design, three to seven axial stages were added sequentially to the front to effect the desired overall pressure ratio.

The effects of fan pressure ratio were examined over the range from 1.15:1 to 1.4:1, with design point efficiency varying as shown in Figure 27. Turbine inlet temperatures ranging from

PERFORMANCE

CRUISE SPEED (15,000 ft ALT)	250	kts (283 mph · 0.399 M)
STALL SPEED (FULL FLAPS)	~60	kts (59.44 mph)
RANGE (WITH 2 CREW)	~700	n.m (+45 min RESERVE)
TAKEOFF DISTANCE (SL STD DAY)	~1564	ft (OVER 35 ft)
LANDING DISTANCE (S.L. STD DAY)	~1676	ft (OVER 50 ft)
SUSTAINED MANEUVER RATE	2.5	g (AT 200 kt/15,000 ft)
MAX RATE OF CLIMB (SL-15,000)	3806-2667	fpm

WEIGHTS

GROSS	3366	lb
EMPTY	2312	lb
OPERATING (INC ONE CREW)	2522	lb
FIXED USEFUL LOAD	210	lb
PAYLOAD	200	lb
FUEL (MAX)	644	lb (96 gal)
FIXED EQUIPMENT GROUP	669	lb
FLIGHT CONTROLS GROUP	78	lb
STRUCTURES GROUP	1110	lb
PROPULSION GROUP	455	lb

DIMENSIONS & AREAS

FUSELAGE		
LENGTH	25	ft
WIDTH	4.5	ft
HEIGHT	4.5	ft
WING		
AREA	75	sq ft
SPAN	27.3	ft
GEOMETRIC MEAN CHORD	2.84	ft
ASPECT RATIO	10	-
TAPER RATIO	0.5	-
THICKNESS/CHORD (ROOT & TIP)	0.17	-
HORIZONTAL TAIL		
AREA	21.3	sq ft
SPAN	9.87	ft
MEAN CHORD	2.24	ft
ASPECT RATIO	4.57	-
THICKNESS/CHORD	0.07	-
VERTICAL TAIL		
AREA	15.5	sq ft
SPAN	5.33	ft
MEAN CHORD	2.9	ft
ASPECT RATIO	1.84	-
THICKNESS/CHORD	0.075	-
ENGINE NACELLE (2)		
MEAN DIAMETER	1.67	ft
LENGTH	3.25	ft
LANDING GEAR		
MAIN TIRE SIZE	4.4 x 18	
NOSE TIRE SIZE	4.4 x 14	
WHEEL BASE	7.5	ft
WHEEL TREAD	8.33	ft

ENGINE DATA (2 ea)

THRUST (SLS STD DAY)	74
AIRFLOW ( )	38.1
CRUISE THRUST (250 kts/15,000)	16
WEIGHT (DRY)	15

AERODYNAMIC DATA

WING LOADING
CRUISE DRAG POLAR
LANDING GEAR C_D INCREMENT
EFFECTIVE FLAT PLATE AREA
WETTED AREA
MEAN SKIN FRICTION COEFFICIENT
CRUISE REYNOLDS NO. / FOOT
C_L MAX (WITH FULL FLAPS)
HORIZONTAL TAIL VOLUME COEFFICIENT
VERTICAL TAIL VOLUME COEFFICIENT

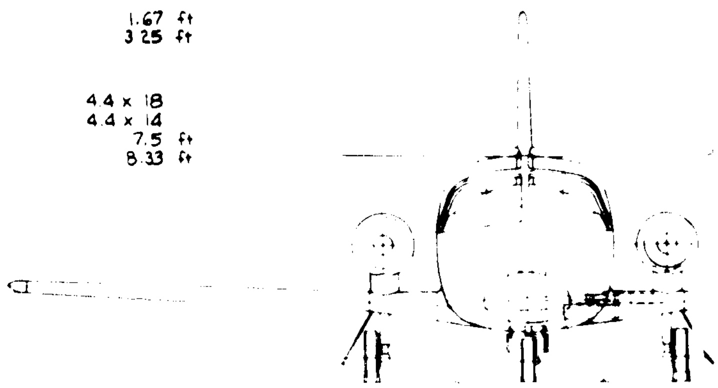


Figure 25. Three-View Drawing Side-By-Side Twin E

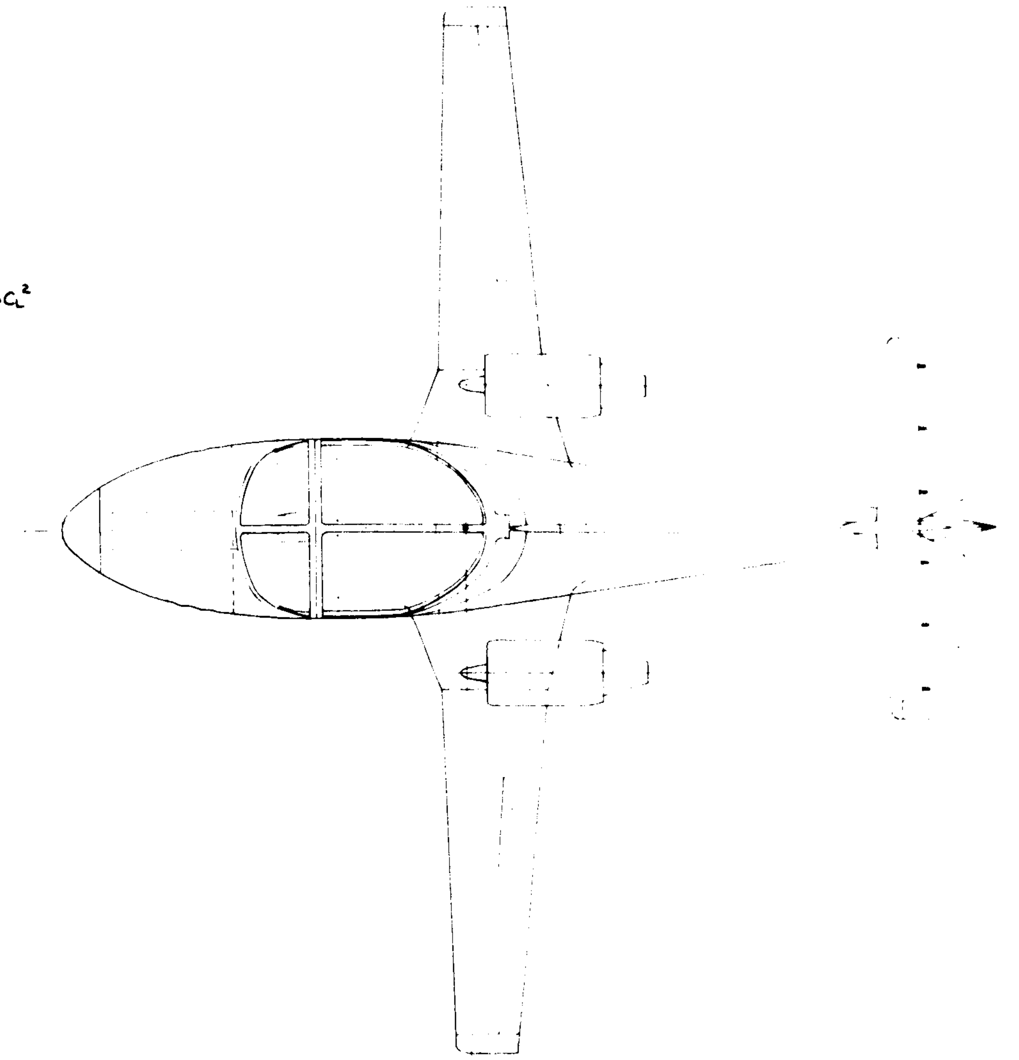
FOLDOUT FRAME

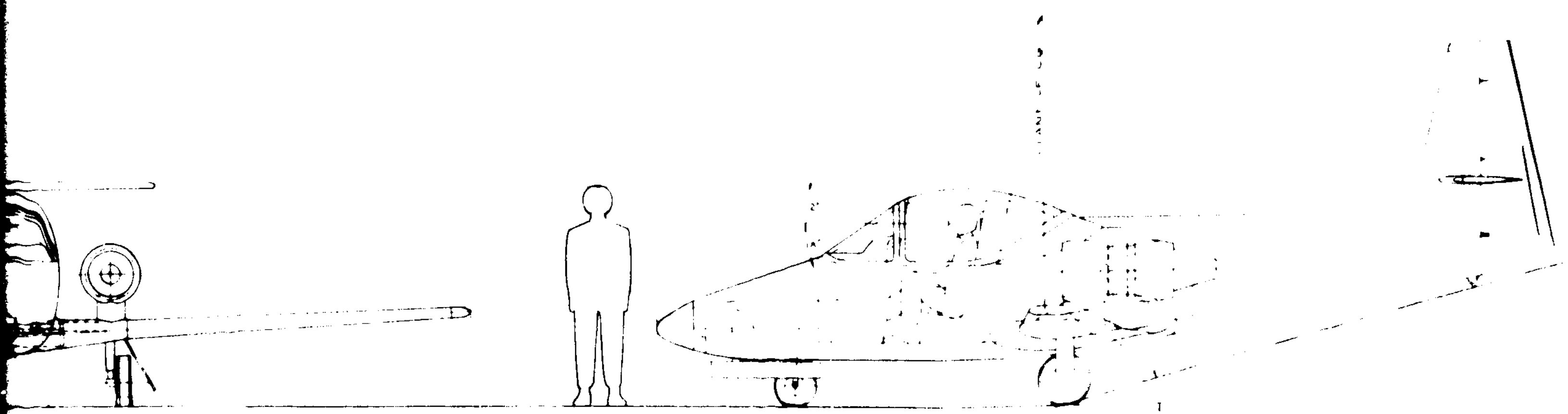
NGINE DATA (2 ea)

THRUST (SLS STD DAY)	746 lb
AIRFLOW (" " ")	38.18 lb/sec
CRUISE THRUST (250 kts/15,000)	167 lb (REQD)
WEIGHT (DRY)	159 lb

ERODYNAMIC DATA

WING LOADING	45 lb/sq ft
CRUISE DRAG POLAR	$C_D = 0.0291 + 0.0406 C_L^2$
LANDING GEAR C_D INCREMENT	0.02838
EFFECTIVE FLAT PLATE AREA	2.18 sq ft
WETTED AREA	498 sq ft
MEAN SKIN FRICTION COEFFICIENT	0.00438
CRUISE REYNOLDS NO./FOOT	1.843×10^6
C_L MAX (WITH FULL FLAPS)	3.75
HORIZONTAL TAIL VOLUME COEFFICIENT	1.123
VERTICAL TAIL VOLUME COEFFICIENT	0.094





wing Side-By-Side Twin Engine Configuration.

PROPRIETARY NOTICE

LAYOUT

FOLDOUT FRAME 2

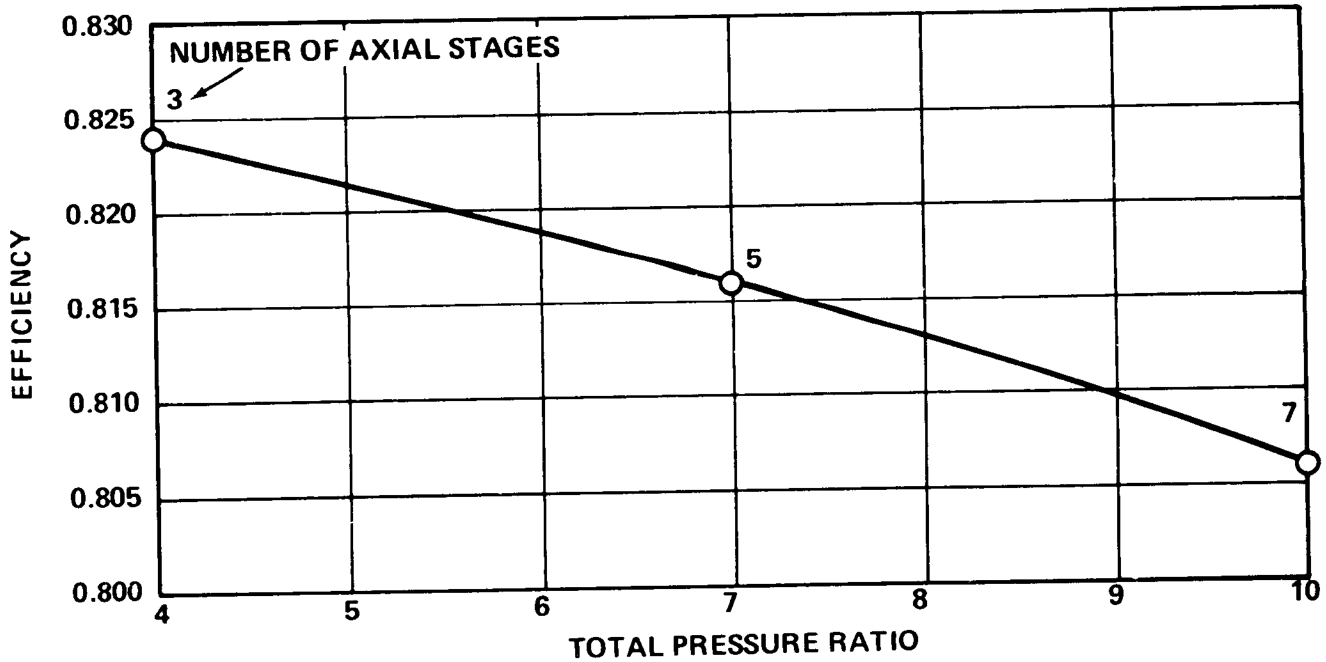


Figure 26. Estimated Core Compressor Efficiency Variation With Compressor Pressure Ratio.

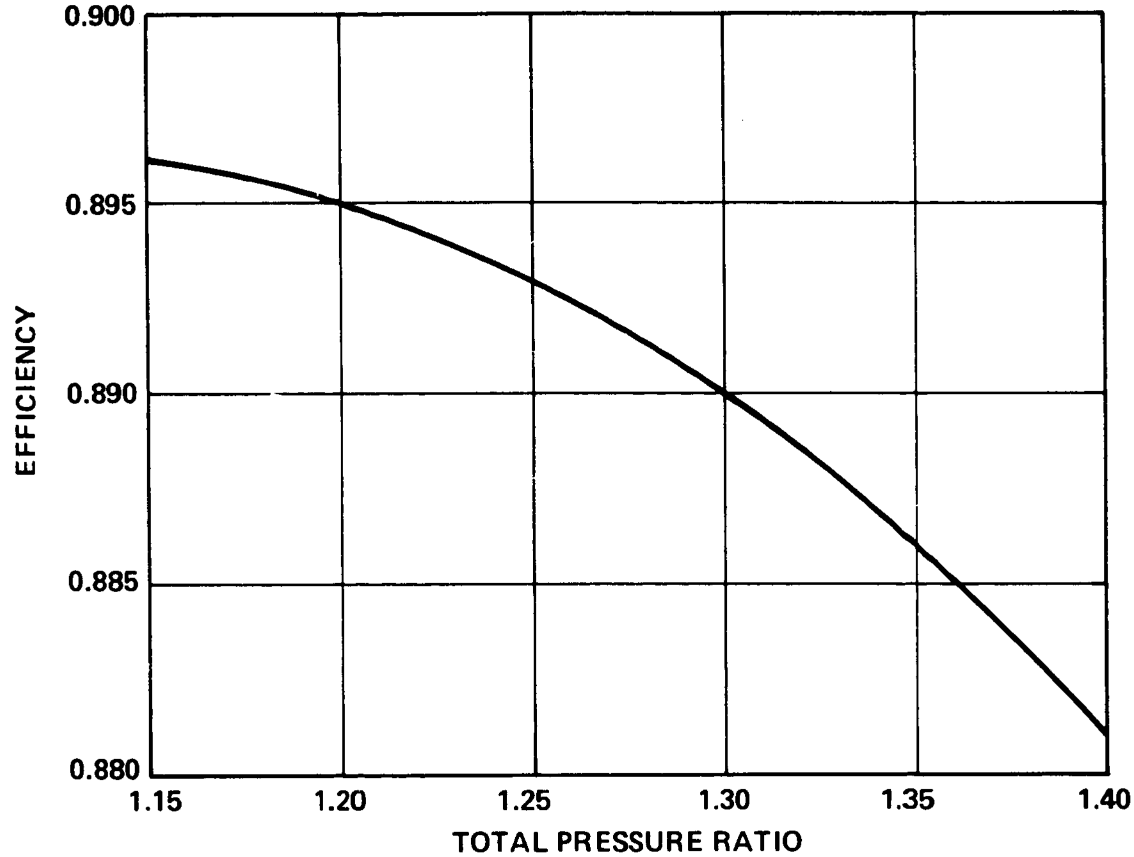


Figure 27. Estimated Design Point Fan Efficiency Variation With Fan Pressure Ratio.

1140 to 1228°K (1600 to 1750°F) were evaluated and bypass ratios were varied sufficiently to ascertain the values for minimum specific fuel consumption.

The other cycle parameters assumed to be constant were:

Combustion efficiency	0.99
Core turbine efficiency	0.852
Fan turbine efficiency	0.89
Total system pressure loss	8.5 percent

The system pressure loss is divided between the combustor (4 percent), the bypass duct, fan-compressor transition duct, inter-turbine transition and turbine exhaust ducts (1.5 percent each). The parasitic power losses in the engine (fuel and oil, windage, bearing seal, and gear friction) are assumed to require 1 percent of the power developed by the fan turbine.

The engine cycles were derived and compared on the basis of installed performance, with the following airframe installation factors accounted for:

Inlet ram pressure recovery	0.99
Power extraction	7.5 kw (10 hp)
Bleed-air extraction	4.5 kg/min (10 lb/min)

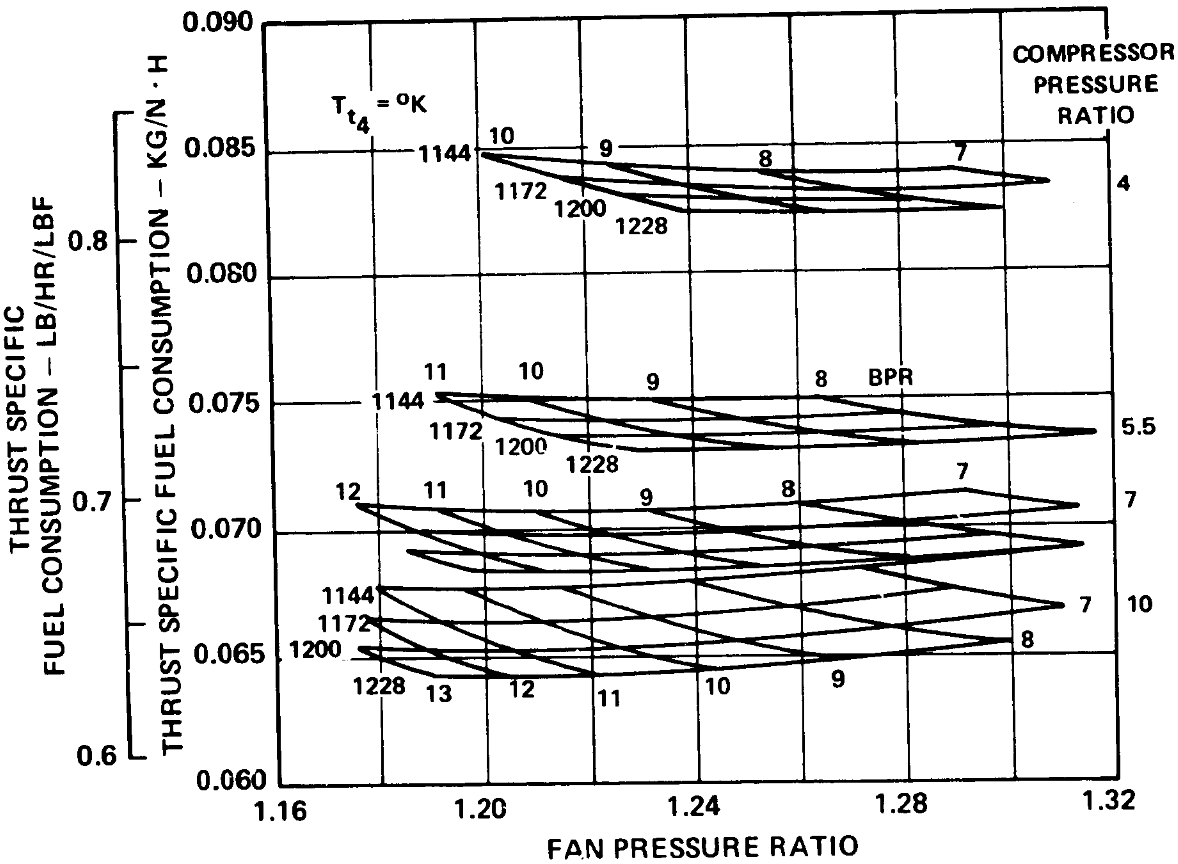
The parametric engines were sized to provide an installed net thrust of 2464 N (554 lb) at 463 km/hr (250 kt) and 4572 m (15,000 ft) ISA. This thrust level is approximately that required by the single-engine airplane to provide the 2.5 g's sustained load factor capability.

The many possible combinations of fan pressure ratio, compressor pressure ratio, turbine inlet temperature and bypass ratio are shown in the following parametric cycle analysis results. The influence on specific fuel consumption of these variables is shown in Figure 28. Over the range of values examined, it can be seen that the effect of core pressure ratio is most significant. Also shown in these results of parametric cycle analysis is the irrelevance of the term bypass ratio as a fundamental operator on specific fuel consumption or specific thrust. On a high bypass ratio engine, a large percentage of the thrust is generated by the bypass flow--approximately in proportion to the bypass ratio. Furthermore, the engine specific thrust, or thrust per unit of airflow, is a direct function of fan pressure ratio. Therefore, for an engine of a given thrust, the engine airflow (and essentially the fan, engine, and nacelle

ALTITUDE = 4572 M (15,000 FT)

VELOCITY = 463 KM/H (250 KNOTS)

THRUST = 2464 N (554 LBF)



TURBINE INLET TEMPERATURE	
°K	°F
1144	1600
1172	1650
1200	1700
1228	1750

Figure 28. Parametric Cycle Analysis Results.

dimensions) can be established by first selecting the fan pressure ratio. It can also be shown that by evaluating nacelle drag versus fan pressure ratio, a best pressure ratio may be selected that maximizes net installed thrust and consequently maximizes the propulsive efficiency component of net installed specific fuel consumption. It can be seen in Figure 28 that turbine inlet temperature and bypass ratio are interrelated functions if fan and core compressor pressure ratios are selected. For example, if a 1.2:1 pressure ratio fan and a 10:1 pressure ratio core are selected, Figure 28 shows that for a turbine inlet temperature of 1144°K (1600°F), the optimum bypass ratio is less than 10:1. Similarly, for 1228°K (1750°F), the optimum bypass ratio is greater than 12:1. This illustrates the effect of turbine inlet temperature on the size of the engine core. The difference in specific fuel consumption between these cases is only the effect of turbine inlet temperature on the thermal efficiency of the cycle.

Figure 29 shows that fan pressure ratio is the predominent factor in the important engine performance parameter--specific thrust. Any point on the curves in this figure is an optimized engine cycle. That is, for a given point on one of the compressor pressure ratio lines and at a given turbine inlet temperature (with constant component efficiencies and loss assumptions), there is no variation in engine cycle that will provide a lower specific fuel consumption. Only one kind of variation is possible; it is the choice of energy split between that delivered to the core jet nozzle and that of the fan. The energy split that is most efficient, in terms of specific fuel consumption, occurs at only one bypass ratio and by definition provides the optimized engine cycle referred to previously.

The dominant influence that turbine inlet temperature has on the size of the engine core is further illustrated in Figure 30. The core inlet corrected airflow is shown to vary substantially over the small range of turbine inlet temperatures examined. The core size varies to a lesser extent with compressor pressure ratio, which affirms that its influence on the specific power of a gas turbine cycle is small.

In defining the best engine candidates for engine cycle quality sensitivity analyses, the parametric cycle analysis results were carefully evaluated. It was determined that the best fan pressure ratio, as it affects net installed specific fuel consumption, could be ascertained directly. Figure 31 shows specific fuel consumption versus cycle pressure ratio for two cases of fan pressure ratio. The dashed line in Figure 31 represents the best attainable TSFC, discounting the effects on net performance of nacelle drag and engine weight. Along this line, fan pressure ratio varies from about 1.2:1 to 1.25:1. The solid

ALTITUDE = 4572 M (15,000 FT)

VELOCITY = 463 KM/HR (250 KNOTS)

THRUST = 2464 N (554 LBF)

T_{t_2} = 267°K (20.4°F)

P_{t_2} = 6.318 N/CM^2 (9.164 PSIA)

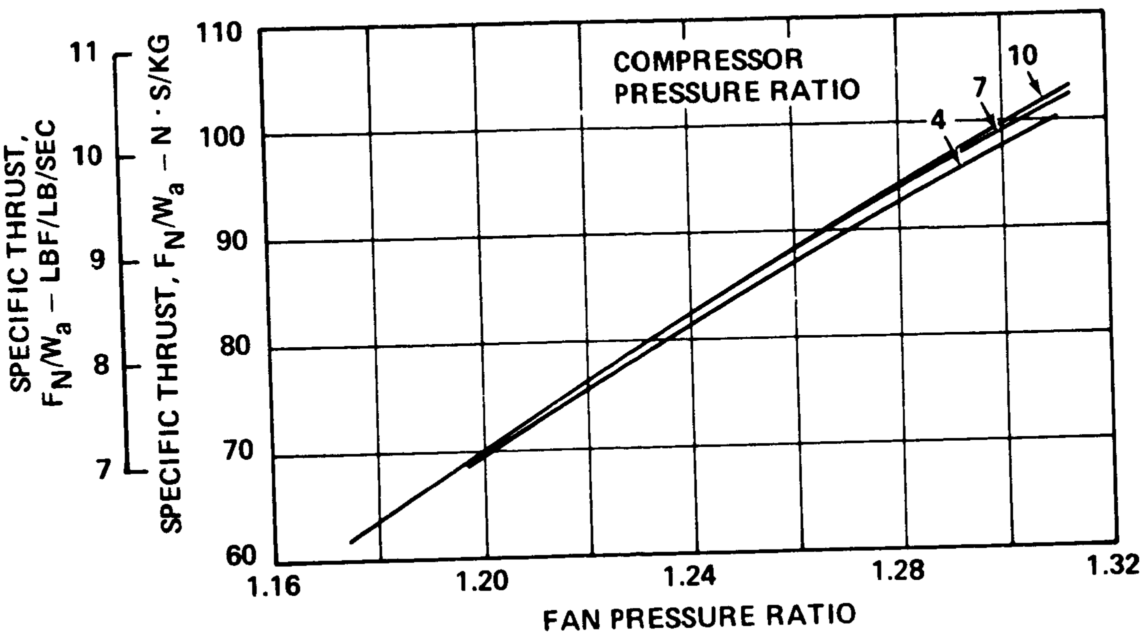


Figure 29. Engine Specific Thrust (Thrust/Airflow) Variation With Fan Pressure Ratio.

AIRFLOWS ARE FOR AN ENGINE SIZED FOR
2464 N (554 LB) NET CRUISE THRUST AT
463 KM/HR (250 KNOTS) 4572 M (15,000 FT)

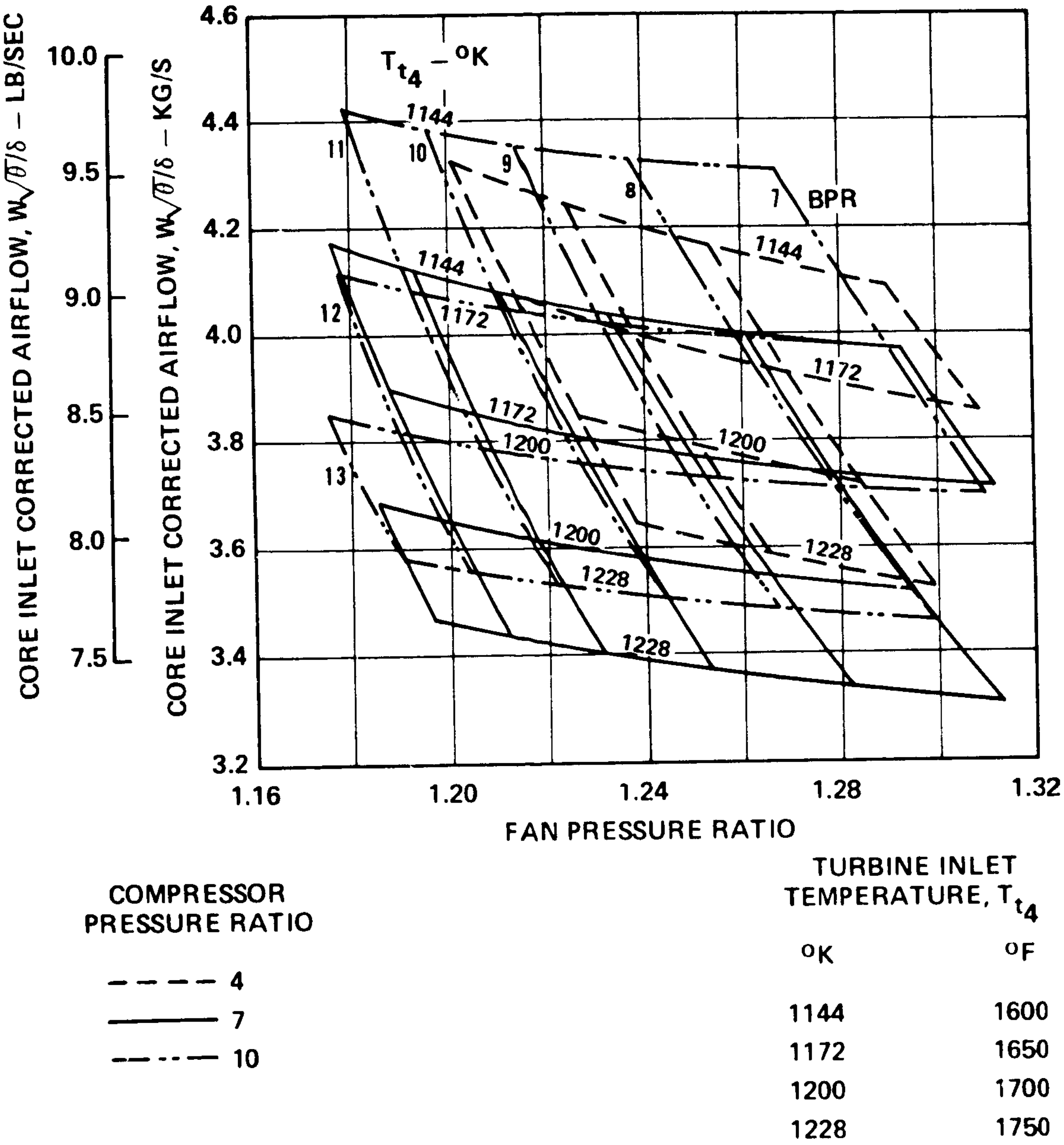


Figure 30. Parametric Analysis Results of Core Size Variation.

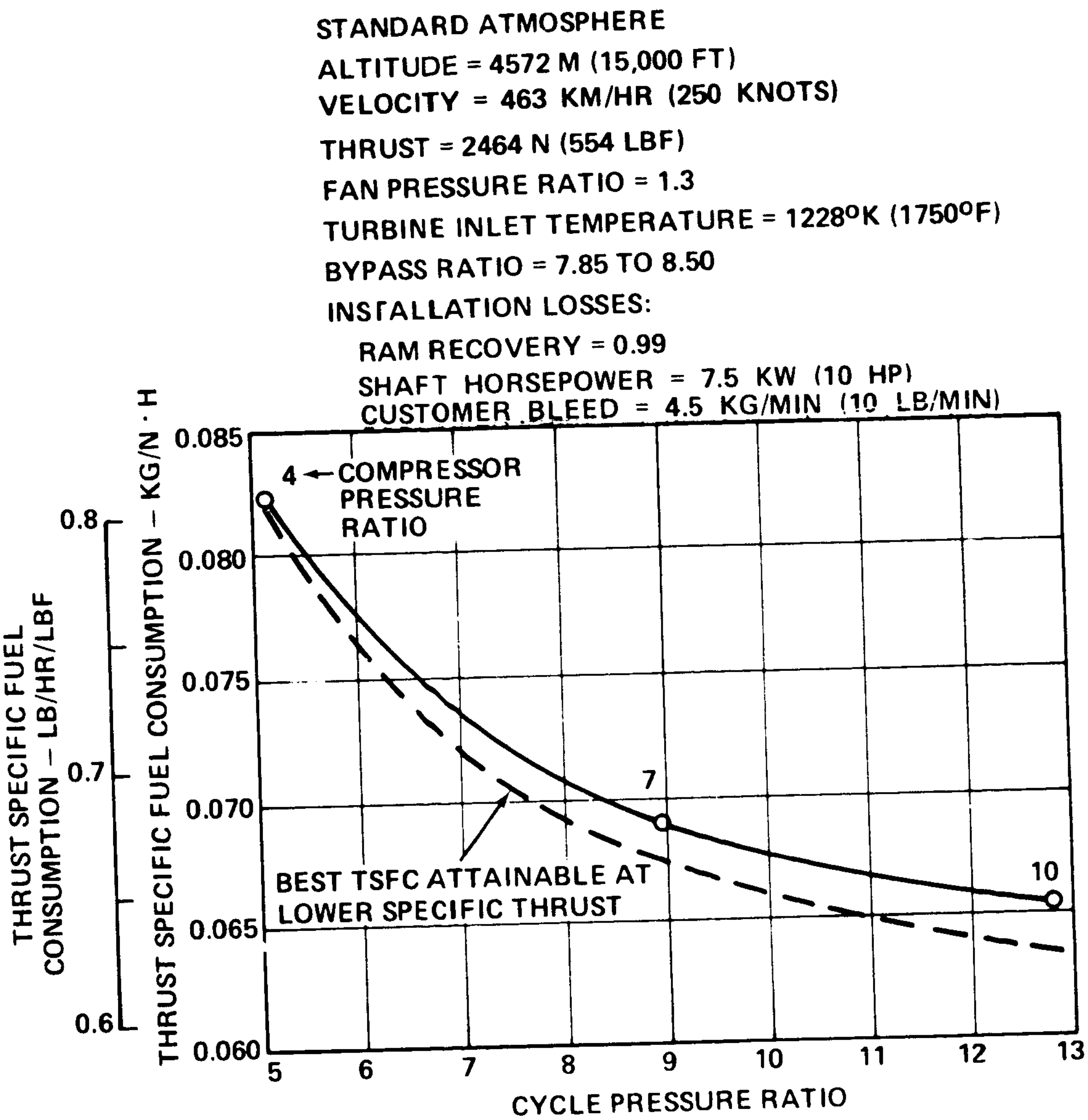


Figure 31. Thrust Specific Fuel Consumption Variation With Cycle Pressure Ratio.

line in Figure 31 represents a constant fan pressure ratio of 1.2:1, (and constant specific thrust), which is near optimum with respect to the tradeoffs between TSFC, and nacelle drag and engine weight. By examining Figures 28, 29 and 31, it can be seen that in the 10:1 core pressure ratio case, the 1.3 pressure ratio fan provides a 40 percent higher specific thrust than the 1.2 pressure ratio fan which provides the lowest TSFC. The effect on nacelle drag and engine weight of the higher specific thrust offsets the 3 percent increase in TSFC.

Because of its relatively small influence on TSFC and engine specific weight, turbine inlet temperature was similarly eliminated as a cycle quality variable in the synthesis sensitivity analyses. A turbine inlet temperature of 1228°K (1750°F) was chosen for sensitivity candidates as the highest temperature that would permit the use of an uncooled core turbine while retaining substantial growth potential.

The parametric cycle analysis results show that the variable exerting the greatest impact on cycle quality is core compressor pressure ratio. Over the range examined, specific fuel consumption varies about 30 percent. The candidate engine cycles selected for use in the synthesis sensitivity analyses differ in configuration, size, weight, and specific fuel consumption since three different pressure ratios were selected for comparison. Table XI lists the candidate engine cycle parameters, as well as the installed net thrust and TSFC for each engine.

Weights for the candidate engines were derived from the calculated weight of the original baseline engine, which is similar to Engine II in Table XI. Figure 32 is a plot of engine specific weight versus core pressure ratio, as calculated by a weight estimating computer program that was calibrated with the baseline engine weight. These weights were subsequently used in the synthesis sensitivity analyses.

Engine cycle quality sensitivity results. - With use of the GASP program, "solution" airplanes were calculated with each of the three engine candidates to evaluate the effects of cycle quality on airplane size and cruise fuel consumption. In Figure 33, the resultant gross and empty weights are plotted against engine core pressure ratio. These curves show that the 7:1 core chosen for the baseline engine minimizes empty weight and results in near minimum gross weight. Beyond 7:1, the empty weight increase is attributable to increased engine weight, which is partly offset by a small reduction in airplane structural weight because of lower fuel weight. The effect on gross weight is similar beyond 8.5:1 pressure ratio.

TABLE XI. CANDIDATE ENGINE CYCLE PARAMETER DEFINITION

	I	II	III
Inlet (Fan) Corrected Airflow	38.317 kg/s (84.475 lb/s)	38.317 kg/s (84.475 lb/s)	38.317 kg/s (84.475 lb/s)
Fan Pressure Ratio	1.3	1.3	1.3
Fan Efficiency	0.89	0.89	0.89
Bypass Ratio	8.0	8.397	7.85
Compressor Inlet Corrected Airflow	3.468 kg/s (7.645 lb/s)	3.321 kg/s (7.322 lb/s)	3.526 kg/s (7.774 lb/s)
Compressor Pressure Ratio	10.0	7.0	4.0
Compressor Efficiency	0.806	0.816	0.824
Turbine Inlet Temperature	1228°K (1750°F)	1228°K (1750°F)	1228°K (1750°F)
HPT Inlet Corrected Flow	0.713 kg/s (1.572 lb/s)	0.976 kg/s (2.152 lb/s)	1.821 kg/s (4.015 lb/s)
HPT Pressure Ratio	3.85	2.799	1.902
HPT Efficiency	0.852	0.852	0.852
LPT Inlet Corrected Flow	2.473 kg/s (5.451 lb/s)	2.540 kg/s (5.599 lb/s)	3.345 kg/s (7.375 lb/s)
LPT Pressure Ratio	2.837	2.764	2.377
LPT Efficiency	0.89	0.89	0.89
Installed Net Thrust(1)	2464N 554 lb	2464N 554 lb	2464N 554 lb
Installed TSFC(1)	0.0654 kg/N.h 0.641 lb/hr/lb	0.0689 kg/N.h 0.676 lb/hr/lb	0.0824 kg/N.h 0.808 lb/hr/lb

(1)At design point, 463 kg/hr (250 kt), 4572 m (15,000 ft).

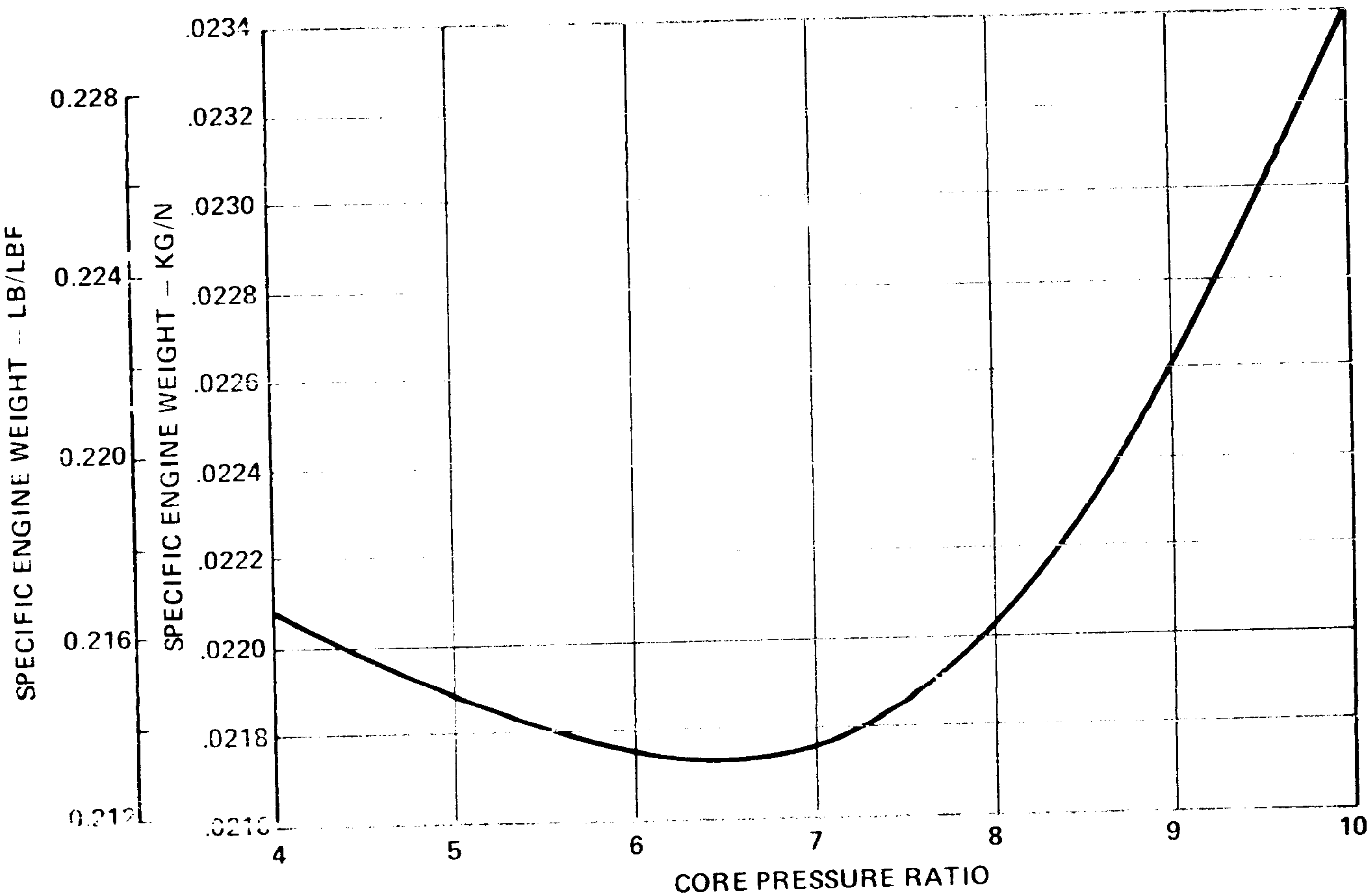


Figure 32. Calculated Specific Engine Weight (Weight ÷ Thrust) Variation With Core Compressor Pressure Ratio.

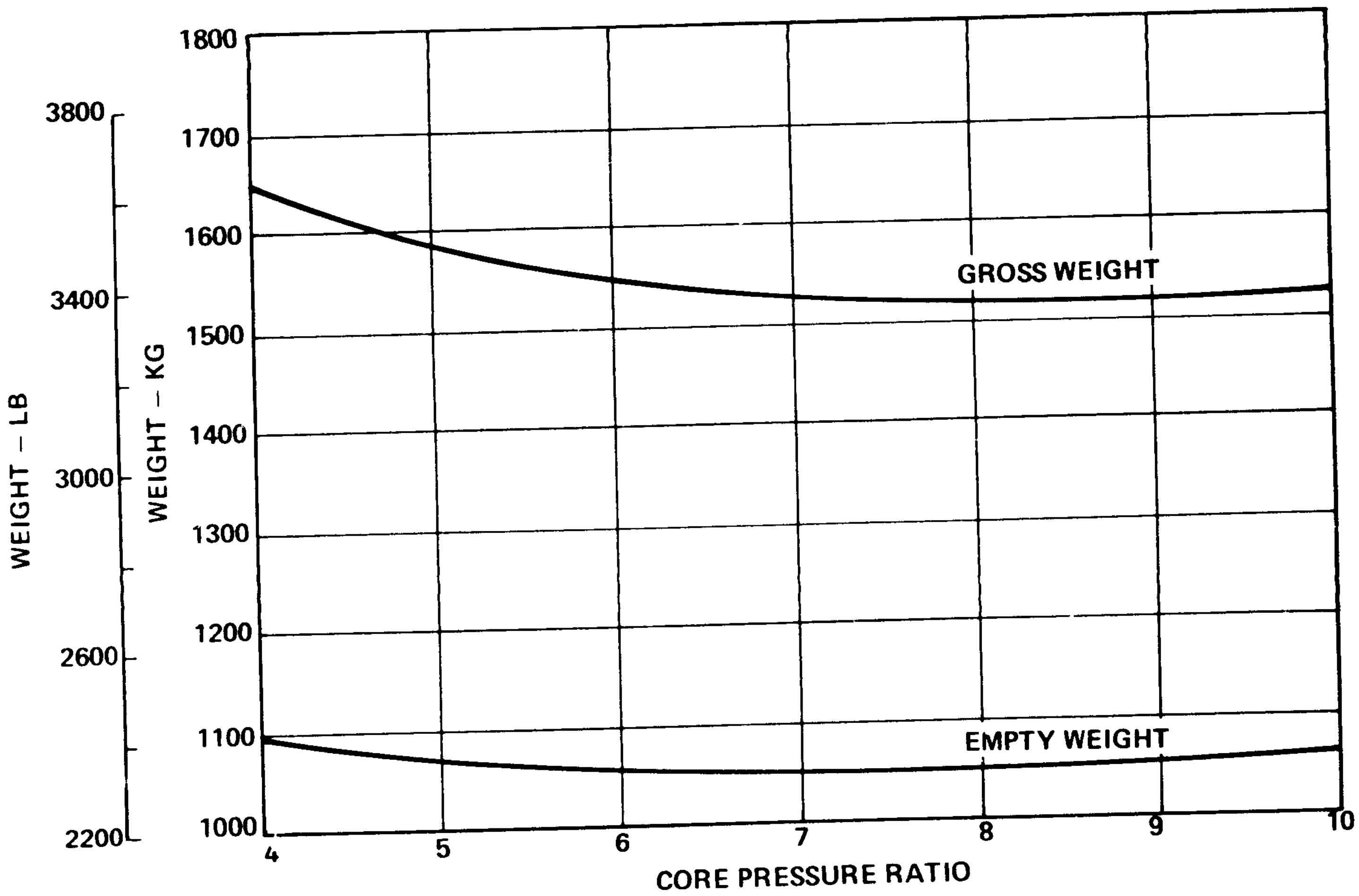


Figure 33. Gross and Empty Weight Variation With Core Compressor Pressure Ratio.

The rate of cruise fuel consumption of the "solution" airplanes is plotted in Figure 34 against core pressure ratio. As shown, fuel consumption will continue to decrease beyond 7:1 pressure ratio, but only by a small amount. It can be anticipated that life-cycle cost analyses would show that the benefits derived from the higher pressure ratio, in the trade-off between inherent higher engine cost and lower fuel cost, are negligible.

The results of this sensitivity analysis indicate that the cycle quality chosen for the baseline engine is near optimum for the mission and performance requirements selected in the Randolph study. Final designs of the four airplanes which resulted from this study therefore use the 7:1 core pressure ratio engine.

Airplane drag sensitivity results. - It can be observed that airplanes differ widely with respect to their zero-life drag coefficient, which is a measure of aerodynamic quality. Comparisons of any two airplanes designed at the same time to essentially the same mission problem statement will reveal surprising differences in aerodynamic quality as manifested in either airplane size or performance. In exercising the airplane design art, choosing between the fundamental configuration alternatives has a major effect on resultant aerodynamic quality. Beyond that, however, it has been shown that "attention to detail" is of equal or more significance.

This art is, of course, well known to airplane designers. Proper execution of wing root fillets and fairings, the "flushing" of protuberances and excrescencies, sealing gaps, shutting off circulation through holes and wheel wells, specification of skin finish and smoothness, and the shaping or tailoring of component forms are a few examples of the art as it is practiced. An example that has become a text book classic is the work of NACA's R.H. Lange on the Grumman TBM. Using an incremental approach, i.e., fixing one "draggy" detail at a time, he was able to effect a total drag reduction of 30 percent, which is equivalent to removing either the wing or the fuselage increment from the total airplane profile drag.

It is apparent that not all airplanes undergo equal treatment in the minimization of drag. It is probable that designers are often stopped short in deference to arbitrary cost-effectiveness considerations. In general aviation, for example, a drag increment is generally equated to the performance increment that it causes. It is doubtful that the question is ever asked, "If all mission and performance factors are held constant, what is the effect of a drag increment on the size, initial and operating costs of the 'solution airplane?" A cost-effectiveness judgment can be made by answering this question.

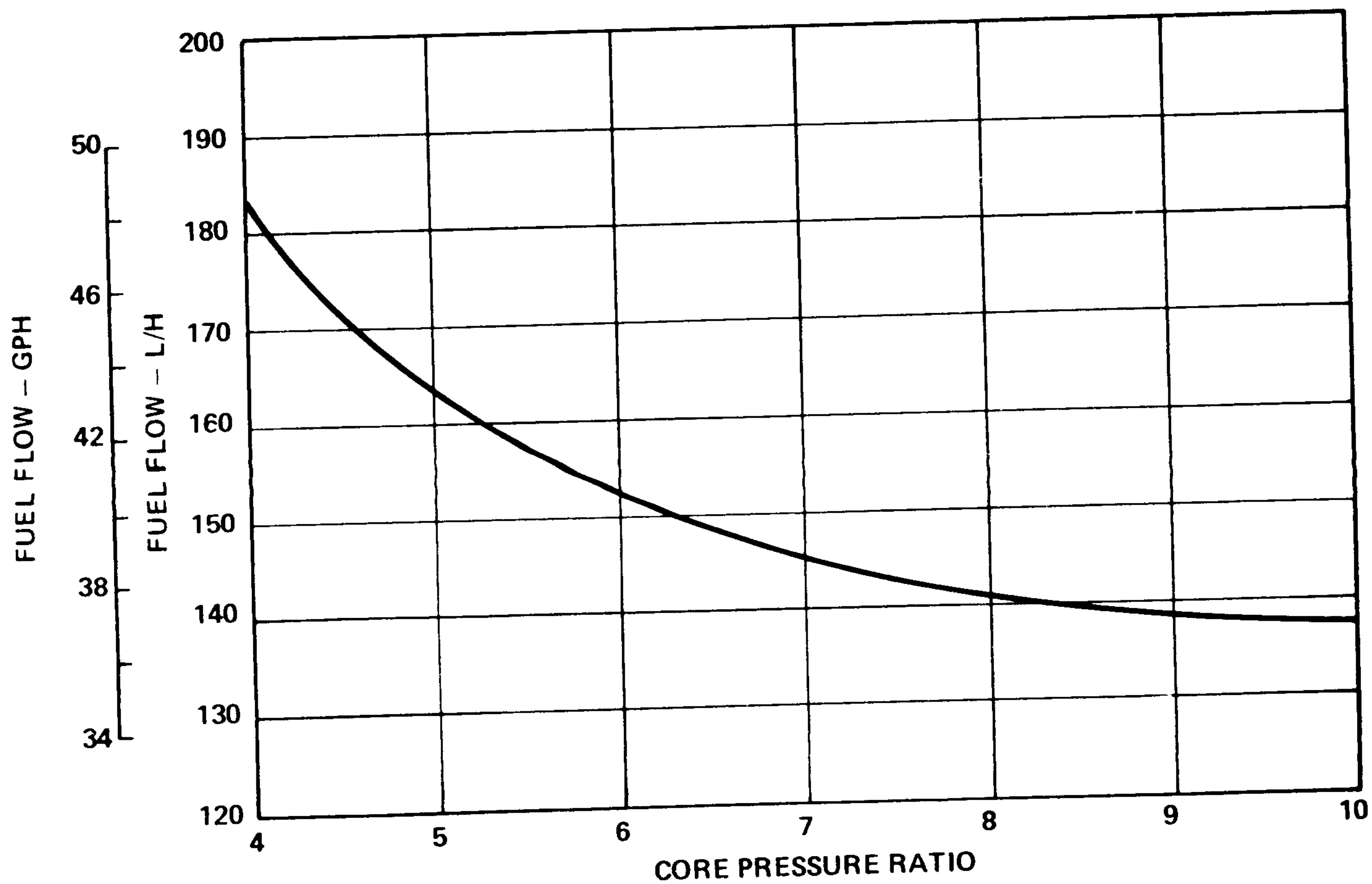


Figure 34. Cruise Fuel Consumption Variation With Core Compressor Pressure Ratio.

The GASP synthesis computer program makes it possible to answer this question easily. By adding a drag increment while holding all other input parameters constant, it will compute a new "solution" airplane and list the factors that determine its initial and operating costs. Then, a valid cost-effectiveness evaluation can be made, and the drag-reduction efforts of the designer will probably be continued.

In checking the GASP results on the side-by-side single-engine airplane, Cessna's drag buildup analysis indicated that the GASP-calculated zero-lift drag coefficient was optimistic. The Cessna analysis showed several areas where configuration changes and "attention to detail" would serve to eliminate the discrepancy. For example, moving the engines aft on the wing would reduce nacelle-wing interference drag, and shortening the engine nacelles would reduce their profile drag. Therefore, it was determined that a drag sensitivity analysis should be performed. By using GASP to show the effect on airplane size and fuel consumption of an added drag increment, it was expected that incentive would be provided to find the means whereby the original GASP-calculated coefficient could be achieved.

Figure 35 shows that for a 10-percent increase in zero-lift drag coefficient, the airplane empty weight would increase 3.2 percent, and the gross weight would increase 5.3 percent. The effect on engine size and airplane fuel consumption is illustrated in Figure 36. For the same 10-percent drag coefficient increase, the new "solution" airplane would have a 10.5-percent larger engine, and 13 percent greater fuel consumption. It was estimated that these effects could add $6000 to the flyaway cost of the airplane and cause it to consume 280,000 liters (74,000 gallons) of additional fuel in its lifetime. The 10-percent drag increment that would cause these increases is equivalent to the addition of two automotive-type rear-view mirrors on the exterior of the airplane.

Fixed equipment weight sensitivity results. - The fixed equipment group includes all avionics, instruments, furnishings, electrical equipment, lines, and fittings--that is, all airplane parts that remain fixed in size, quantity, and weight regardless of the "solution" airplane size. Cessna prepared an equipment list and a weight breakdown of all such equipment applicable to each baseline airplane design, which was consistent with the Randolph study requirements list. The Cessna weight estimates were based on state-of-the-art hardware wherever possible. However, since optimum hardware is usually the most costly and difficult to obtain, it is often subjected to relaxation of optimistic weight targets during a design and development program.

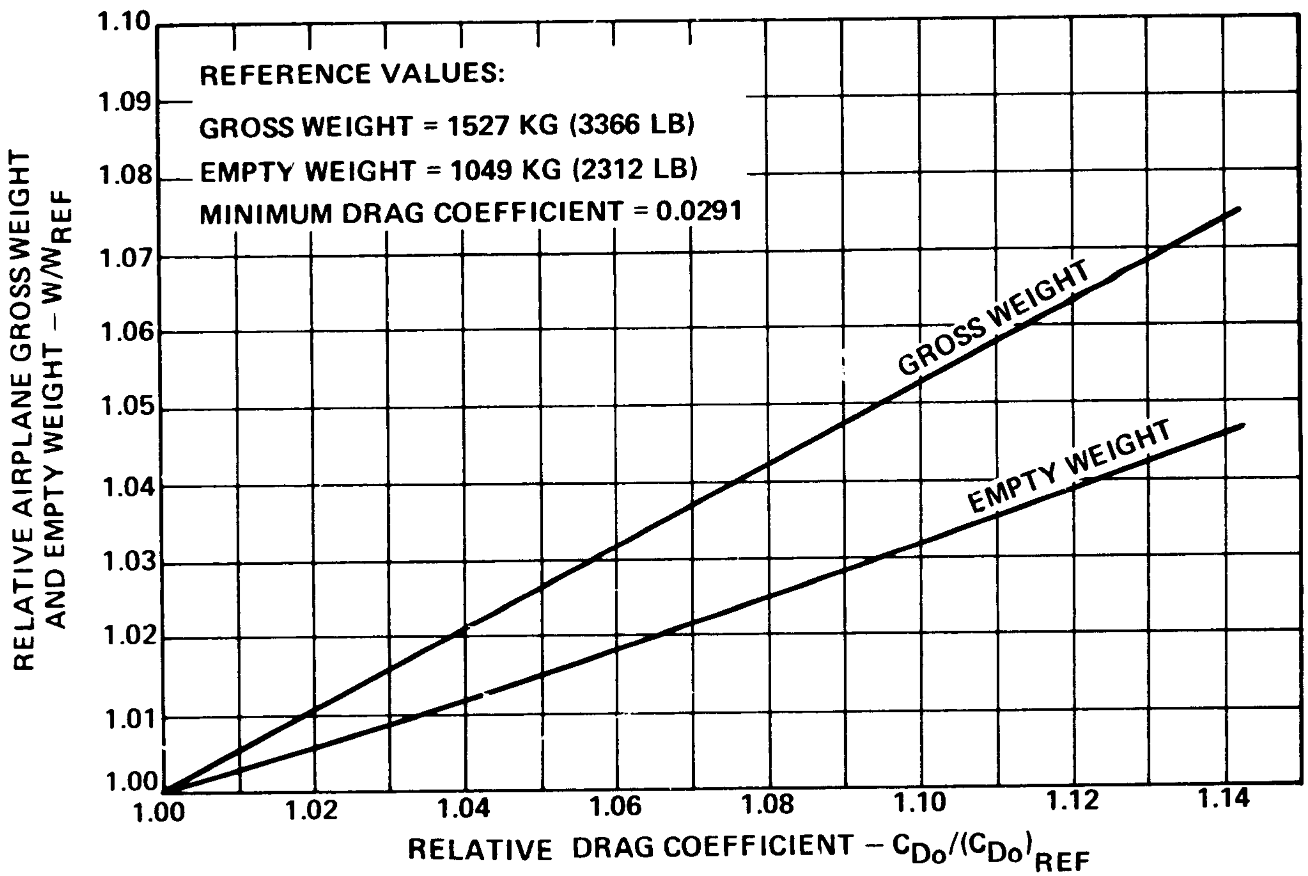


Figure 35. Gross and Empty Weight Sensitivity to Zero-Lift Drag Coefficient.

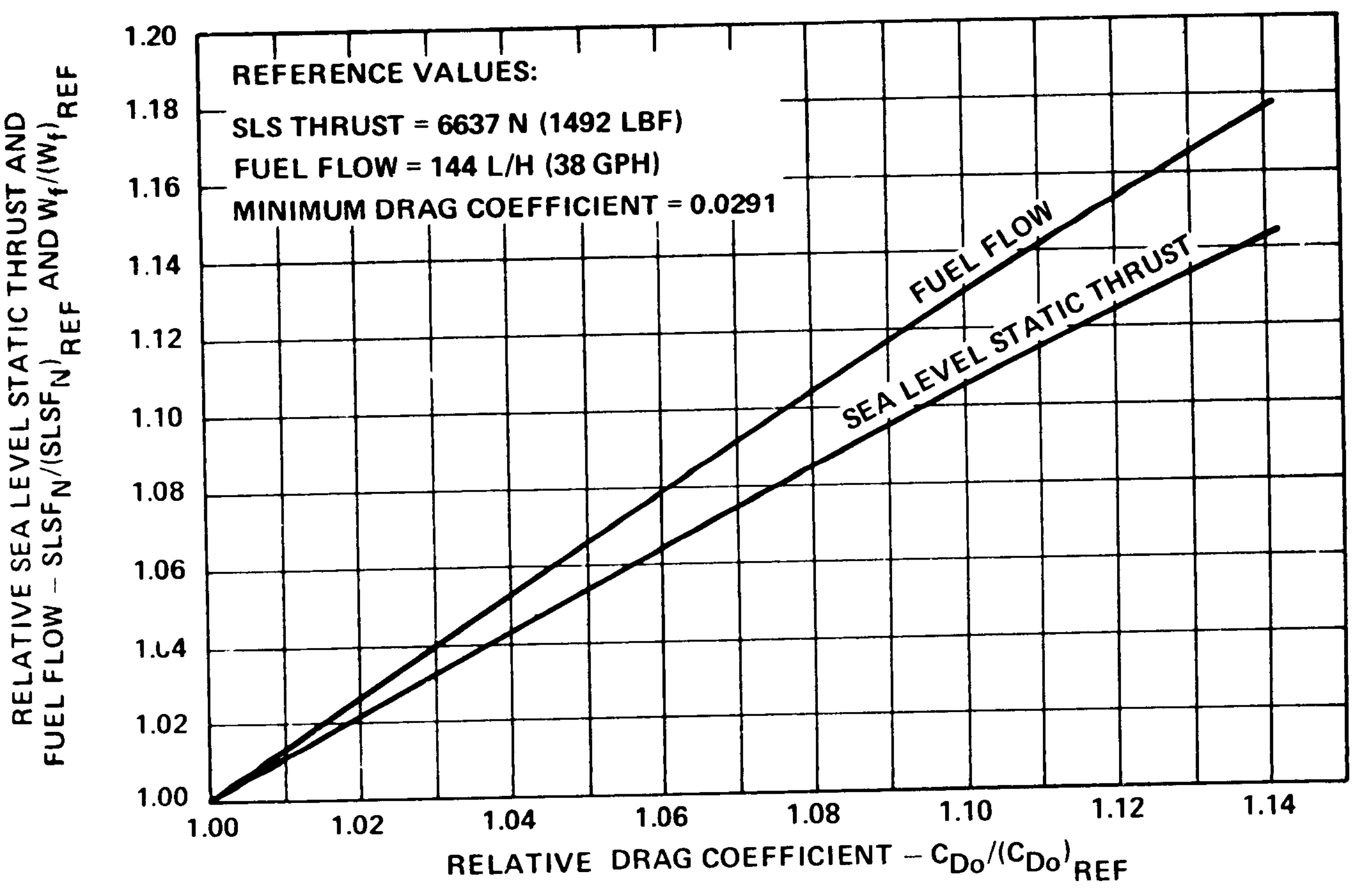


Figure 36. Engine Size and Cruise Fuel Consumption Sensitivity to Zero-Lift Drag Coefficient.

For the same reason that the drag sensitivity study was performed, a similar analysis was made on the impact of fixed equipment weight increases. Cost-effectiveness of a change in component weight can be evaluated only if the effects on "solution" airplane size, unit cost, and operating cost are known. Figure 37 shows that over the range examined, a 0.45-kg (1 lb) equipment weight increase over the 303-kg (669 lb) baseline value would increase empty weight 0.785 kg (1.73 lb) and gross weight 0.916 kg (2.02 lb). Similarly (refer to Figure 38), the 0.45-kg (1 lb) change would increase required engine thrust by 1.38 N (0.31 lb), and cruise fuel consumption by 0.08 L/H (0.02 gph).

The significance of these "per-pound" increments must be evaluated carefully. It is estimated that a 10-percent, or 30-kg (67 lb) equipment weight increase would increase the airplane cost by about $3000 and its lifetime fuel consumption by about 75,700 liters (20,000 gallons). Any cost savings in a 30-kg (67 lb) heavier but less expensive equipment fit would have to offset these life-cycle cost increases in order to be cost-effective. The foregoing sensitivity factors apply as well to fuselage structural weight and to anything carried in the fuselage, including payload. Again, these factors apply when all mission and performance requirements are unchanged.

Wing weight sensitivity results. - There are several well known wing weight prediction equations used in airplane preliminary design. These equations are based on derived correlations between the known weights of existing wings, the weights and design load factors of the airplanes they support, and pertinent wing geometry variables. GASP contains such an equation in its weight calculating module. The GASP-calculated wing weight of the baseline airplane was compared with the results from two other equations. Less than 5 percent difference was found. However, the equation used by Cessna in their weight estimates yielded a heavier wing. It was determined that this equation was simply conservative. Again a sensitivity study was called for, to determine the impact of wing weight variation on "solution" airplane characteristics that determine cost.

Figures 39 and 40 show sensitivities to wing weight increase nearly identical with those shown for fixed equipment weight. The example used in the fixed equipment weight analysis is therefore applicable here, and should foster a similar concern for the achievement of a light wing.

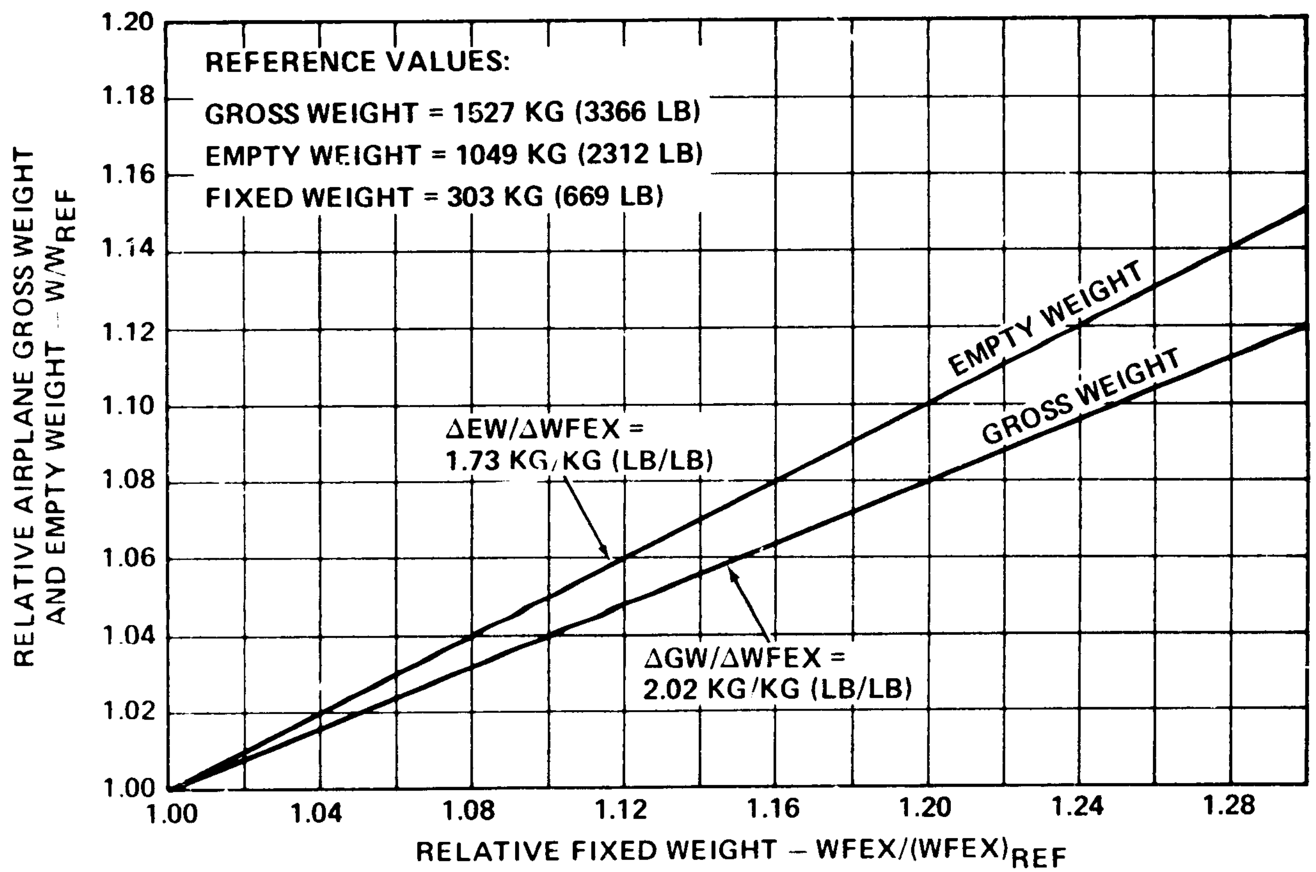


Figure 37. Gross and Empty Weight Sensitivity to Fixed Equipment Weight.

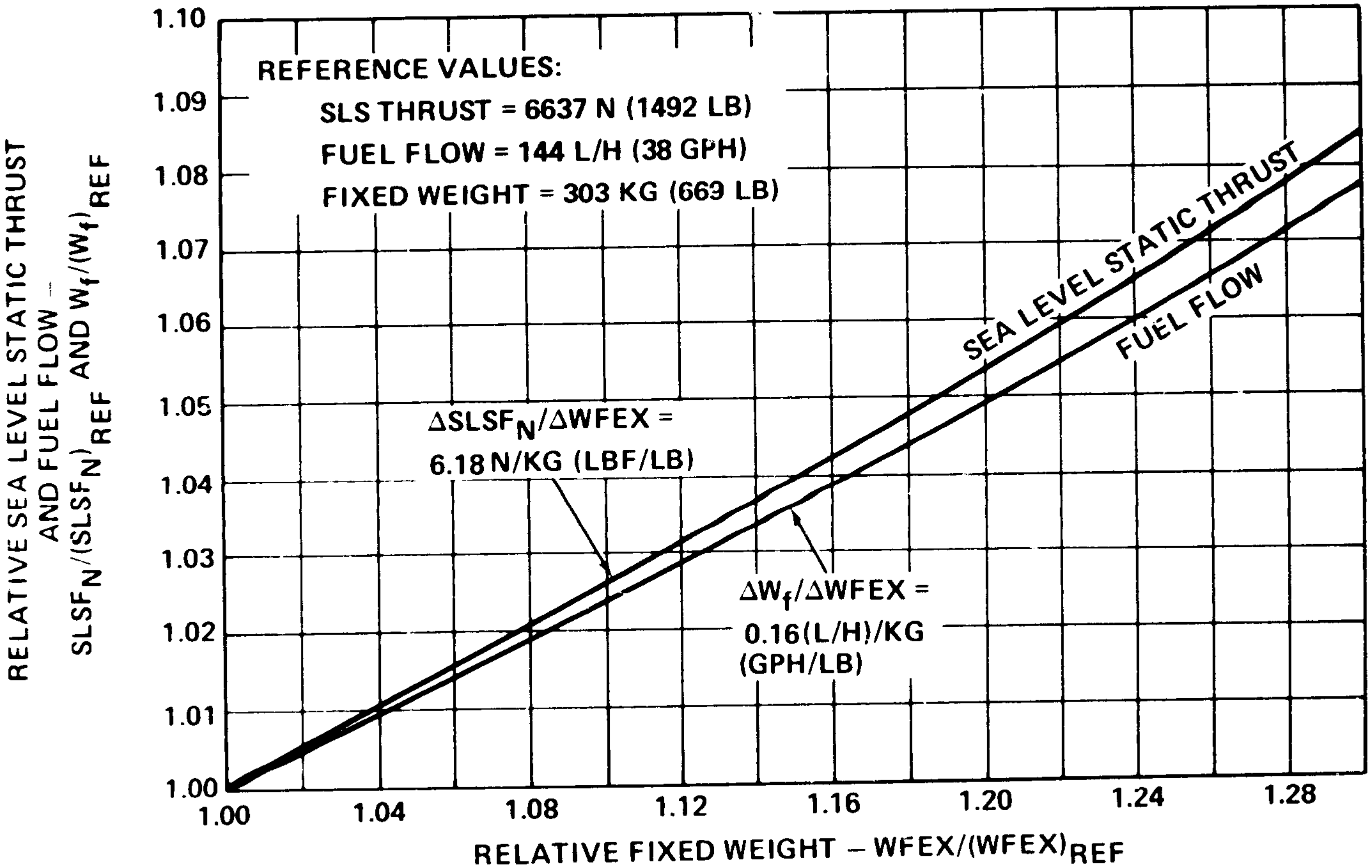


Figure 38. Engine Size and Cruise Fuel Consumption Sensitivity to Fixed Equipment Weight.

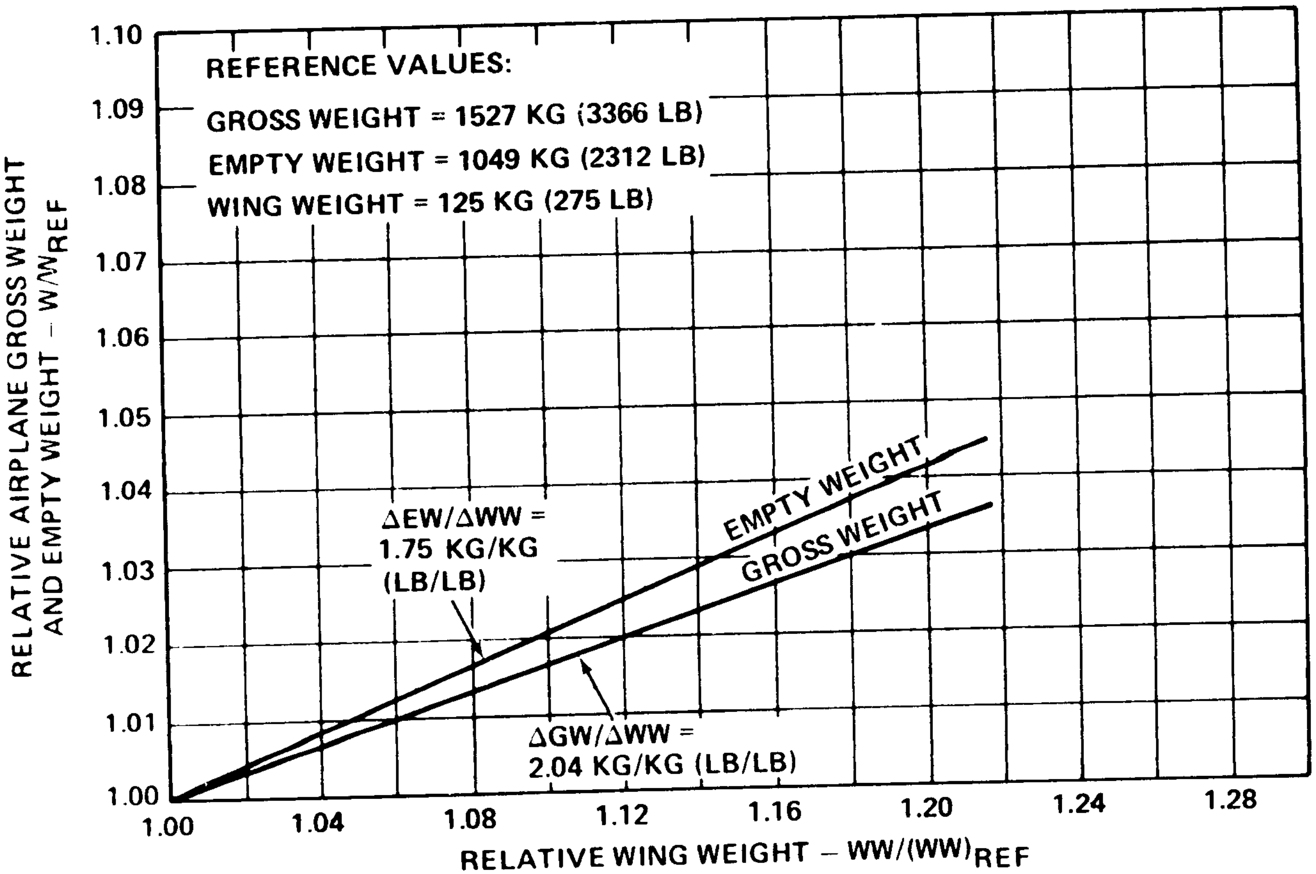


Figure 3[illegible]. Gross and Empty Weight Sensitivity to Wing Weight.

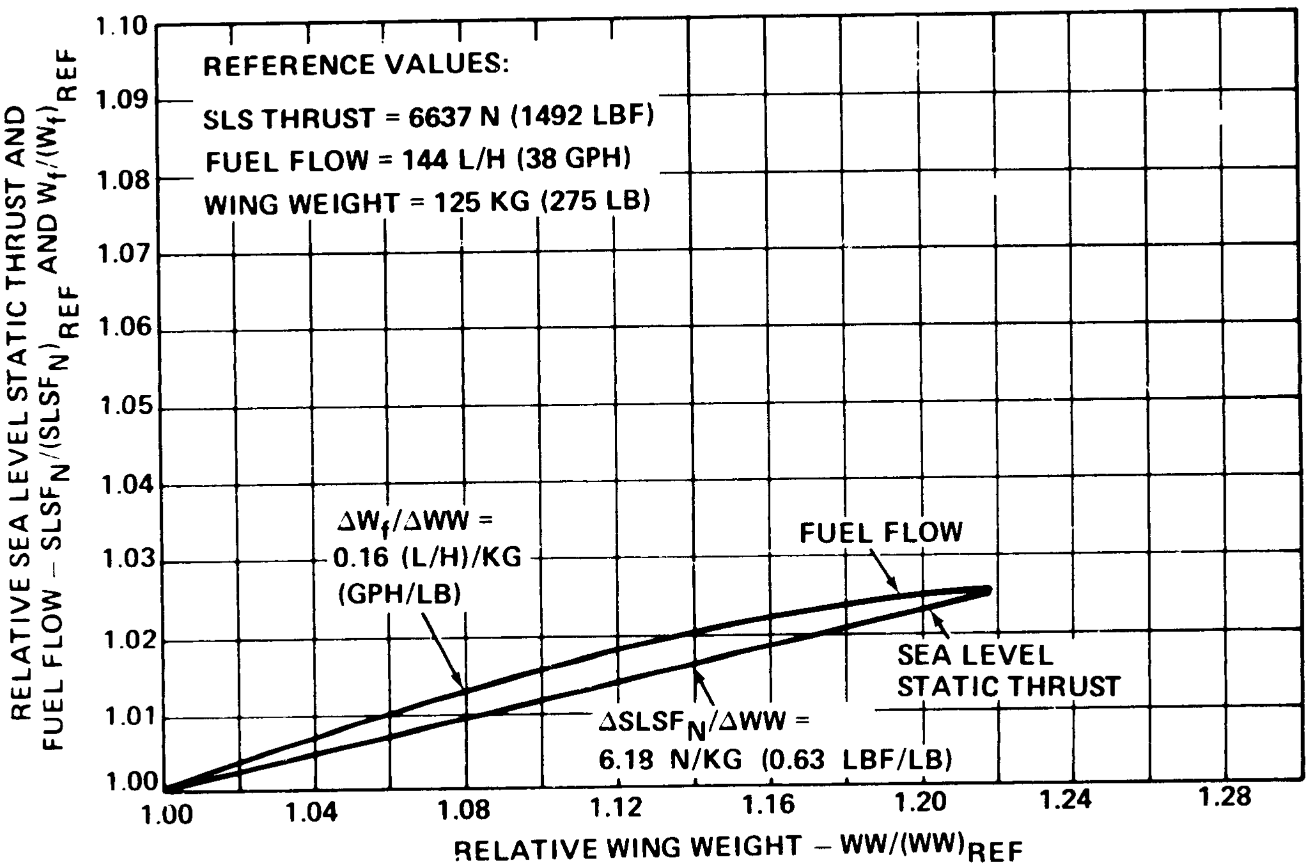


Figure 40. Engine Size and Cruise Fuel Consumption Sensitivity to Wing Weight.

Sustained load factor sensitivity results. - It was shown in the section concerning wing- and thrust-loading studies that the design requirement for 2.5 g's sustained load factor, or maneuver rate, at 4572 m (15,000 ft) altitude sized the engines. This desirable turning capability was undoubtedly weighed carefully in the Randolph study before it was made a requirement. There is no evidence, however, that the cost of this capability was examined in the study. For example, a lower cost alternative would be to specify the 2.5 g's at some lower altitude, or to specify 2.0 g's at 4572 m (15,000 ft) altitude.

The latter case may be examined in Figures 41 and 42. At 2.0 g's sustained load factor, the "solution" airplane would have a gross weight 7.5 percent less and an empty weight 8.25 percent less. Cruise fuel consumption would be reduced about 11 percent, and the engine sea level static thrust would be 26 percent lower. It was estimated that the unit cost of the airplane could be reduced about $15,000, and lifetime fuel consumption reduced by 227,000 liters (60,000 gallons). Expert knowledge of primary training aerobatics would be required to judge the value of the 0.5-g sustained load factor increment and evaluate its cost-effectiveness.

Combined sensitivity effects. - To this point in the study, the multiplying or synergistic effects of single-parameter variations have been demonstrated. It thus is essential to know whether combining parametric variables would cause gross changes in "solution" airplane size and economics.

As a test for this effect, a GASP calculation was made wherein arbitrarily decremented values were assigned to three selected input parameters. By manipulation of the wing weight equation, specific wing weight was increased 20 percent. The zero-lift drag coefficient and the fixed equipment weight were both increased 10 percent. The results of this analysis showed a 12-percent gross weight increase, 13 percent empty weight increase, 15 percent increase in cruise fuel consumption, and 13 percent increase in engine thrust. The overall effect of combined-parameter variables was surprisingly small. With such decremented quality, the trainer designed to the Randolph study problem statement, but using a "best" engine, still improved on the Randolph results by a large margin.

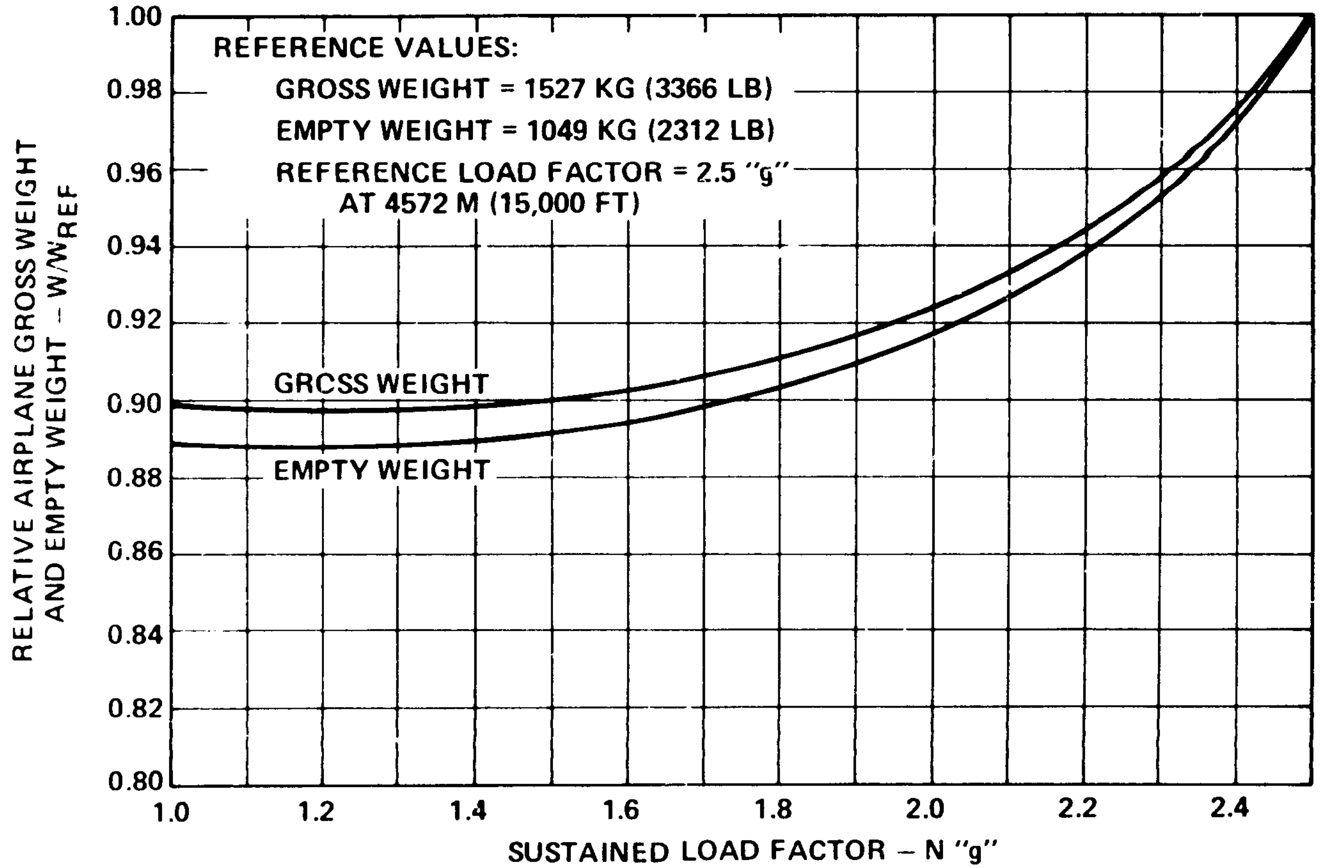
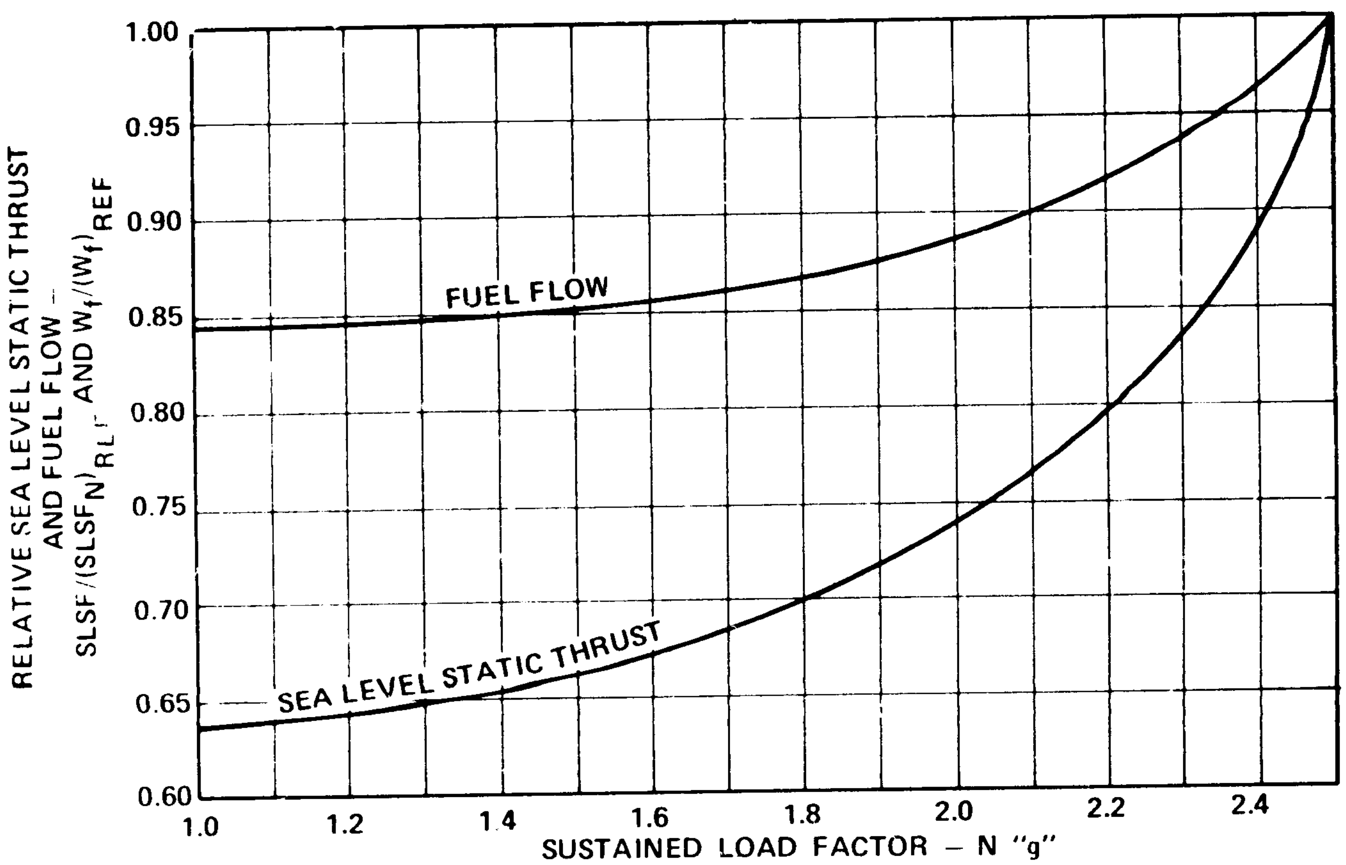


Figure 41. Gross and Empty Weight Sensitivity to Sustained Load Factor Capability.



REFERENCE VALUES:

SLS THRUST = 6637 N (1492 LBF)

FUEL FLOW = 144 L/H (38 GPH)

SUSTAINED LOAD FACTOR = 2.5 g's AT 4572 M (15,000 FT)

Figure 42. Engine Size and Cruise Fuel Consumption Sensitivity to Sustained Load Factor Capability.

TASK III - EVALUATION OF THE TANDEM-SEAT, TWIN-ENGINE TRAINER

As described in the introduction of this report, the original contract work statement was modified early in the program to permit a more thorough study of military primary trainers than was originally intended. From discussions held with training command personnel concerning side-by-side versus tandem seating, and single versus twin-engine configurations, it was learned that varying training requirements and philosophies dictate the choice between such alternatives. It was therefore determined that a proper course for this investigation was to define and evaluate airplanes having each of the configuration alternatives. In this manner, the impact of airplane configuration on the installed performance of high bypass ratio engines could be evaluated.

The tandem seating configuration provides significant advantages in instrument training. For example, the simulation of instrument flight rule conditions is facilitated by the ability to hood the rear cockpit, thereby eliminating the student's visual cues to flight attitude. A further advantage is the unimpaired visability provided to the front seat instructor or safety pilot. The elimination of instructor "presence" permits a psychological effect also considered important.

In this study, the tandem configuration differs in design from the side-by-side airplane only in those things that result from the seating arrangement. Fixed equipment weight is greater because of the duplication of some cockpit instrumentation and avionics. There is a small increase in fuselage dimensions, weight, and drag resulting from the greater cockpit volume. As a consequence of these increases, the tandem configuration airplane has 126 kg (227 lb) greater gross weight or approximately eight percent. Ten percent higher thrust engines are required, and cruise fuel consumption is approximately eleven percent greater than that of the side-by-side airplane solution. It was estimated that this growth would result in about $9000 greater unit cost than the equivalent side-by-side airplane, and lifetime fuel consumption would be about 227,000 liters (60,000 gallons) greater. It is reasonable to assume, however, that the advantages offered by the tandem configuration for instrument training could more than offset these increases. By using this more capable primary trainer for a higher percentage of total training syllabus hours, significant savings in overall training cost and fuel consumption could be achieved.

The three-view drawing presented in Figure 43 defines the tandem-seat, twin-engine configuration. GASP printouts given in Appendix B list additional data.

TASK IV - EVALUATION OF THE TANDEM-SEAT, SINGLE-ENGINE TRAINER

Configuration Studies and Final Design Results

This configuration presented an inherent design problem that was the most difficult of those addressed in the study. The factors that create the problem appear to be inalterable and make the final "solution" airplane the least attractive of the configurations studied. In essence, there is no reasonable location for the engine that will achieve the low installation losses permitted in the other configurations. It is an inherent characteristic of turbofan engines having low fan pressure ratio and high bypass ratio that their performance is greatly affected by inlet and exhaust-duct-system pressure losses. The engine cycle found to be optimum in this study exhibits this sensitivity to duct pressure loss.

The airplane configuration difference that contributes to the problem is the high cockpit canopy profile required to accommodate the elevated rear seat. Modern tandem-seat trainers incorporate this feature in order to maximize the instructor's forward visibility and it is considered essential. Therefore, if the inlet configuration and engine installation were executed in the manner of the side-by-side single, the engine thrust axis would be high above the airplane center of pressure. The consequent trim drag would have the same effect on airplane solutions as alternative high-pressure-loss installations. Raising the wing to a mid-fuselage location to raise the airplane center of pressure was found to be undesirable due to increased restriction of downward visibility from the instructor's seat. Lowering the engine and fuselage tail cone by increasing the amount of inlet duct offset would lower the thrust axis, but would result in high pressure loss.

Based on evaluations of several alternative designs, a fuselage-buried engine installation was chosen for the final design and GASP calculation. An assessment was made of the pressure loss for rectangular-shaped side inlets, and engine performance was altered to reflect this loss. Figure 44 shows a three-view drawing of this solution. Corresponding GASP printouts are given in Appendix B.

The disappointing results of this design show a 19-percent increase in gross weight, a 14-percent increase in empty weight, a 26-percent larger engine, and a 47-percent higher cruise fuel consumption over values achieved in the side-by-side single engine airplane.

PERFORMANCE

CRUISE SPEED (15,000 ft ALT)	250 kt (288 mph 0.399 M_n)
STALL SPEED (FULL FLAPS)	~60 kt
RANGE (WITH 2 CREW)	~700 nm (+45 min. RESERVE)
TAKEOFF DISTANCE (SL STD DAY)	~1556 ft (OVER 35 ft)
LANDING DISTANCE (S.L STD DAY)	~1680 ft (OVER 50 ft)
SUSTAINED MANEUVER RATE	2.5 g (AT 15,000 ft)
MAX RATE OF CLIMB (S.L - 15,000 ft)	3800-2660 fpm

WEIGHTS

GROSS	3643 lb
EMPTY	2522 lb
OPERATING (INC ONE CREW)	2732 lb
FIXED USEFUL LOAD	210 lb
PAYLOAD	200 lb
FUEL (MAX)	712 lb (106 gal)
FIXED EQUIPMENT GROUP	736 lb
FLIGHT CONTROLS GROUP	81 lb
STRUCTURES GROUP	1205 lb
PROPULSION GROUP	499 lb

DIMENSIONS & AREAS

FUSELAGE	
LENGTH	29 ft
WIDTH	3.25 ft
HEIGHT	5.5 ft
WING	
AREA	81 sq ft
SPAN	28.5 ft
GEOMETRIC MEAN CHORD	2.95 ft
ASPECT RATIO	10 -
TAPER RATIO	0.5 -
THICKNESS/CHORD (ROOT-TIP)	0.17 -
HORIZONTAL TAIL	
AREA	16.6 sq ft
SPAN	9.1 ft
MEAN CHORD	1.9 ft
ASPECT RATIO	5 -
THICKNESS/CHORD	0.07 -
VERTICAL TAIL	
AREA	16.4 sq ft
SPAN	5.6 ft
MEAN CHORD	3.0 ft
ASPECT RATIO	1.9 -
THICKNESS/CHORD	0.087 -
ENGINE NACELLES (2)	
MEAN DIAMETER	1.96 ft
LENGTH	3.4 ft
LANDING GEAR	
MAIN TIRE SIZE	4.4 × 18
NOSE TIRE SIZE	4.4 × 14
WHEEL BASE	10 ft
WHEEL TREAD	6.67 ft

ENGINE DATA (2 ea)

THRUST (SLS STD DAY)	818 lb
AIRFLOW (" " ")	41.9 lb/sec
CRUISE THRUST (250 kt, 15000 ft)	186 lb (REQD)
WEIGHT (DRY)	175 lb

AERODYNAMIC DATA

WING LOADING	45 lb
CRUISE DRAG POLAR	$C_D = 0.030 + 0.0406 C_L^2$
LANDING GEAR C_D INCREMENT	0.0279
EFFECTIVE FLAT PLATE AREA	2.432 sq ft
WETTED AREA	585 sq ft
MEAN SKIN FRICTION COEFFICIENT	0.00416
CRUISE REYNOLDS NO./FOOT	1.843×10^6
C_L MAX (WITH FULL FLAPS)	3.72
HORIZONTAL TAIL VOLUME COEF	0.93
VERTICAL TAIL VOLUME COEF	0.094

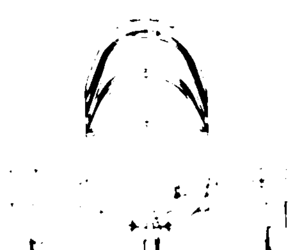


Figure 43. Three-View Drawing Twin-Engine Tan... Se

(2 ea)

S STD DAY)	818 lb
" ")	41.9 lb/sec
ST (250 kt/15000 ft	186 lb (REQD)
Y)	175 lb

DATA

NG	45 lb
POLAR	$C_D = 0.030 + 0.0406\, C_L^2$
R C_D INCREMENT	0.0279
LAT PLATE AREA	2.432 sq ft
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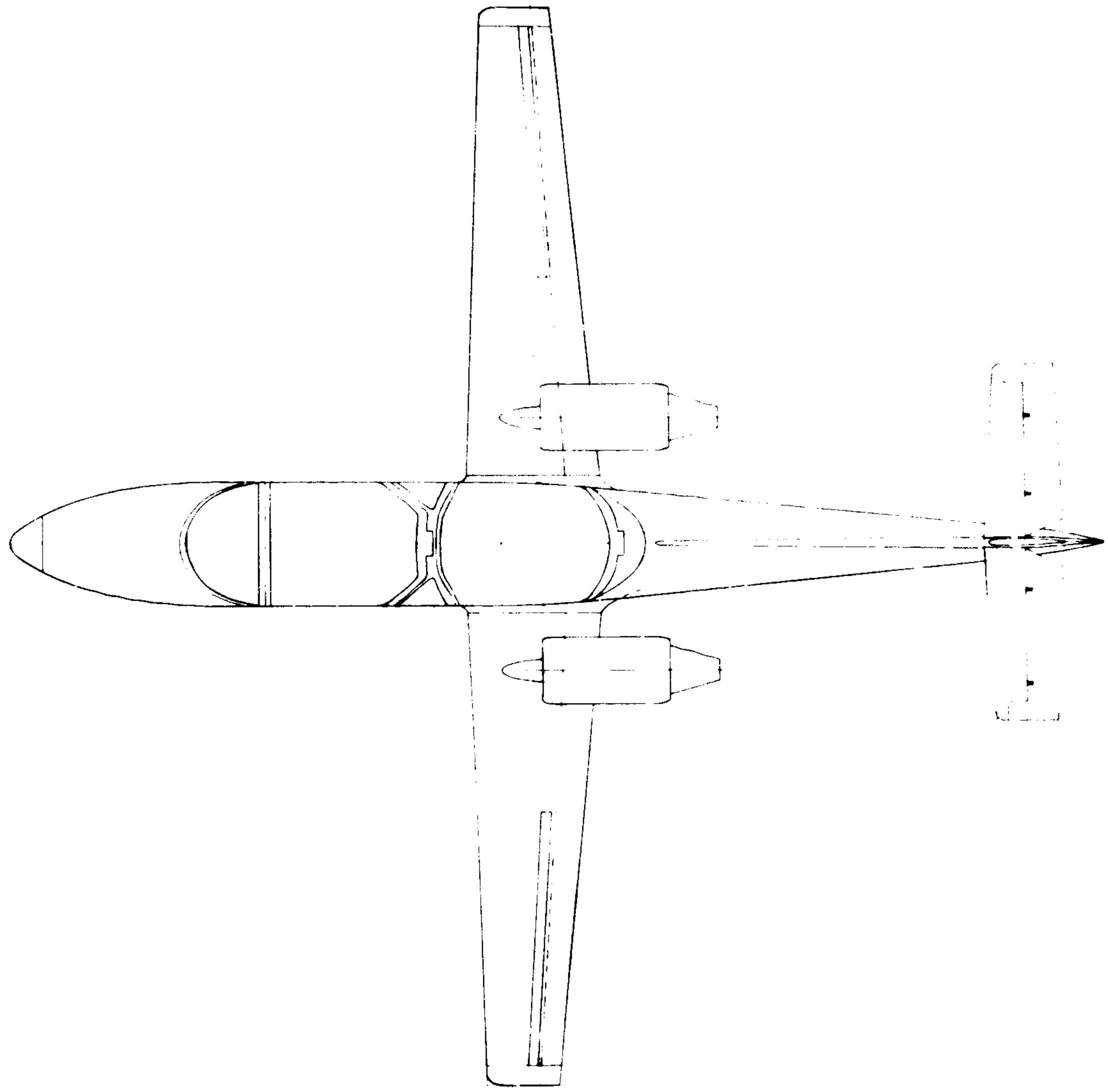
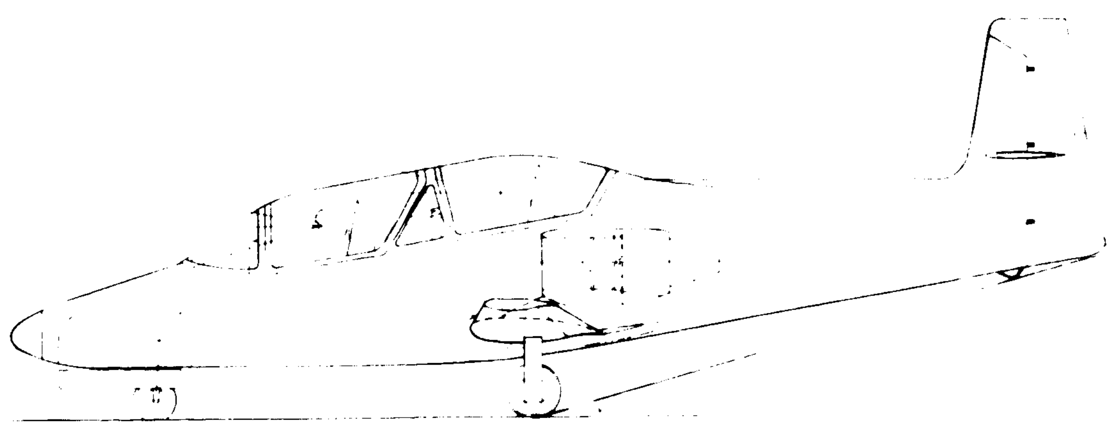
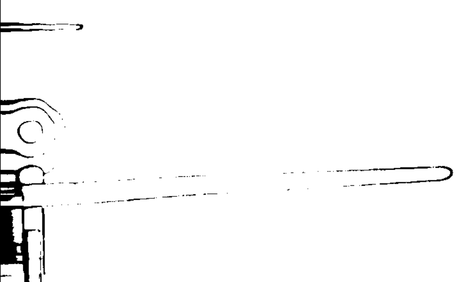






ng Twin-Engine Tandem Seat Configuration.

LAYOUT

C-2

FOLDOUT FRAME 2

PERFORMANCE

CRUISE SPEED (15000 ft ALTITUDE)	250 kt (288 mph · 0.399 Mn)
STALL SPEED (FULL FLAPS)	~60 kt
RANGE (WITH 2 CREW)	~700 nm (+45 min RESERVE)
TAKEOFF DISTANCE	~1560 ft (OVER 35 ft)
LANDING DISTANCE	~1680 ft (OVER 50 ft)
SUSTAINED MANEUVER RATE	2.5 g (AT ~200 kt · 15,000 ft)
MAX RATE OF CLIMB (SL-15,000)	3800-2660 fpm

WEIGHTS

GROSS	4154 lb
EMPTY	2798 lb
OPERATING (INC. ONE CREW)	3008 lb
FIXED USEFUL LOAD	210 lb
PAYLOAD	200 lb
FUEL (MAX)	945 lb (145 gal)
FIXED EQUIPMENT GROUP	736 lb
FLIGHT CONTROLS GROUP	88 lb
STRUCTURES GROUP	1381 lb
PROPULSION GROUP	593 lb

DIMENSIONS & AREAS

FUSELAGE	
LENGTH	29 ft
WIDTH	3.25 ft
HEIGHT	6.5 ft
WING	
AREA	92.3 sq ft
SPAN	30.4 ft
GEOMETRIC MEAN CHORD	3.15 ft
ASPECT RATIO	10 ·
TAPER RATIO	0.5 ·
THICKNESS/CHORD (ROOT & TIP)	0.17 ·
HORIZONTAL TAIL	
AREA	15 sq ft
SPAN	10.01 ft
MEAN CHORD	2.08 ft
ASPECT RATIO	5 ·
THICKNESS/CHORD	0.07 ·
VERTICAL TAIL	
AREA	15 sq ft
SPAN	4.82 ft
MEAN CHORD	3.22 ft
ASPECT RATIO	1.55 ·
THICKNESS/CHORD	0.087 ·
ENGINE NACELLE (BURIED FUSELAGE INSTALLATION)	
LANDING GEAR	
MAIN TIRE SIZE	4.4 × 18
NOSE TIRE SIZE	4.4 × 14
WHEEL BASE	10 ft
WHEEL TREAD	6.67 ft

ENGINE DATA

THRUST (SLS STD. DAY)	1908 lb
AIRFLOW ( )	111 lb/sec
CRUISE THRUST (250 kt/15000 ft)	400 lb (REQD)
WEIGHT (DRY)	407 lb

AERODYNAMIC DATA

WING	45 lb/ft²
CRUISE DRAG POLAR	C_D=0.0282+0.04
LANDING GEAR C_D INCREMENT	0.02721
EFFECTIVE FLAT PLATE AREA	2.602
WETTED AREA	626 sq ft.
MEAN SKIN FRICTION COEFFICIENT	0.00416
CRUISE REYNOLDS NO/FOOT	1.843 × 10^6
C_L MAX (WITH FULL FLAPS)	3.72
HORIZONTAL TAIL VOLUME COEFFICIENT	0.93
VERTICAL TAIL VOLUME COEFFICIENT	0.075

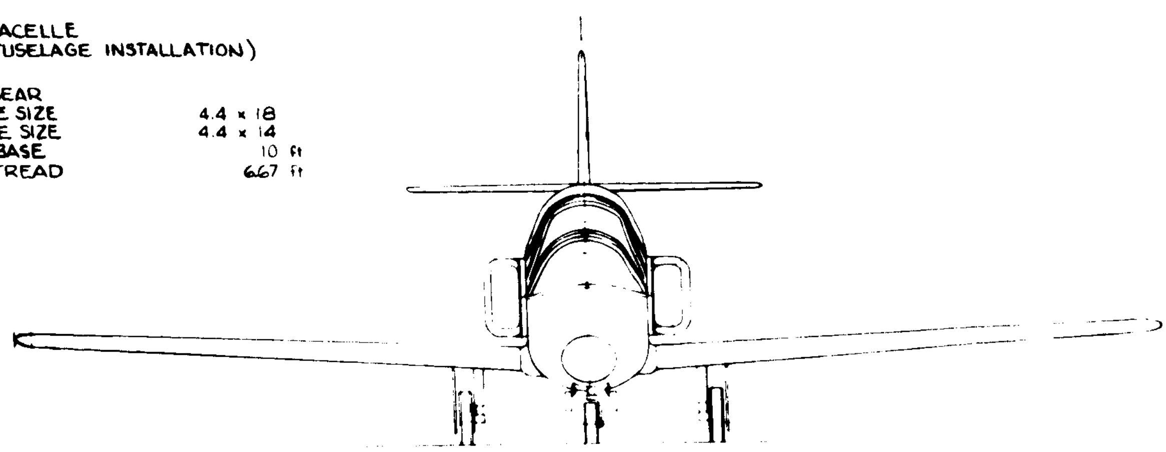


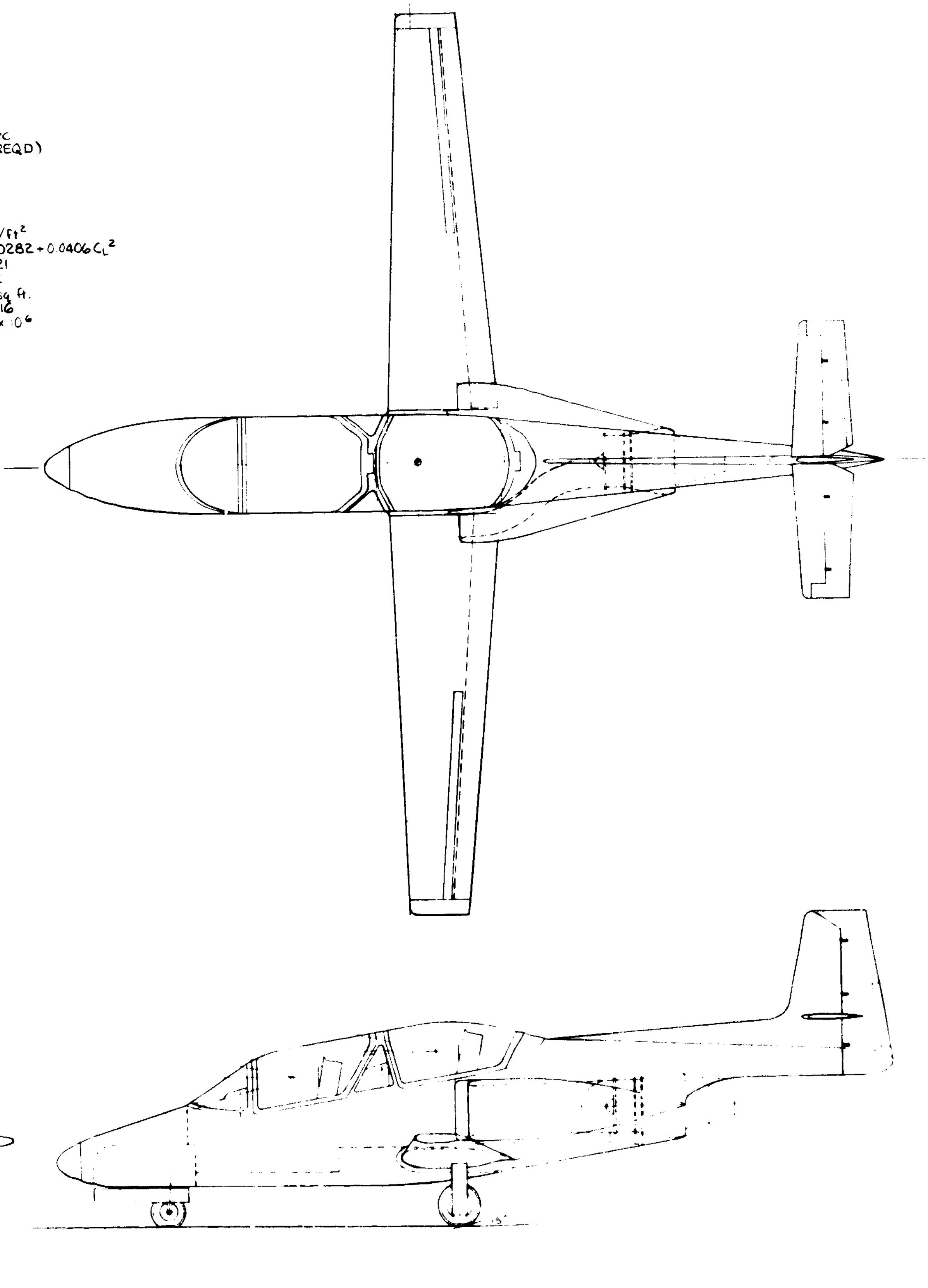
Figure 44. Three-View Drawing Tandem-Seat, Single Engine

TA

(SLS STD DAY)	1908 lb
( )	111 lb/sec
THRUST (250 kt/15000 ft)	400 lb (REQD)
(DRY)	407 lb

IC DATA

	45 lb/ft^2
DRAG POLAR	$C_D = 0.0282 + 0.0406 C_L^2$
GEAR C_D INCREMENT	0.02721
VE FLAT PLATE AREA	2.602
AREA	626 sq ft.
IN FRICTION COEFFICIENT	0.00416
REYNOLDS NO./FOOT	.843 x 10^6
(WITH FULL FLAPS)	3.72
ITAL TAIL VOLUME COEFFICIENT	0.93
L TAIL VOLUME COEFFICIENT	0.075



ndem-Seat, Single Engine Configuration.

PROPRIETARY NOTICE

LAYOUT

Inlet Pressure Loss Sensitivity Analysis

Since the foregoing growth increments can be attributed almost entirely to the compounding effects of a 5 percent inlet loss on engine performance, it was concluded that an inlet loss sensitivity study could be easily completed. By redoing the GASP analysis with use of engine performance data corresponding to an inlet loss of only 1 percent, the impact of inlet loss alone could be assessed. To make this assessment, it was not considered necessary to show how the low-loss inlet would be executed. Table XII shows the pertinent results of this sensitivity analysis. It was estimated that the impact of this inlet loss increment on airplane unit cost could amount to about $12,000. Lifetime fuel consumption differences could be as high as 757,000 liters (200,000 gallons).

As indicated previously, it is characteristic of turbofans with low fan pressure ratios that their performance (both thrust and specific fuel consumption) is extremely sensitive to inlet losses. From this, it may be concluded that a revised cycle, with a higher fan pressure ratio would be better in installations with high losses. Over the range of loss values examined in this study, it was shown that higher fan pressure ratios would raise the uninstalled specific fuel consumption and increase the core size. The installed performance then was no better than it was with the original fan pressure ratio found optimum in this investigation. It is interesting to note that as poor as the tandem-seat, single engine configuration is in comparison with the other airplanes of the study, it is not as heavy nor does it consume as much fuel as the Randolph TA-2 conceptual design.

TABLE XII. SINGLE-ENGINE, TANDEM AIRPLANE GASP RESULTS, WITH TWO VALUES OF ENGINE INLET LOSS.

Inlet Loss	5%	1.0%
Gross weight kg (lb)	1884 (4154)	1679 (3701)
Empty weight kg (lb)	1269 (2798)	1166 (2570)
Engine thrust, N (lb)	8487 (1908)	7384 (1660)
Fuel Consumption liters/hr (gal/hr)	211.6 (55.9)	161.6 (42.7)

ENGINE INSTALLATION STUDIES

Single-Engine Installation

In the description of the tandem-seat, single-engine airplane configuration, the sensitivity of engine performance to duct pressure loss was shown to have a major effect on airplane size. The problem of providing a low-loss engine installation for that configuration was not resolved. Fortunately, with a side-by-side seating arrangement and a shoulder wing location, it was found possible to install a single engine in a manner that avoided compromise to either engine or airplane performance. It was possible to provide a short, nondiffusing, and nearly straight inlet duct, with the result that the inlet duct was estimated to have only one percent pressure loss. A layout drawing was subsequently prepared for this installation.

The merits of this installation are revealed by examining the airplane three-view drawing in Figure 12 and the installation layout drawing in Figure 45. The three-view drawing shows the engine installed at shoulder level--a particular advantage in providing accessibility for routine engine maintenance. Access to the engine is not inhibited by the wing or other airplane structure. A further advantage of this semi-buried installation is that large areas and volumes are available for engine accessories and engine/airframe interfacing hardware. This space is provided in the fuselage tail cone; thus, no drag penalty is incurred.

The installation layout shows the nacelle structure, including large access doors on either side of the engine. Engine accessories, oil tank, and the cabin air bleed line are shown at the bottom of the engine. Engine mounting is accomplished with two trunnions on the engine front frame and a stabilizing link at the rear. Airframe components of the mount system are shown projecting from the "floor" of the nacelle. The ignition exciter box and inlet-lip anti-icing plumbing are easily located on the top half of the engine. Normal engine lines and fittings and the in-coming fuel line are omitted from the drawing, but more than ample space for these items is available.

No difficulties were encountered in achieving this attractive installation. Although small engines are inherently more difficult to install and interface with, this installation appears to be an easy one. It achieves more efficient results than is usually accomplished with large engines.

Twin-Engine Installation

Installation studies were carried out on the wing-mounted, podded engines of both twin-engine configurations. To examine the total engine/airplane interface problem, the pylon and wing section are included in the installation layout drawing given in Figure 46.

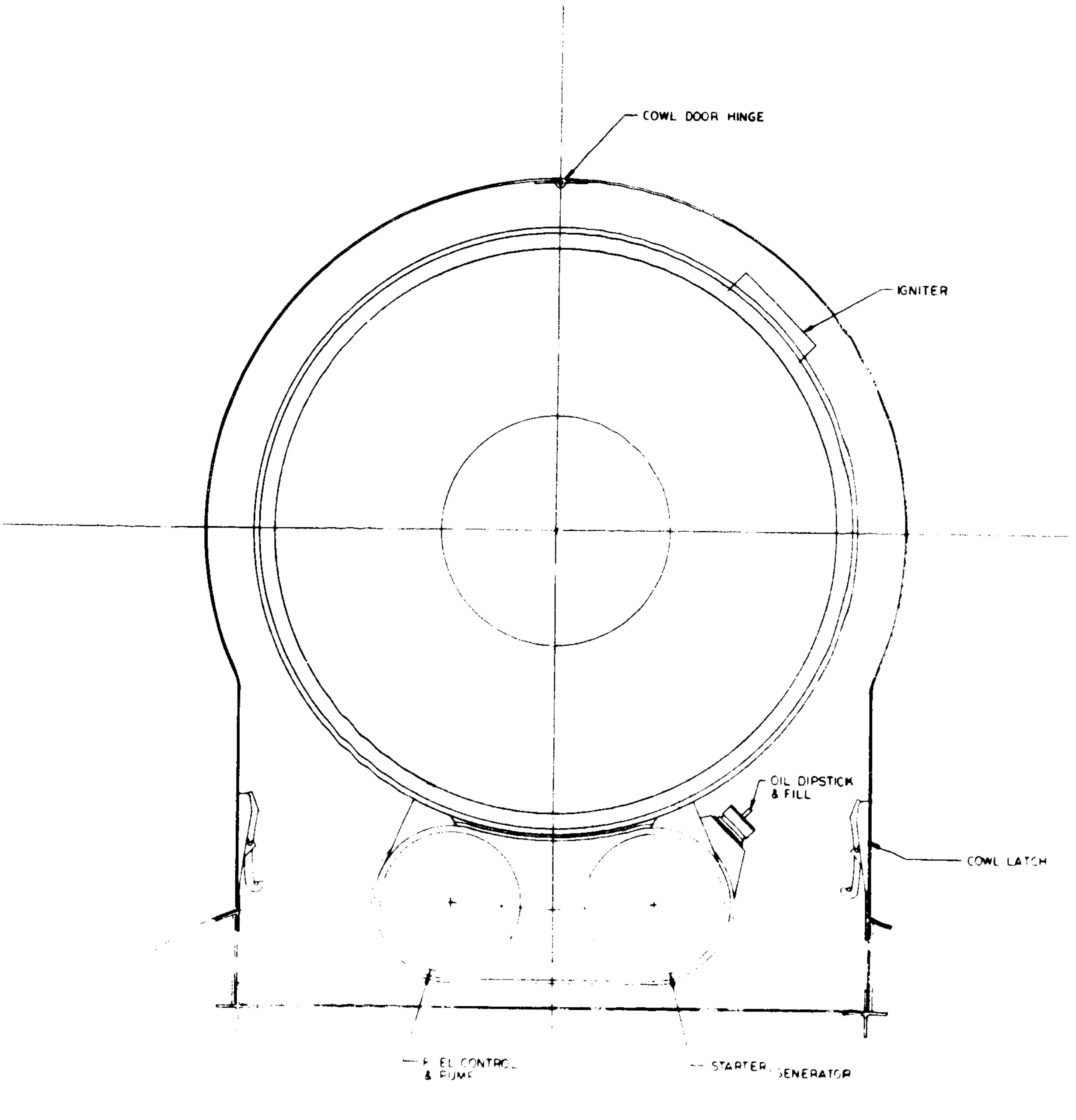
In Cessna's analysis of the aerodynamic effects of over-the-wing installations, the design and wind tunnel test results carried out on the VFW-614 airplane were reviewed in some detail. Based on the data available from this design, location of the nacelle inlet plane at 50 percent of the wing chord produces near-minimum interference drag. In comparison with the VFW-614, the trainer design benefits from an additional effect that lowers interference drag. At the lower design flight speed of the trainer, the ratio of capture area to nacelle frontal area is substantially greater. Thus, super-velocities around the nacelle are reduced, which in turn reduces interference effects.

[illegible] this installation, it is obviously more difficult to achieve a low-drag configuration than was the case with the fuselage-mounted, single engine. In addition to the drag effects of nacelle-pylon-wing interference, there is a greater extent of frontal and wetted areas with which to contend. It was found that nacelle and pylon areas could be minimized by locating engine-mounted accessories in the pylon. With pylon width determined by structural considerations and height by the need to minimize nacelle-wing interference drag, sufficient space is available for this arrangement.

The layout shows the starter-generator, fuel control and pump, hydraulic pump, oil tank, and cabin air bleed line, all outside the nacelle envelope. This permits the engine nacelle to have the smallest dimensions and least possible drag.

The engine mounting is accomplished in a manner similar to that of the single engine installation. Conjectural airframe structure is shown in the nacelle, pylon, and wing, principally to show that space is available for credible load paths in this volume-limited configuration.

In summary, while not as efficient both structurally and with respect to drag as the single-engine installation, the small size of the twin-engine installation does not preclude its being accomplished at least as well as most large turbofan engines. Accessibility for maintenance and engine removal is outstanding, and with proper aerodynamic tailoring, it should be possible to achieve a low installed drag.



SECTION

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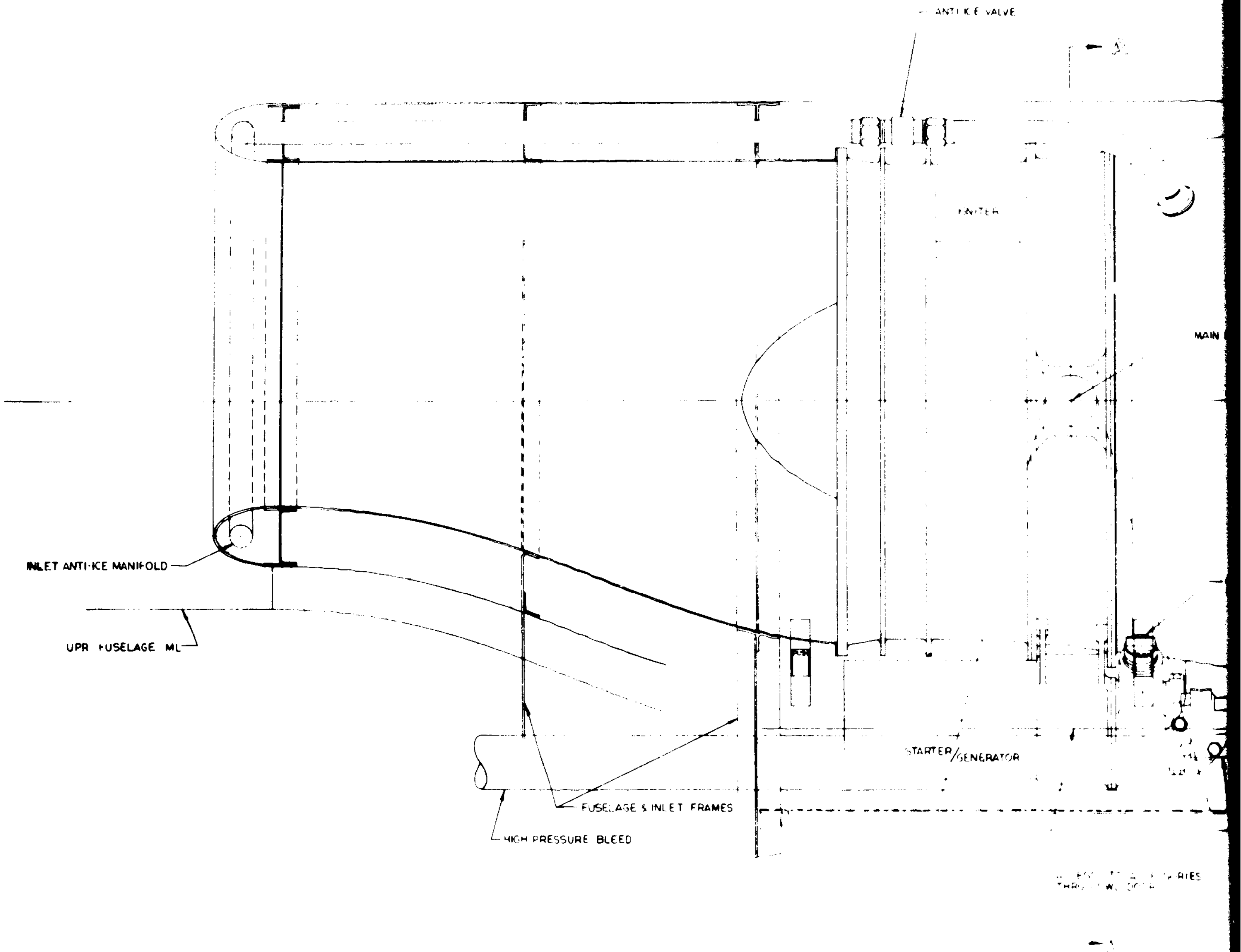
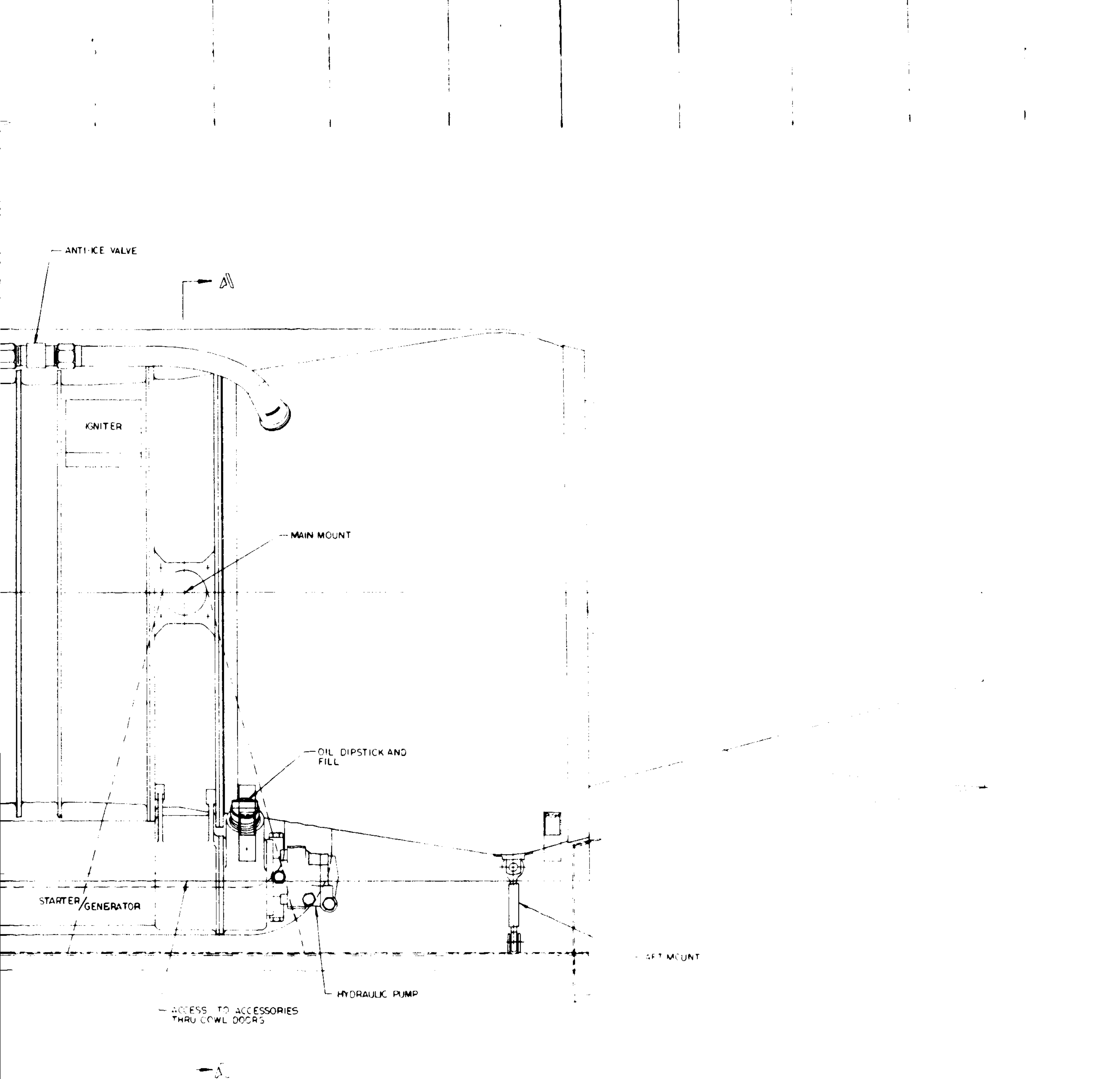
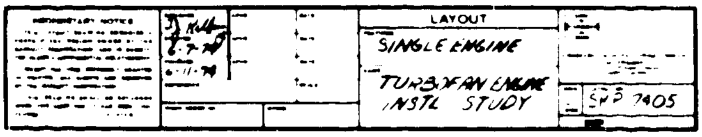


Figure 45. Single-Engine Installatio

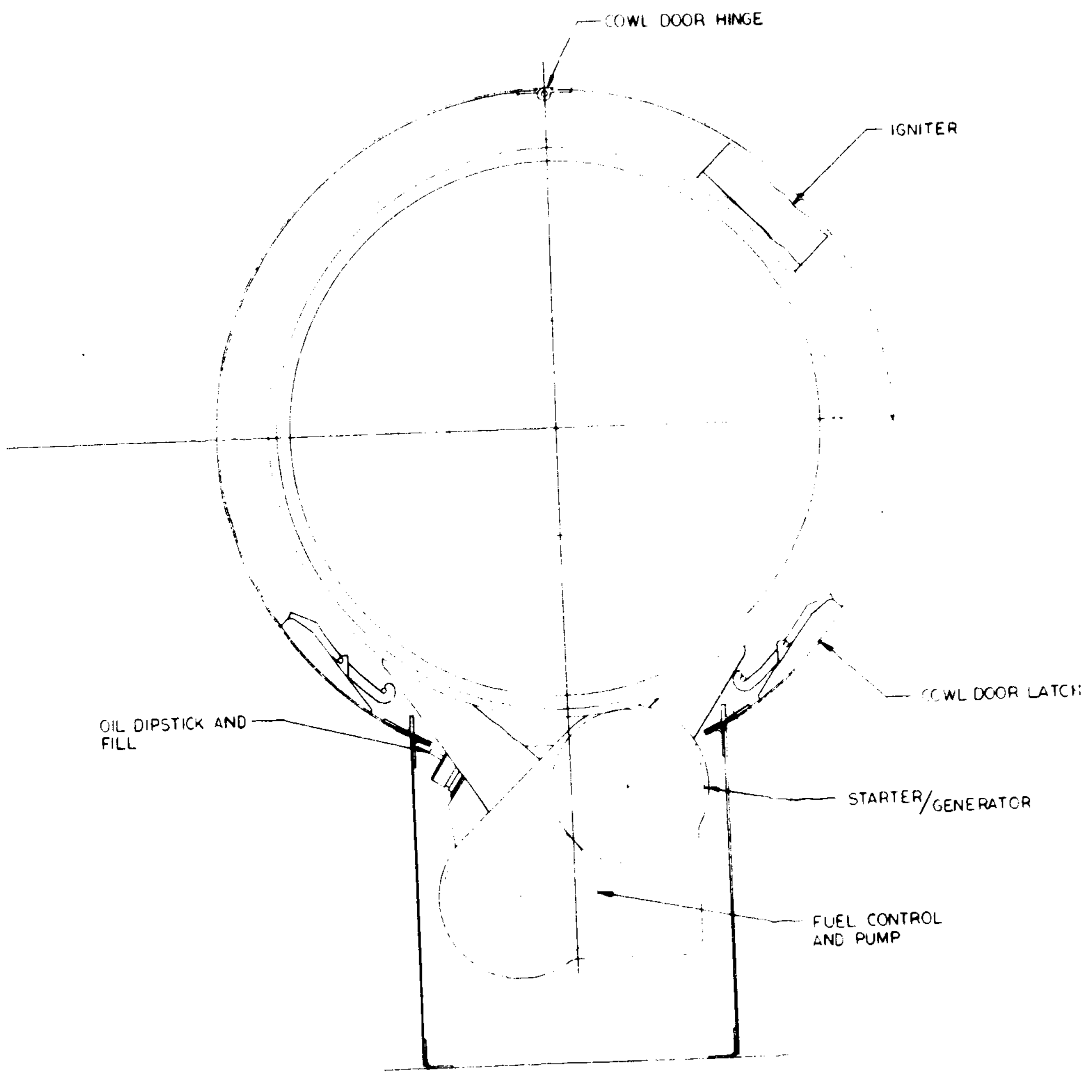
FOLDOUT FRAME



Single-Engine Installation Layout.



FOLDOUT FRAME 2



SECTION B - B

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SKP 17403

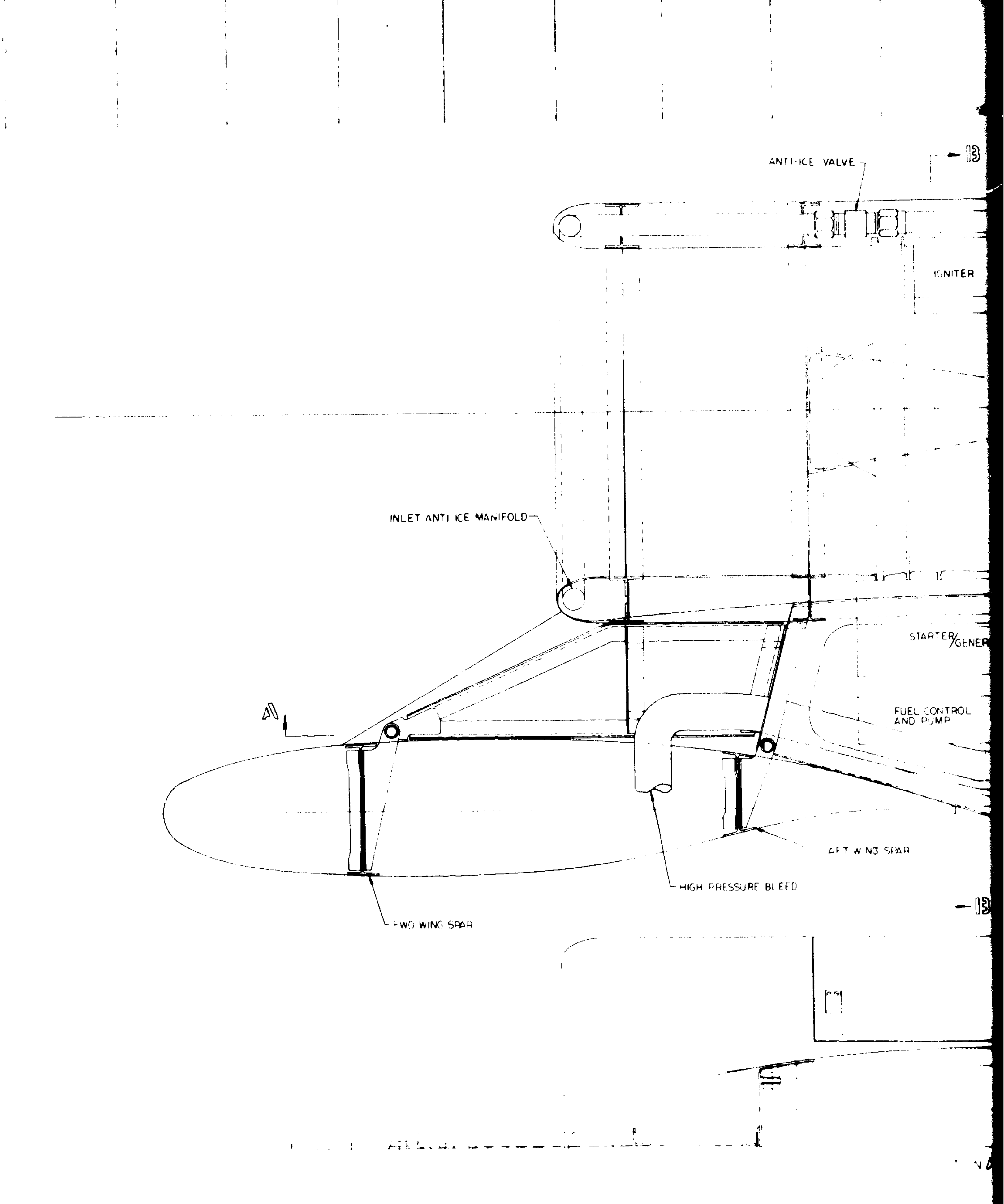

Figure 46. Twin-Engine Installation La

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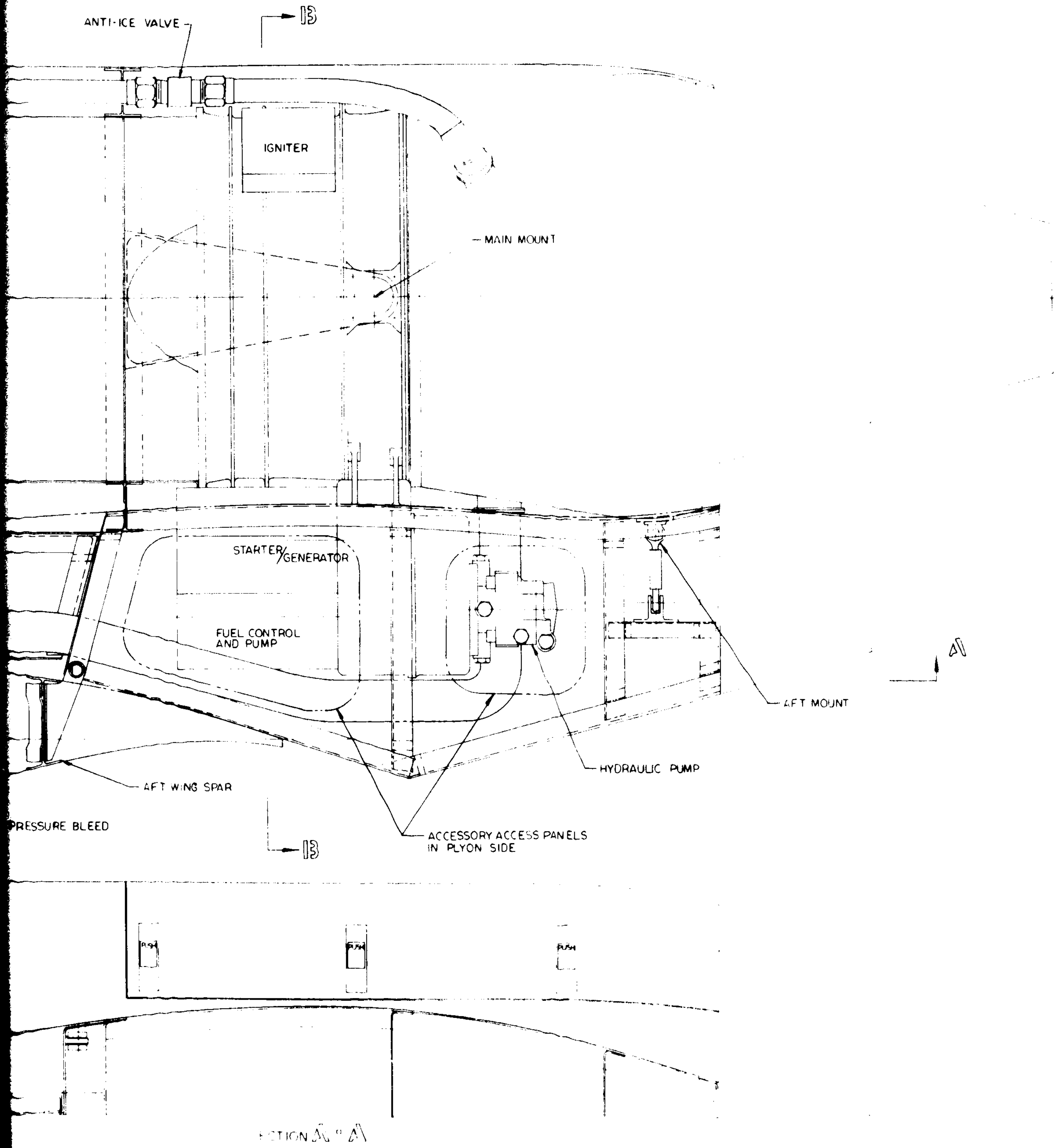

n-Engine Installation Layout.

PROPRIETARY NOTICE			LAYOUT
	6-3-74		TWIN ENGINE TRAINER
	6-6-74	1/2	TURBOFAN ENGINE INSTL - STUDY
			J SKP 17403

FOLDOUT FRAME 2

CONCLUSIONS AND RECOMMENDATIONS

It was demonstrated under the original work statement of this contract, that turbofan engines could be both efficient and cost-effective when installed on small, high-performance, general-aviation airplanes. It has been the purpose of this continuation of the study to determine if the trend continues in applications to still smaller, and lower-performance airplanes. The principal goal has been to determine if small turbofans in the general-aviation class studied were equally attractive in performance and economics, when applied to military trainers. If this could be demonstrated, it could lead to eventual military procurement of turbofans in this class. In turn, this would hasten their availability to general-aviation where the social qualities and economic advantages of turbofans would be welcomed.

Throughout the course of the study, it was felt by the investigators that significant accomplishments were being demonstrated with respect to factors affecting economics. Late in the program, attempts were made to encourage military agencies to participate in life-cycle cost analyses in order to confirm these tentative accomplishments. Although these attempts failed, sufficient data were gathered to permit a brief, but meaningful, economic study to be performed. Since this study lacked the technical rigidity desirable, a discussion of its results is not given in the body of this report, but is provided in Appendix C.

Early in the program, the 1973-74 "energy crisis" impacted on energy-consuming citizens of the world, including those responsible for military operations. The pertinency of this study was made clear by the press reports that, in deference to the shortage of aviation fuel, entire classes were being dropped or stretched out in military undergraduate pilot training programs. It was also clear that, if the results anticipated from this study were confirmed, a small contribution would have been indicated to the solution of the "energy crisis."

It should be pointed out that a great amount of energy is consumed in the preparation of raw materials and in the manufacture of engines and airplanes. Minimizing their size by maximizing their efficiency is then an energy conservation method as important as minimizing in-flight fuel consumption. For this reason, analysis was performed throughout the study in a manner that yielded results that could be given in terms of airplane and engine sizes, as well as fuel consumption.

The study has concentrated on cycle selection and design of the turbofan engine. However, it is significant that the new general-aviation wing design being developed by the NASA was used throughout the study, and the benefit to results was taken for granted. With this wing currently undergoing development and demonstration, taking it for granted may be hazardous. However, it is concluded that complete success in this demonstration is fundamental to achieving a significant step forward in the design of light airplanes.

Also it should be pointed out that the NASA developed aircraft synthesis computer program, GASP, was again used to define engine/airplane/performance interrelationships and to evaluate design results. Proof of its value was demonstrated. It is concluded that its use is indispensable in defining "best" engine and airplane solutions.

From the study results described in this report, the investigators have drawn the following specific conclusions:

1. It has been shown for the second time that when care is taken to define a "best" turbofan propulsion system for a specified airplane flight envelope and mission, the synergistic effects of a "best" engine design yield a surprisingly small and efficient airplane solution.

2. The airplane size and fuel consumption results obtained in this study confirm that at low design flight speed, the "best" turbofan has high overall propulsion system efficiency. It is thought to be greater than propulsion systems using propellers when all installation and synergistic effects are accounted for.

3. The engine performance quality found to be "best" in this study can be attained without recourse to high-pressure, high-temperature, advanced-technology cycles. The design and development of these turbofans need not await the invention and demonstration of the technology features generally thought applicable to future gas turbine engines.

4. The low-noise, low-emissions characteristics of the engine designs addressed in this study would provide a "bonus" fallout of improved social acceptability for under-graduate pilot training operations with no economic penalty.

Finally, tentative conclusions can be made relating to the primary objective of this study, namely, the applicability of small civil turbofan engines to military trainer aircraft. By adhering strictly to the basic trainer specifications set down by the Air Force in their under-graduate pilot training study, an aircraft-engine combination has been derived which is smaller, less costly and more efficient than either current basic trainer aircraft or those that have been conceptually designed with advanced high-subsonic engines. This leaves open the question of other approaches to military pilot training such as greater use of simulators or one aircraft designed for both basic and advanced training.

The engines defined in this study could have widespread use in civil light aircraft because of their high efficiency, low noise and low emissions. A remaining question is cost; and substantial benefits may accrue to both military and civil sectors if the engine commonality established in this study could be exploited.

APPENDIX A

CESSNA FINAL REPORT ON GARRETT AIRESEARCH SMALL TURBOFAN-POWERED TRAINER DESIGNS

Cessna Wallace Division, under subcontract to Garrett AiResearch, has acted as a consultant to critique small turbofan military trainer designs developed by Garrett and to provide Cessna experience in the area of trainer requirements. Trainer designs were critiqued in the areas of weight, balance, stability, control, and general performance. Modifications to the designs were suggested, along with corrections of some of the performance data. Basic data on the T-37 and on the latest technology airfoils was supplied to help achieve the most efficient performance and state-of-the-art projections. Cessna representatives also participated in the interim oral review at Garrett and in trips to Air Force and Navy offices to help determine future customer requirements, trends, and interest in the aircraft types under study.

The Garrett configurations provide a new trainer concept with good potential for reducing operating cost and unit cost. The fuel availability and cost situation is generating the need for further consideration of small turbofan engines.

The Garrett configurations should be less expensive to manufacture and to maintain than current larger trainers. They do offer design challenges since they are based on projections in the state-of-the art aerodynamics as well as propulsion systems. Their suitability in the training role will require further study outside the limited scope of the current program. Additional detail studies should include design effects on aerobatic characteristics; the number of takeoff and landings available; range profiles; training profiles; stability and control dynamics; fuselage lift influences; inertia distributions; spin characteristics; and airspace allotment requirements.

Critique on Single-Engine Configuration.

1. Tip tanks are not considered necessary, since the fuel volume required is available in the wings. Tip tanks and fuel increase rolling inertia (potentially undesirable in spins) and a trend towards roll/yaw dynamic coupling.

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APPENDIX A

2. Higher aspect ratio vertical tails may be desirable, particularly for spin recovery. It is desirable to have a considerable difference between roll and pitch inertia for good spin recovery. Higher pitch inertia goes with rudder recovery. Higher roll inertia goes with elevator recovery. Higher pitch inertia is recommended, since one of the vertical tails will be fully exposed during a spin.

3. Wing stall will begin in the trailing-edge root area, producing low dynamic pressure and possibly reverse flow. The engine will have to tolerate this flow during stalls. Lowering the wing or raising the engine intake slightly would reduce the problem. Other than stall, airflow over the canopy can be made to remain attached (similar to the T-37 trainer).

4. If the main landing gear should utilize spring struts similar to Cessna single-engine landing gears, the gear height should be sufficient to avoid banging the belly during very hard landings by students.

5. The tail-cone contours and exhaust fairing will require special attention to preclude exhaust attachment along the tail cone, leading to elevator problems.

6. The use of spoilers only for roll control on a fully aerobatic trainer airplane is feasible. However, some development testing will be required to perfect manually powered aerodynamically balanced spoilers.

7. Cessna's calculation of airplane weight is within 50 pounds of the Garrett computer-calculated weight. There are greater differences in group weight distributions, but Cessna's check provides the credibility necessary within the scope of this study. Much of the equipment has been pared down in weight commensurate with state of the art projections. The total integrated airplane follows this philosophy, and it should not be considered as utilizing "off-the-shelf" engineering or equipment.

8. A tandem seating arrangement would require additional avionics weight. It is assumed that this could be done with extra control heads and switches.

9. The avionics package was based on those functions required by the Randolph AFB UPT-study. Future training needs could conceivably require additions to this list such as weather radar, microwave landing equipment, autopilot, very low frequency navigation, and equipment for flight-into-known-icing conditions. Since the Garrett design philosophy includes a fully optimized and integrated trainer for the greatest training efficiency, it is very important to establish future equipment requirements by close coordination with the military for finalizing a new trainer design. A small aircraft design is fairly sensitive to changes in fixed equipment requirements.

10. Additional comments in the twin-engine trainer critique, where applicable, apply to the single-engine configuration, and vice versa.

Critique on Twin-Engine Configuration - The twin-engine, side-by-side seating trainer configuration received the greatest study verification effort by Cessna. Those items studied included weight, balance, stability, wing aerodynamics, empennage aerodynamics, nacelle aerodynamics, landing distance, takeoff distance, stall speeds, drag, and cruise performance.

1. Cessna's calculation of empty weight is about 300 pounds greater than the Garrett computer analysis, most of which is in fixed equipment. The computer employs percentages to establish group weights. This is not believed to be totally realistic as an airplane is reduced in size since some fixed equipment must remain the same. Garrett has run a sensitivity analysis to test the impact of weight variation on the overall aircraft, which has proven an excellent tool to judge necessary changes in preliminary engineering.

2. For a balance check, avionics was placed in the panel and nose, battery and oxygen in the nose, and hydraulics and air-cycle aft of the pressurized cockpit. By assuming a most aft center-of-gravity of 30 percent mean aerodynamic chord, it was determined that the engines should be moved aft about 15 inches. This engine placement is also compatible with Cessna's recommendation for reducing engine interference drag. The oxygen bottle was moved to the tail cone to reduce fire hazard.

3. Volume for equipment is satisfactory. Fuel volume and tank configuration included a wet tank integral with the wing and additional fuel in the wing carry-through. With one integral tank per side, the fuel system and fuel management should be very simple.

4. Static stability was checked, including a larger vertical tail to lower engine-out minimum control speed to near the stall speed. Recommended empennage changes were incorporated by Garrett. The larger vertical tail will also improve spin recovery characteristics.

5. There is some concern about Reynolds-number effects due to the rather small wing chords. The latest NASA low-speed airfoils are used but have not been tested below about 2 million Reynolds number. Low-speed lift characteristics near the stall and during low dynamic-pressure maneuvers may be erratic or at least require airfoil tailoring by wind tunnel and flight testing research. The new NASA airfoils are considered desirable due to their high L/D characteristics at high lift coefficients.

6. Due to airframe equipment power requirements, starter-generators, alternators, hydraulics, and engine bleed for pressurization-heating-cooling will require about the same engine gearbox power and pad ratings, and bleed flow as today. It would be desirable in future small-engine studies, possibly under NASA sponsorship, to study integration of some power systems directly in the engine.

7. Recommendations were made to relocate the engines for drag reduction. Nacelle-fuselage interference drag could be reduced by moving the nacelles outboard to the wing planform break. The Garrett nacelle inlet is located at about 15.5 percent of the local wing chord. Moving the intake back to about 70 percent wing chord would reduce the nacelle drag coefficient by at least 25 percent. This is based on VFW-614 studies. Further aft movement and changing to fuselage pylon-mounted nacelles would reduce drag even further, although balance would become a problem. Raising the nacelles about 2.5 inches would further reduce nacelle-wing interference drag. Shortening the nacelle intake seems feasible and would reduce the amount of engine movement aft for balance considerations. Moving the engines aft 15 inches and

shortening the intake would decrease overall airplane drag by approximately 5 percent at the 250-knot/15,000-foot cruise point.

8. Garrett's calculation of thrust required at the cruise point (250 knots at 15,000 feet) was 167 pounds per engine. Cessna's drag buildup calculation, including nacelle position changes, yielded 212 pounds per engine. This difference was not resolved, since Cessna calculations are based on cross-section areas, interference drags, and drag due to lift, while the Garrett calculations are based on wetted area. It is suspected that most of the difference is in the compensation for interference drag based on Cessna's Citation experience and data from VFW-614 studies.

9. Stall speed with full flaps calculated by Garrett was 52 knots while Cessna calculated it to be 67.5 knots. The increase is due to Cessna's reduction in maximum coefficient of lift due to Reynolds number, wing planform, and adjusted exposed wing-area effects. The higher stall speed is not considered critical for a trainer.

10. Takeoff distance over a 35-foot obstacle was calculated by Garrett as 1564 feet and by Cessna as 1612 feet. This is well within the band of accuracy for this type of study.

11. Landing distance over a 50-foot obstacle as calculated by Garrett was 1676 feet and by Cessna was 2155 feet due to Cessna's higher stall speed calculation.

12. The wing-body-nacelle juncture and shapes may produce local flow separation, causing drag, potential tail buffet, and loss of elevator effectiveness. This area will require considerable attention to nacelle placement, wing fillets, and local area ruling. These details do not affect the credibility of the configuration as representative of the aircraft class.

13. Dynamic response characteristics of the configuration should not be a problem.

14. A relatively detailed analysis of wing downwash characteristics was made to provide data for empennage configuration recommendations supplied to Garrett. Static directional and longitudinal stability estimates were made to establish empennage recommendations. Single-engine minimum control speed was calculated and used to recommend vertical tail size. The stick fixed neutral point was calculated to be at 60 percent mean aerodynamic chord in cruise.

15. Fuselage planform is nearly the same as wing area. Final analysis should include fuselage lift effects in addition to the wing.

Conclusion - The Garrett trainer configurations are good representatives of the aircraft class for the scope of this study. This class of aircraft should be less expensive to manufacture and to maintain. It does offer design challenges. The configuration concepts, due to size versus performance, are in an area not yet explored for use in a training role. However, the economic advantages justify a close look at small turbofan-powered aircraft.

In this small size and weight class, fixed equipment becomes a higher percentage of empty weight, packaging is tight due to volume limitations, fuselages have more relative effect on lift and drag, Reynolds numbers become more critical due to small wing chords, and weight reduction may be limited by practical skin thicknesses rather than strength requirements. As a result, there will be a need for future airframe as well as engine research, development, and testing.

Small turbofan engine power offers very good fuel economics, configuration design flexibility, low noise, minimum vibration, very limited pollution emissions, good ground safety, and inherent reliability/maintainability.

Limited knowledge in some aerodynamic areas as they apply to a trainer and its mission will require further investigation. Low Reynolds-number effects may influence normal stall characteristics, aerobatic maneuvers, and spin sensitivity. Fuselage size effects relative to the wing and empennage will require close investigation. Moderately high wing loadings may pose problems for primary training. "Spoilers only" for roll control on a fully aerobatic trainer will require some development testing. Close attention must be paid to equipment design and weight, and to justify any secondary airplane mission functions that may impact primary mission efficiency.

The NASA computer synthesis airplane design program has proven to be an excellent tool for configuration fundamentals and trade-off studies. Final design finesse is still required but much time is saved in integrated optimization studies.

The Garrett-Airesearch trainer designs are feasible and offer considerable operating cost reductions. The configurations are pared down to do one specific job, which is training efficiency. As a result, early firm requirements of desired equipment functions and mission requirements must be made to preclude oversizing the aircraft for growth during development or for future growth. Airframe, aerodynamics, and equipment research and development will be required in addition to the engine development.

APPENDIX B

GASP Printout For
Side-By-Side Single-Engine Solution Airplane

VDIVE = 340. KTS VMO = 289. KTS MMO = .792
ULT. LF = 9.00 MAN. LF = 6.00 GUST LF = 2.98

PROPULSION GROUP		
PRIMARY ENGINES	(WEP)	322.
PRIMARY ENGINE INSTL.	(WPEI)	80.
FUEL SYSTEM	(WFSS)	57.
PROPULSOR WEIGHT	(WPROP)	0.
TOTAL PROP.GROUP WT.	(WP)	459.
STRUCTURES GROUP		
WING	(WW)	380.
HOR. TAIL	(WHT)	52.
VERT. TAIL	(WVT)	26.
FUSELAGE	(WB)	545.
LANDING GEAR	(WLG)	166.
PRIMARY ENG. SECTION	(WPES)	76.
GROUP WEIGHT INC.	(DELWST)	0.
TOTAL STRUC.GROUP WT.	(WST)	1246.
FLIGHT CONTROLS GROP		
COCPIT CONTROLS	(WCC)	33.
FIXED WING CONTROLS	(WCFW)	17.
SAS	(WSAS)	0.
GROUP WEIGHT INC.	(DELWFC)	29.
TOTAL CONTROL WT.	(WFC)	79.
WT. OF FIXED EQUIPMENT	(WFE)	669.
WEIGHT EMPTY	(WE)	2453.
FIXED USEFUL LOAD	(WFUL)	210. (INC. CREW OF 1)
OPERATING WEIGHT EMPTY	(OWE)	2663.
PAYLOAD	(WPL)	200. (PAX= 1.)
FUEL	(WFA)	635. (WFW= 635.) (WFTP= 0.)
GROSS WEIGHT	(WG)	3497.

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APPENDIX B

Side-By-Side Single (Cont'd)

GROSS WEIGHT = 3497 PASSENGERS = 1. PLUS CREW OF 1

FUSELAGE	LENGTH	(ELF)	26.65	FT
	WIDTH	(SWF)	4.67	FT
	WETTED AREA	(SF)	282.	SQFT
	DELTA P	(DELP)	4.20	PSI
WING	ASPECT RATIO	(AR)	10.00	
	AREA	(SW)	77.7	SQFT
	SPAN	(B)	27.9	FT
	GEOM. MEAN CHORD	(CBARW)	2.81	FT
	QUARTER CHORD SWEEP	(DLMC4)	0.0	DEG
	TAPER RATIO	(SLM)	.750	
	ROOT THICKNESS	(TCR)	.170	
	TIP THICKNESS	(TCT)	.170	
	WING LOADING	(WGS)	45.0	PSF
	WING FUEL VOLUME	(VFW)	23.8	CUFT
HOR. TAIL	ASPECT RATIO	(ARHT)	4.57	
	AREA	(SHT)	18.5	SQFT
	SPAN	(BHT)	9.19	FT
	MEAN CHORD	(CBARHT)	2.08	FT
	THICKNESS/CHORD	(TCHT)	.070	
	MOMENT ARM	(ELTH)	13.3	FT
	VOLUME COEFF.	(VBARH)	1.123	
VERT. TAIL	ASPECT RATIO	(ARVT)	1.24	
	AREA	(SVT)	12.2	SQFT
	SPAN	(BVT)	3.89	FT
	MEAN CHORD	(CBARVT)	3.26	FT
	THICKNESS/CHORD	(TCVT)	.087	
	MOMENT ARM	(ELTV)	13.2	FT
	VOLUME COEFF.	(VBARV)	.075	
ENG.NACELLES	LENGTH	(ELN)	4.83	FT
	MEAN DIAMETER	(DBARN)	.71	FT
	NUMBER ENGINES	(ENP)	1.0	
	WETTED AREA	(SN)	10.77	SQFT

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APPENDIX B

Side-By-Side Single (Cont'd)

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CRUISE RE.NUM. PER FT.    = 1.843E+06    FLATPLATE CF AT RE=10**7 IS  .00287

AERODYNAMICS DATA
     TOTAL EFFECTIVE FLATPLATE AREA    (FE)         2.108    SQFT
     TOTAL WETTED AREA                 (SWET)       494.5    SQFT
     MEAN SKIN FRICTION COEFF.         (CFBARF)    .00426

DRAG BREAKDOWN IN SQFT
     WING                              (FEW)         .684
     FUSELAGE                          (FEF)        1.103
     VERT. TAIL                        (FEVT)        .109
     HOR. TAIL                         (FEHT)        .170
     ENGINE NACELLES                   (FEN)         .010
     TIP TANKS                         (FETP)       0.000
     INCREMENTAL FE                    (DLTAFE)      .031

AERODYNAMIC COEFF.
     A1                                             .6840
     A2                                            -.1156
     A3                                             .0782
     A4=.75*(T/C)                                   .1275
     A5=CDO--                                       .0183
     A6                                            3.0644
     A7=1/(PI*SEE*AR)                               .0406
     3-D LIFT SLOPE AT CRUISE MACH     (CLALPH)    5.5188    PER RADIAN
     OSWALD FACTOR                     (SEE)        .7646

CRUISE  CD =  .0271 +   .0406 CL**2
LANDING GEAR CD INCREMENT= .02816

*****START OF INPUT FOR CONTROL

END OF INPUT DATA, JOB COMPLETE.
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OF POOR QUALITY

APPENDIX B

GASP Printout For
Side-By-Side Twin-Engine Airplane Solution

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VDIVE =  340. KTS   VMO =  289. KTS   MMO =  .792
ULT. LF =  9.00   MAN. LF =  6.00   GUST LF =  2.98

PROPULSION GROUP
   PRIMARY ENGINES            (WFP)        318.
   PRIMARY ENGINE INSTL.      (WPEI)        79.
   FUEL SYSTEM                (WFSS)        58.
   PROPULSOR WEIGHT          (WPROP)         0.
   TOTAL PROP.GROUP WT.       (WP)         455.

STRUCTURES GROUP
   WING                       (WW)         275.
   HOR. TAIL                  (WHT)         57.
   VERT. TAIL                 (WVT)         32.
   FUSELAGE                   (WB)         459.
   LANDING GEAR               (WLG)        160.
   PRIMARY ENG. SECTION       (WPES)       127.
   GROUP WEIGHT INC.          (DELWST)       0.
   TOTAL STRUC.GROUP WT.      (WST)       1110.

FLIGHT CONTROLS GROP
   COCPIT CONTROLS            (WCC)         33.
   FIXED WING CONTROLS        (WCFW)        16.
   SAS                        (WSAS)         0.
   GROUP WEIGHT INC.          (DELWFC)      29.
   TOTAL CONTROL WT.          (WFC)         78.

WT. OF FIXED EQUIPMENT        (WFE)        669.

WEIGHT EMPTY                  (WE)        2311.

FIXED USEFUL LOAD             (WFUL)       210. (INC. CREW OF 1)

OPERATING WEIGHT EMPTY        (OWE)       2521.

PAYLOAD                       (WPL)        200. (PAX=  1.)

FUEL                          (WFA)        645. (WFW=  645.) (WFTP=    0.)

GROSS WEIGHT                  (WG)        3366.
```

APPENDIX B

Side-By-Side, Twin (Cont'd)

GROSS WEIGHT = 3366. PASSENGERS = 1. PLUS CREW OF 1

FUSELAGE	LENGTH	(ELF)	25.00	FT
	WIDTH	(SWF)	4.67	FT
	WETTED AREA	(SF)	261.	SQFT
	DELTA P	(DELP)	4.20	PSI
WING	ASPECT RATIO	(AR)	10.00	
	AREA	(SW)	74.8	SQFT
	SPAN	(B)	27.3	FT
	GEOM. MEAN CHORD	(CBARW)	2.84	FT
	QUARTER CHORD SWEEP	(DLMC4)	0.0	DEG
	TAPER RATIO	(SLM)	.500	
	ROOT THICKNESS	(TCR)	.170	
	TIP THICKNESS	(TCT)	.170	
	WING LOADING	(WGS)	45.0	PSF
	WING FUEL VOLUME	(VFW)	24.5	CUFT
HOR. TAIL	ASPECT RATIO	(ARHT)	4.57	
	AREA	(SHT)	21.3	SQFT
	SPAN	(BHT)	9.87	FT
	MEAN CHORD	(CBARHT)	2.24	FT
	THICKNESS/CHORD	(TCHT)	.070	
	MOMENT ARM	(ELTH)	11.2	FT
	VOLUME COEFF.	(VBARH)	1.123	
VERT. TAIL	ASPECT RATIO	(ARVT)	1.24	
	AREA	(SVT)	13.0	SQFT
	SPAN	(BVT)	4.01	FT
	MEAN CHORD	(CBARVT)	3.36	FT
	THICKNESS/CHORD	(TCVT)	.087	
	MOMENT ARM	(ELTV)	11.8	FT
	VOLUME COEFF.	(VBARV)	.075	
ENG.NACELLES	LENGTH	(ELN)	4.83	FT
	MEAN DIAMETER	(DBARN)	1.19	FT
	NUMBER ENGINES	(ENP)	2.0	
	WETTED AREA	(SN)	36.11	SQFT

APPENDIX B

Side-By-Side, Twin (Cont'd)

```
CRUISE MACH =  .399          CRUISE ALTITUDE =  15000.

CRUISE RE.NUM. PER FT.    = 1.843E+06    FLATPLATE CF AT RE=10**7 IS  .00287

AERODYNAMICS DATA
    TOTAL EFFECTIVE FLATPLATE AREA     (FE)        2.180    SQFT
    TOTAL WETTED AREA                  (SWET)      498.2    SQFT
    MEAN SKIN FRICTION COEFF.          (CBARF)    .00438

DRAG BREAKDOWN IN SQFT
    WING                               (FEW)        .658
    FUSELAGE                           (FEF)       1.046
    VERT. TAIL                         (FEVT)       .115
    HOR. TAIL                          (FEHT)       .193
    ENGINE NACELLES                    (FEN)        .138
    TIP TANKS                          (FETP)      0.000
    INCREMENTAL FE                     (DLTAFE)     .030

AERODYNAMIC COEFF.
    A1                                             .6850
    A2                                            -.1158
    A3                                             .0781
    A4=.75*(T/C)                                   .1275
    A5=CDO--                                       .0204
    A6                                            3.0591
    A7=1/(PI*SEE*AR)                               .0406
    3-D LIFT SLOPE AT CRUISE MACH      (CLALPH)   5.5188    PER RADIAN
    OSWALD FACTOR                      (SEE)       .7846

CRUISE  CD =  .0291 +   .0406 CL**2
LANDING GEAR CD INCREMENT= .02838

*****START OF INPUT FOR CONTROL
    NPC=2, NSC=6, IDC=0,
    *

*****START OF INPUT FOR MISSION DESCRIPTION
    ICOND=0, ISEG=1,   *
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APPENDIX B

GASP Printout For
Tandem; Twin-Engine Airplane Solution

```
VDIVE =  340. KTS    VMO =  289. KTS    MMO =  .792
ULT. LF =  9.00   MAN. LF =  6.00   GUST LF =  2.98

PROPULSION GROUP
   PRIMARY ENGINES              (WEP)         349.
   PRIMARY ENGINE INSTL.        (WPEI)         86.
   FUEL SYSTEM                  (WFSS)         64.
   PROPULSOR WEIGHT            (WPROP)          0.
   TOTAL PROP.GROUP WT.         (WP)          499.

STRUCTURES GROUP
   WING                         (WW)          340.
   HOR. TAIL                    (WHT)          53.
   VERT. TAIL                   (WVT)          40.
   FUSELAGE                     (WB)          463.
   LANDING GEAR                 (WLG)         173.
   PRIMARY ENG. SECTION         (WPES)        137.
   GROUP WEIGHT INC.            (DELWST)        0.
   TOTAL STRUC.GROUP WT.        (WST)        1205.

FLIGHT CONTROLS GROUP
   COCKPIT CONTROLS             (WCC)          34.
   FIXED WING CONTROLS          (WCFW)         19.
   SAS                          (WSAS)          0.
   GROUP WEIGHT INC.            (DELWFC)       29.
   TOTAL CONTROL WT.            (WFC)          81.

WT. OF FIXED EQUIPMENT          (WFE)         736.

WEIGHT EMPTY                    (WE)         2522.

FIXED USEFUL LOAD               (WFUL)        210. (INC. CREW OF 1)

OPERATING WEIGHT EMPTY          (OWE)        2732.

PAYLOAD                         (WPL)         200. (PAX=  [illegible].)

FUEL                            (WFA)         712. (WFW=  712.) (WFTP=    0.)

GROSS WEIGHT                    (WG)         3643.
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APPENDIX B

Tandem; Twin (Cont'd)

GROSS WEIGHT = 3643. PASSENGERS = 1. PLUS CREW OF 1

FUSELAGE	LENGTH	(ELF)	29.00	FT
	WIDTH	(SWF)	3.25	FT
	WETTED AREA	(SF)	336.	SQFT
	DELTA P	(DELP)	4.20	PSI
WING	ASPECT RATIO	(AR)	10.00	
	AREA	(SW)	81.0	SQFT
	SPAN	(B)	28.5	FT
	GEOM. MEAN CHORD	(CBARW)	2.95	FT
	QUARTER CHORD SWEEP	(DLMC4)	0.0	DEG
	TAPER RATIO	(SLM)	.500	
	ROOT THICKNESS	(TCR)	.170	
	TIP THICKNESS	(TCT)	.170	
	WING LOADING	(WGS)	45.0	PSF
	WING FUEL VOLUME	(VFW)	27.5	CUFT
HOR. TAIL	ASPECT RATIO	(ARHT)	5.00	
	AREA	(SHT)	16.3	SQFT
	SPAN	(BHT)	9.04	FT
	MEAN CHORD	(CBARHT)	1.87	FT
	THICKNESS/CHORD	(TCHT)	.070	
	MOMENT ARM	(ELTH)	13.6	FT
	VOLUME COEFF.	(VBARH)	.930	
VERT. TAIL	ASPECT RATIO	(ARVT)	1.90	
	AREA	(SVT)	15.3	SQFT
	SPAN	(BVT)	5.39	FT
	MEAN CHORD	(CBARVT)	2.94	FT
	THICKNESS/CHORD	(TCVT)	.087	
	MOMENT ARM	(ELTV)	14.2	FT
	VOLUME COEFF.	(VBARV)	.094	
ENG.NACELLES	LENGTH	(ELN)	4.83	FT
	MEAN DIAMETER	(DBARN)	1.19	FT
	NUMBER ENGINES	(ENP)	2.0	
	WETTED AREA	(SN)	36.11	SQFT

APPENDIX B

Tandem; Twin (Cont'd)

```
CRUISE MACH =  .399            CRUISE ALTITUDE =  15000.

CRUISE RE.NUM. PER FT.    = 1.843E+06    FLATPLATE CF AT RE=10**7 IS  .00287

AERODYNAMICS DATA
    TOTAL EFFECTIVE FLATPLATE AREA    (FE)        2.432    SQFT
    TOTAL WETTED AREA                 (SWET)      585.0    SQFT
    MEAN SKIN FRICTION COEFF.         (CBARF)    .00416

DRAG BREAKDOWN IN SQFT
    WING                              (FEW)       .707
    FUSELAGE                          (FEF)      1.263
    VERT. TAIL                        (FEVT)      .139
    HOR. TAIL                         (FEHT)      .153
    ENGINE NACELLES                   (FEN)       .138
    TIP TANKS                         (FETP)     0.000
    INCREMENTAL FE                    (DLTAFE)    .032

AERODYNAMIC COEFF.
    A1                                           .6850
    A2                                          -.1158
    A3                                           .0781
    A4=.75*(T/C)                                 .1275
    A5=CDO--                                     .0213
    A6                                          3.0388
    A7=1/(PI*SEE*AR)                             .0406
    3-D LIFT SLOPE AT CRUISE MACH     (CLALPH)  5.5188    PER RADIAN
    OSWALD FACTOR                     (SEE)      .7846

CRUISE  CD =  .0300 +   .0406 CL**2
LANDING GEAR CD INCREMENT= .02793
```

APPENDIX B

GASP Printout For
Tandem; Single-Engine Airplane
Solution with 0.95 Ram Recovery

```
VDIVE =  340. KTS    VMO =  289. KTS    MMO =  .792
ULT. LF =  9.00   MAN. LF =  6.00   GUST LF =  2.98

PROPULSION GROUP
   PRIMARY ENGINES            (WEP)        407.
   PRIMARY ENGINE INSTL.      (WPEI)       101.
   FUEL SYSTEM                (WFSS)        85.
   PROPULSOR WEIGHT          (WPROP)         0.
   TOTAL PROP.GROUP WT.       (WP)         593.

STRUCTURES GROUP
   WING                       (WW)         477.
   HOR. TAIL                  (WHT)         64.
   VERT. TAIL                 (WVT)         39.
   FUSELAGE                   (WB)         517.
   LANDING GEAR               (WLG)        197.
   PRIMARY ENG. SECTION       (WPES)        87.
   GROUP WEIGHT INC.          (DELWST)       0.
   TOTAL STRUC.GROUP WT.      (WST)       1381.

FLIGHT CONTROLS GROP
   COCPIT CONTROLS            (WCC)         36.
   FIXED WING CONTROLS        (WCFW)        23.
   SAS                        (WSAS)         0.
   GROUP WEIGHT INC.          (DELWFC)      29.
   TOTAL CONTROL WT.          (WFC)         88.

WT. OF FIXED EQUIPMENT        (WFF)        736.

WEIGHT EMPTY                  (WE)        2798.

FIXED USEFUL LOAD             (WFUL)       210. (INC. CREW OF 1)

OPERATING WEIGHT EMPTY        (OWE)       3008.

PAYLOAD                       (WPL)        200. (PAX=  1.)

FUEL                          (WFA)        945. (WFW=  945.) (WFTP=    0.)

GROSS WEIGHT                  (WG)        4154.
```

APPENDIX B

Tandem; Single (Cont'd)
$\eta_R = 0.95$

GROSS WEIGHT = 4154. PASSENGERS = 1. PLUS CREW OF 1

FUSELAGE	LENGTH	(ELF)	29.00	FT
	WIDTH	(SWF)	3.25	FT
	WETTED AREA	(SF)	336.	SQFT
	DELTA P	(DELP)	4.20	PSI
WING	ASPECT RATIO	(AR)	10.00	
	AREA	(SW)	92.3	SQFT
	SPAN	(B)	30.4	FT
	GEOM. MEAN CHORD	(CBARW)	3.15	FT
	QUARTER CHORD SWEEP	(DLMC4)	0.0	DEG
	TAPER RATIO	(SLM)	.500	
	ROOT THICKNESS	(TCR)	.170	
	TIP THICKNESS	(TCT)	.170	
	WING LOADING	(WGS)	45.0	PSF
	WING FUEL VOLUME	(VFW)	33.5	CUFT
HOR. TAIL	ASPECT RATIO	(ARHT)	5.00	
	AREA	(SHT)	20.1	SQFT
	SPAN	(BHT)	10.01	FT
	MEAN CHORD	(CBARHT)	2.08	FT
	THICKNESS/CHORD	(TCHT)	.070	
	MOMENT ARM	(ELTH)	13.5	FT
	VOLUME COEFF.	(VBARH)	.930	
VERT. TAIL	ASPECT RATIO	(ARVT)	1.55	
	AREA	(SVT)	15.0	SQFT
	SPAN	(BVT)	4.82	FT
	MEAN CHORD	(CBARVT)	3.22	FT
	THICKNESS/CHORD	(TCVT)	.087	
	MOMENT ARM	(ELTV)	14.0	FT
	VOLUME COEFF.	(VBARV)	.075	
ENG.NACELLES	LENGTH	(ELN)	4.60	FT
	MEAN DIAMETER	(DBARN)	3.36	FT
	NUMBER ENGINES	(ENP)	1.0	
	WETTED AREA	(SN)	48.56	SQFT

APPENDIX B

Tandem; Single (Cont'd)
$\eta_R = 0.95$

```
CRUISE MACH =  .399          CRUISE ALTITUDE =  15000.

CRUISE RE.NUM. PER FT.    = 1.843E+06    FLATPLATE CF AT RE=10**7 IS  .00287

AERODYNAMICS DATA
    TOTAL EFFECTIVE FLATPLATE AREA      (FE)       2.602   SQFT
    TOTAL WETTED AREA                   (SWET)     626.0   SQFT
    MEAN SKIN FRICTION COEFF.           (CBARF)   .00416

DRAG BREAKDOWN IN SQFT
    WING                                (FEW)       .797
    FUSELAGE                            (FEF)      1.263
    VERT. TAIL                          (FEVT)      .134
    HOR. TAIL                           (FEHT)      .185
    ENGINE NACELLES                     (FEN)       .186
    TIP TANKS                           (FETP)     0.000
    INCREMENTAL FE                      (DLTAFE)    .037

AERODYNAMIC COEFF.
    A1                                             .6850
    A2                                            -.1158
    A3                                             .0781
    A4=.75*(T/C)                                   .1275
    A5=CD0--                                       .0196
    A6                                            3.0057
    A7=1/(PI*SEE*AR)                               .0406
    3-D LIFT SLOPE AT CRUISE MACH       (CLALPH)  5.5188   PER RADIAN
    OSWALD FACTOR                       (SEE)      .7846

CRUISE  CD =  .0282 +   .0406 CL**2
LANDING GEAR CD INCREMENT= .02721

*****START OF INPUT FOR CONTROL

END OF INPUT DATA, JOB COMPLETE.
```

APPENDIX B

GASP Printout For
Tandem; Single-Engine Airplane
Solution with 0.99 Ram Recovery

```
VDIVE =  340. KTS    VMO =  280. KTS    MMO =   .792
ULT. LF =   9.00    MAN. LF =   6.00    GUST LF =  2.98

PROPULSION GROUP
   PRIMARY ENGINES                  (WFP)          354.
   PRIMARY ENGINE INSTL.            (WPEI)          88.
   FUEL SYSTEM                      (WFSS)          65.
   PROPULSOR WEIGHT                 (WPROP)          0.
   TOTAL PROP.GROUP WT.             (WP)           507.

STRUCTURES GROUP
   WING                             (WW)           413.
   HOR. TAIL                        (WHT)           52.
   VERT. TAIL                       (WVT)           32.
   FUSELAGE                         (WB)           507.
   LANDING GEAR                     (WLG)          175.
   PRIMARY ENG. SECTION             (WPES)          67.
   GROUP WEIGHT INC.                (DELWST)         0.
   TOTAL STRUC.GROUP WT.            (WST)         1246.

FLIGHT CONTROLS GROUP
   COCKPIT CONTROLS                 (WCC)           34.
   FIXED WING CONTROLS              (WCFW)          19.
   SAS                              (WSAS)           0.
   GROUP WEIGHT INC.                (DELWFC)        29.
   TOTAL CONTROL WT.                (WFC)           82.

WT. OF FIXED EQUIPMENT              (WFE)          736.

WEIGHT EMPTY                        (WE)          2570.

FIXED USEFUL LOAD                   (WFUL)         210. (INC. CREW OF 1)

OPERATING WEIGHT EMPTY              (OWE)         2780.

PAYLOAD                             (WPL)          200. (PAX=   1.)

FUEL                                (WFA)          720. (WFW=   720.) (WFTP=     0.)

GROSS WEIGHT                        (WG)          3701.
```

APPENDIX B

Tandem; Single (Cont'd)
$\eta_R = 0.99$

GROSS WEIGHT = 3701. PASSENGERS = 1. PLUS CREW OF 1

FUSELAGE	LENGTH	(ELF)	29.00	FT
	WIDTH	(SWF)	3.25	FT
	WETTED AREA	(SF)	336.	SQFT
	DELTA P	(DELP)	4.20	PSI
WING	ASPECT RATIO	(AR)	10.00	
	AREA	(SW)	82.2	SQFT
	SPAN	(B)	28.7	FT
	GEOM. MEAN CHORD	(CBARW)	2.97	FT
	QUARTER CHORD SWEEP	(DLMC4)	0.0	DEG
	TAPER RATIO	(SLM)	.500	
	ROOT THICKNESS	(TCR)	.170	
	TIP THICKNESS	(TCT)	.170	
	WING LOADING	(WGS)	45.0	PSF
	WING FUEL VOLUME	(VFW)	28.2	CUFT
HOR. TAIL	ASPECT RATIO	(ARHT)	5.00	
	AREA	(SHT)	16.0	SQFT
	SPAN	(BHT)	8.95	FT
	MEAN CHORD	(CBARHT)	1.86	FT
	THICKNESS/CHORD	(TCHT)	.070	
	MOMENT ARM	(ELTH)	14.2	FT
	VOLUME COEFF.	(VBARH)	.930	
VERT. TAIL	ASPECT RATIO	(ARVT)	1.55	
	AREA	(SVT)	12.0	SQFT
	SPAN	(BVT)	4.32	FT
	MEAN CHORD	(CBARVT)	2.89	FT
	THICKNESS/CHORD	(TCVT)	.087	
	MOMENT ARM	(ELTV)	14.7	FT
	VOLUME COEFF.	(VBARV)	.075	
ENG.NACELLES	LENGTH	(ELN)	4.60	FT
	MEAN DIAMETER	(DBARN)	3.36	FT
	NUMBER ENGINES	(ENP)	1.0	
	WETTED AREA	(SN)	48.66	SQFT

APPENDIX B

Tandem; Single (Cont'd)
$\eta_R = 0.99$

```
CRUISE MACH =  .399            CRUISE ALTITUDE =  15000.

CRUISE RE.NUM. PER FT.    = 1.843E+06    FLATPLATE CF AT RE=10**7 IS  .00287

AERODYNAMICS DATA
    TOTAL EFFECTIVE FLATPLATE AREA    (FF)        2.459   SQFT
    TOTAL WETTED AREA                 (SWET)      592.7   SQFT
    MEAN SKIN FRICTION COEFF.         (CBARF)    .00415

DRAG BREAKDOWN IN SQFT
    WING                              (FEW)        .717
    FUSELAGE                          (FEF)       1.263
    VERT. TAIL                        (FEVT)       .109
    HOR. TAIL                         (FEHT)       .150
    ENGINE NACELLES                   (FEN)        .186
    TIP TANKS                         (FETP)      0.000
    INCREMENTAL FF                    (DLTAFE)     .033

AERODYNAMIC COEFF.
    A1                                            .6850
    A2                                           -.1158
    A3                                            .0781
    A4=.75*(T/C)                                  .1275
    A5=CDO--                                      .0212
    A6                                           3.0348
    A7=1/(PI*SEF*AR)                              .0406
    3-D LIFT SLOPE AT CRUISE MACH     (CLALPH)   5.5188   PER RADIAN
    OSWALD FACTOR                     (SEE)       .7846

CRUISE  CD =  .0299 +   .0406 CL**2
LANDING GEAR CD INCREMENT= .02784

*****START OF INPUT FOR CONTROL

END OF INPUT DATA. JOB COMPLETE.
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APPENDIX C

ECONOMIC CONSIDERATIONS - THE POTENTIAL FOR FUEL AND OPERATING COST SAVINGS

All who participated in the investigation summarized in this report are convinced that the results merit further study, particularly the economic questions left unanswered. It is the intent of this appendix to outline some of these considerations, but due to its brevity, no claim is made for its conclusiveness. Rather, it is intended to indicate the need for in-depth cost estimates, and life-cycle cost analyses of the military primary trainer defined in this study.

The "Randolph study," repeatedly referred to in the text of this report, received such attention because of the TA-2 conceptual trainer design that is described in the study group"s final report. This conceptual trainer is most representative of a turbofan-powered light airplane that could become an actual military requirement. Furthermore, the concept was supported by a massive amount of U.S. Air Force-sponsored analyses. This comprehensive work resulted in many recommendations among which was to undertake the procurement of the TA-2 airplane. This economics appendix supports that recommendation.

The Randolph study was, as its title states, a complete "Mission Analysis on Future Undergraduate Pilot Training: 1975 through 1990." Three considerations appear to have been given to every question addressed: requirements for; effectiveness of; and cost of. Avoiding the issue of requirements, this NASA/Garrett investigation has provided an in-depth analysis of both effectiveness- and cost-related factors. The work appears to have improved upon the Air Force results affecting these factors. Smaller, presumably less costly, candidate airplanes were derived. It has been shown how operating costs, as influenced by low fuel consumption and less maintenance, could be reduced from the Air Force estimates. Finally, the "energy crisis" provided a sense of urgency to the need for maximum efficiency attainments.

The General Officer Steering Committee that reviewed the Randolph study results recommended: "The decision to procure any new conceptual aircraft (TA-2, TA-3, or TA-4) should be deferred until the 1979-1982 time period." That recommendation, made in 1971, does not, of course, reflect the urgency that must have been felt in 1974, when undergraduate pilot production was reduced for lack of fuel. Nor does it reflect the fact that recent appropriations of T-37's by other U.S. Air Force commands may cause an early insufficiency at Training Command. Also, it does not reflect the fact that an equally capable, but more

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economical airplane than the Randolph study TA-2 could be achieved.

Based upon information contained in the Randolph study, cost data from various other sources, and the results of this investigation, the following extensions of fuel and cost savings are projected for a new primary trainer:

FUEL SAVINGS (vs. T-37 FLEET)

Current Annual Consumption @ 190 gph, 90 hr/graduate, 3665 grad./yr.	62.7 million gal.
Potential Annual Consumption @ 38 gph, 90 hr/graduate, 3665 grad./yr.	12.6 million gal.
Potential Annual Fuel Savings	50.1 million gal.
Projected Fuel Cost Savings	
@ 15¢/gal	$ 7.5 million/yr.
@ 20¢/gal	10.0 million/yr.
@ 30¢/gal	15.0 million/yr.
@ 50¢/gal	22.5 million/yr.

OPERATING COST SAVINGS (vs. T-37 FLEET)

Current Annual Operating Cost @ $125/hr, 90 hr/graduate, 3665 grad/yr.	$41.2 million/yr.
Potential Annual Operating Cost @ $31/hr, 90 hr/graduate, 3665 grad/yr.	10.2 million/yr.
Potential Annual Operating Cost Savings	31.0 million/yr.

In discussions held with various USAF personnel, it was learned that, with the increased performance, range, and IFR capability of the TA-2 primary trainer concept, it could displace the T-38 in a small portion of the training syllabus. If 10 hours were to be considered a reasonable such displacement, then the following additional savings could be projected:

FUEL SAVINGS (vs. T-38 IN 10 SYLLABUS HOURS)

Current Annual Consumption @ 390 gph, 10 hr/graduate 3665 grad/yr.	14.3 million gal.
Potential Annual Consumption @ 38 gph, 10 hr/graduate, 3665 grad/yr.	1.4 million gal.
Potential Annual Fuel Savings	12.9 million gal.

APPENDIX C

Projected Fuel Cost Savings	
@ 15¢/gal	$ 1.94 million/yr.
@ 20¢/gal	2.58 million/yr.
@ 30¢/gal	3.87 million/yr.
@ 50¢/gal	6.45 million/yr.

OPERATING COST SAVINGS (vs. T-38 IN 10 SYLLABUS HOURS)

Current Annual Operating Cost @ $250/hr, 10 hr/graduate, 3665 grad/yr.	$ 9.2 million/yr.
Potential Annual Operating Cost @ $31/hr, 10 hr/graduate, 3665 grad/yr.	1.1 million/yr.
Potential Annual Operating Cost Savings	$ 8.1 million/yr.

Combining the potential fuel savings estimated here, the total is 63 million gallons saved per year. It is doubtful that there is another area in military aviation where a potential for savings of more than 80 percent exists.

Similarly, with a potential total annual operating cost savings of $39.1 million, the case for a new primary trainer merits careful review.

Finally, it has been estimated that, at a total program cost of less than $200 million, the U.S. Air Force could procure a new primary trainer fleet. If this cost and the foregoing projected savings are accurate, such a program would pay for itself in about 5 years.

REFERENCES

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