TL. 113-58/10
SECRET
261

AVRO AIRCRAFT LIMITED

MALTON – ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT: ARROW 3

REPORT NO: ____________

FILE NO:

NO. OF SHEETS 52

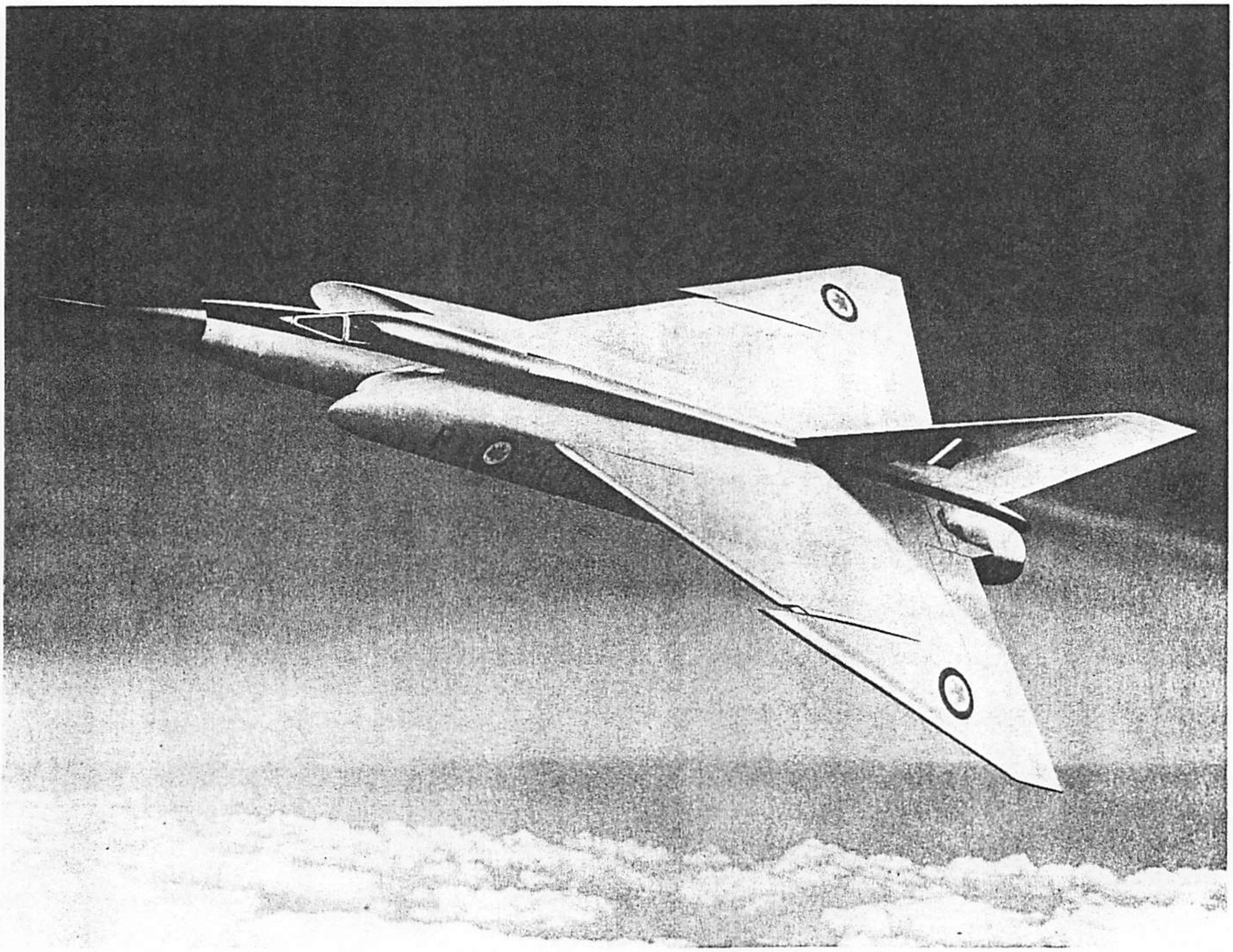
TITLE: PRELIMINARY PROPOSAL FOR ARROW 3

PREPARED BY C. H. Petterson DATE Oct 27th/1958

RECOMMENDED FOR APPROVAL ____________ DATE ____________

APPROVED Jim Frame DATE Oct. 27/58.

APPROVED FOR RELEASE C. V. Lindow DATE Oct /58



ARROW 3

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1. SUMMARY

This report presents the results of an investigation into the possibilities of stretching the potential of the present Arrow design. The proposed aircraft is called the Arrow 3.

The Arrow 3 retains most of the features of the Arrow 2 and approximates to the same configuration, but it has been found possible to increase the maximum speed of this variant to a Mach number of 3.0, the combat ceiling to 70,500 ft., the high speed combat range to 389 nautical miles, and the long range ferry mission range on internal fuel to 1,707 nautical miles. This has been accomplished by the adoption of multi-ramp variable geometry air intakes, the installation of a modified Iroquois engine, the use of external thermal insulation to protect the aircraft structure, and the provision of additional internal fuel stowage.

The structural changes between the Arrow 2 and Arrow 3 have been kept to a minimum. The wing structure is basically unchanged, apart from the use of sculptured structure for the wing outboard of the transport joint to permit additional fuel stowage. Sculptured structure has also been adopted for the fin, which has been increased in span to satisfy lateral stability requirements for Mach 3.0 flight.

The structural changes to the fuselage are more severe, due to the necessity to use stainless steel structure in the vicinity of the intakes and adjacent to the engine bays, and considerable redesign has been necessary. The fuselage centre section has also been redesigned with sculptured skins to act as an integral fuel tank.

Apart from minor changes the Arrow 2 systems can be adopted unaltered. The changes required are generally to the heat exchangers, and are made necessary by the higher free stream recovery temperatures at Mach 3.0 flight.

These modifications, together with the additional fuel, have resulted in an increase in all up weight of 16,909 lb. over that for the Arrow 2. A comparison of Arrow 3 weights and performance with those for the Arrow 2 is given in the following table.

AIRCRAFT	ARROW 2	ARROW 3
Operational Weight Empty	46,045 lb.	53,654 lb.
Maximum Useable Internal Fuel	19,438 lb.	28,738 lb.
Gross take-off weight (max. internal fuel)	65,483 lb.	82,392 lb.
Combat Weight (1/2 max. int. fuel)	55,764 lb.	68,023 lb.
Normal design landing gross weight	49,177 lb.	59,111 lb.
Take-off distance over a 50 ft. obstacle at S.L., gross take-off weight, max. thrust, A/B lit	3,770 ft.	6,500 ft.
Landing distance over a 50 ft. obstacle at S.L., normal design landing gross weight	4,750 ft.	5,650 ft.
Steady state rate of climb at sea level, combat weight, 0.92 M, max. thrust, A/B lit	43,000 ft./min.	29,400 ft./min.
Combat ceiling with 500 ft./min. rate of climb, combat weight, optimum Mach No., max. thrust, A/B lit	59,700 ft. (1.8M)	70,500 ft. (3.0M)
Max. level speed (placard)	M = 2.0	M = 3.0

2. INTRODUCTION

The Arrow 2 is a two seat, twin engined, all weather interceptor which is scheduled to go into squadron service in 1960. The preliminary design was based on RCAF specification AIR 7-4 which calls for combat at 50,000 ft. altitude and a Mach number of 1.5. Predicted performance of the Arrow 2, however, gives a combat capability of 60,000 ft. and a Mach number of 2.0. The prototype of this aircraft, the Arrow 1 has already flown, and flight testing and development are proceeding at the present time. Analysis of flight test results to date indicates that the predicted performance figures are conservative.

The Arrow design is completely new, so that considerable development of the airframe and engines may be expected during the operational life of the aircraft. Normally, aircraft development is directed towards increasing speed and range, therefore a preliminary investigation into the modifications necessary to vary these parameters was initiated. As a result of this investigation, the concept of the Arrow 3, a development of the Arrow family that retains very largely the structure and systems of the Arrow 2, has been formulated.

Three criteria were laid down for the design investigation. These were:

1. That development costs were to be held to a minimum. This can best be done by retaining as much as possible of the existing structure and systems.
2. That the aircraft should have Mach 3 capabilities at 70,000 ft. This

necessitates designing for a Mach number slightly greater than 3 to cover the case when a pilot inadvertently exceeds the design Mach number.

3. That maximum internal fuel storage should be provided, to ensure a reasonable range at M = 3. 0.

The requirement for sustained Mach 3 flight with its high temperatures imposes severe thermodynamic problems on the aircraft structure and systems. At this speed structures attain a steady state temperature of about 500°F., and, as this is well above the working limits of aluminum and magnesium, unprotected structures have to be made of titanium or steel. Also considerable difficulty is met in developing systems to operate satisfactorily at this temperature level, and the flow of heat to the fuel raises its temperature above the allowable limits. These considerations indicate that the use of an uninsulated structure would lead to a practically complete redesign of the aircraft structure and systems. On the other hand, use of an external insulation on the aircraft permits the existing light alloy structure to be largely retained, lowers the operating temperature of the systems, and retards the flow of heat to the fuel, Therefore the decision is made to use an external insulation to preclude major design changes.

A further result of the Mach 3 requirements is the adoption of modified engines and the redesign of the air intakes. The engines are Iroquois 3, developments of the Iroquois 2 which are suitable for operation at Mach 3. The two-dimensional single-ramp type air intakes on the Arrow 2 are only suitable for use up to a Mach number of approximately 2. 3, therefore completely new intakes of the multi-shock type, which give better pressure recoveries at high Mach numbers, have been investigated. Results indicate that a three-dimensional triple-ramp type variable geometry intake, running subcritically up to M = 2.5 and supercritically at higher Mach numbers, is suitable. Geometric considerations dictate that such an inlet can most satisfactorily be installed if the ramp is mounted on the side of the inlet outboard from the fuselage, therefore this configuration is adopted.

Additional fuel space is provided in the inboard wing forward of the under-carriage wells, in the outer wing, in the fin, and in the fuselage centre section. This permits an additional 1188 imperial gallons of fuel to be stored internally in the aircraft.

The effect of insulation, changes to the structure and systems due to the higher operating temperatures, and the additional fuel, is to increase the all up weight of the aircraft by 16,909 lb. Therefore some strengthening of the landing gear is necessary.

Weights used for design and performance purposes are derived by adding incremental weight increases due to the modifications to the weights of the Arrow 2. The Arrow 2 weights published on March 1st, 1958, are selected as the basic weights for the analysis, all subsequent variations being neglected.

The Arrow 3 is described in detail in the following pages. A general arrangement drawing is presented in Fig. 1.

3. MODIFICATIONS TO THE ARROW 2

3.1 Aerodynamic Requirements

3.1.1 Performance

As mentioned in the introduction, design of the Arrow 2 was based on a 1.5 Mach number requirement. Therefore, although the aircraft was actually designed to have Mach 2.0 capabilities, individual design points were often optimised for flight at Mach 1.5 or lower. Cases where this has occurred on the Arrow 2 are the adoption of leading edge droop, which is beneficial to drag at all speeds up to about M = 1.5 but imposes a drag penalty at higher Mach numbers, and the leading edge notches, which were adopted from subsonic stability considerations and give additional drag at all speeds. Another case is the ejectors, which were designed for a Mach number intermediate between 1.5 and 2.0 so as to give reasonable performance throughout this range.

Although the adoption of the above principal has had no serious effects on the Arrow 2 design, the weighting of design decisions towards Mach 1.5 flight may compromise Mach 3.0 performance. Therefore, such decisions effecting items which do not involve major structural changes should be reconsidered for the Arrow 3 design.

The major item involved is the wing leading edge droop. Wind tunnel tests on the Arrow 2 indicated that with an undrooped leading edge the weather-cock stability for small angles of yaw was zero or negative subsonically, due apparently to the effects of the wing-body junction on the flow over the fin. Further tests with a drooped leading edge indicated that the aircraft in this configuration would be laterally stable, and that there was a reduction in drag under most flight conditions at Mach numbers up to about 1.5, with no serious increase in drag at higher Mach numbers up to 2.0. Therefore a drooped leading edge configuration was adopted for the Arrow 2.

Estimates for the Arrow 3, however, indicate that at a Mach number of 3.0 the effect of leading edge droop is to increase the drag by at least 10 drag counts. This is a heavy penalty to pay if use of the droop can be avoided. The Arrow 3 has new intakes (Section 3.2.2) which it is hoped will clear up the flow in the wing body junction, and this in conjunction with the proposed larger fin (Section 3.1.2) is expected to provide sufficient lateral stability subsonically to permit the leading edge droop to be removed. However, before any final decision is made on this matter it will be necessary to carry out a wind tunnel programme on a model of the Arrow 3. As a result of this programme it is hoped that it will also be possible to fill in the leading edge notches, and thus gain a further saving in drag throughout the whole flight regime.

3.1.2 Stability and Control

The predicted stability and control characteristics of the Arrow 1 are based on wind tunnel measurements which have been corrected to allow for the effects of thrust and of structural elasticity. Where stability augmentation has been used in a control circuit its effect has been investigated on an analogue computer, using aircraft characteristics derived by the above method. Preliminary results of the Arrow 1 flight test programme confirm the validity of the method of prediction.

The stability and control of the Arrow 3 has been investigated in a similar manner. However, as no wind tunnel results are available for the Arrow 3 configuration, recourse has had to be made to using the Arrow 1 wind tunnel results corrected for geometry changes between the two aircraft. Also, as Arrow 1 results do not cover the flight range $2.0 < M < 3.0$, values in this region were obtained by extrapolating the test results for lower Mach numbers, the trends being checked to ensure that they were consistent with theory.

The results of this investigation indicate that the longitudinal stability of the Arrow 3 is satisfactory in all flight regions, and that, apart from a possible rescheduling of the gains in the damping circuit, no modifications to the controls, control circuits, or damping circuits, of the Arrow 2 will be required to suit them to the Arrow 3.

The results of the lateral stability investigation are not as satisfactory as those for the longitudinal. Arrow 2 results with the lateral dampers out show a region of marginal lateral stability when operating at a Mach number of 2.0 at 30,000 ft. altitude. With dampers in, stability in this region is satisfactory. Arrow 3 results, however, indicate that as the Mach number is increased the lateral stability characteristics become steadily worse, and by the time a Mach number of 3.0 is reached a definite instability occurs at all flight altitudes with dampers out. An investigation on the analogue computer indicates that, while this instability can be cured by modifying the rudder damper circuit, considerable increases in the scheduled gains of rudder motion to lateral acceleration will be required. These large increases in gains will, however, introduce problems in scheduling, servo saturation, rudder rates, and hinge moment limitations.

An investigation into the effect of increasing the fin size shows that an extension of the fin span by approximately 2.5 ft., together with a small increase in length of the fin tip chord, will improve the lateral stability characteristics sufficiently for the rudder damper redesign to present no serious problems. The damper modification will consist of revisions to the gain schedules of the roll and yaw axis, and the possible introduction of altitude scheduling to cover flights at extreme altitude and maximum Mach number.

It is proposed that this increase in fin area and modification to the rudder damper circuit be adopted for the Arrow 3.

3.2 Propulsion Requirements

3.2.1 Engines

The engines used for the Arrow 2 are Iroquois 2. These engines were designed for flight up to a Mach number of 2.0, and have a sea level static thrust of 19,250 lb. without afterburner and 26,000 lb. with afterburner.

For powering a Mach 3 development of the Arrow, the manufacturers of the Iroquois 2 have suggested a modified version of this engine which is called the Iroquois 3. The Iroquois 3 is fundamentally an Iroquois 2, but has been designed for operating at speeds up to Mach 3.0 under standard day conditions at 45,000 ft. The engine has a sea level static thrust of 19,800 lb. without afterburner, and 26,700 lb. with afterburner.

Major modifications to the Iroquois 2 to obtain the improved high Mach number performance are the use of titanium blades and rotor disks in the compressors wherever permitted by the stage temperatures, use of a larger compressor annulus for both the low pressure and the high pressure stages, shortening of the combustion chamber length by 6 inches, reduction of the chord length of the turbine blades to reduce the turbine stage lengths, and the insulating of the oil tank, oil and fuel lines, hydraulic units, bearing oil sumps, and the control units. The engine nozzle is of the fully variable convergent type. It is fully modulated between 60% and 100% thrust.

Apart from the general shortening of the engine due to the decreased combustion chamber and turbine stage lengths, the basic engine profile is little changed from the Iroquois 2. There has, however, been a considerable increase in the length and diameter of the final nozzle.

Design details and performance data for the Iroquois 3 are presented in reference 1. It is this engine that it is proposed should be used to power the Arrow 3.

3.2.2 Intakes

A single fixed-ramp two-dimensional intake, such as on the Arrow 2, gives acceptable pressure recoveries, solvable shock - boundary layer interaction problems, and reasonable bleed areas for flight up to Mach 2.0.

At higher Mach numbers, however, the large ramp angles required for acceptable pressure recoveries lead to large cowl angles, with resultant high drags, and to excessive flow turning angles to re-align the flow with the engine centre-line.

The basic problem is to develop an intake for the Arrow 3 that provides workable solutions to the following requirements:

(a) There must be no deterioration in performance in comparison with the Arrow 2 at speeds below Mach 2.

(b) There must be the minimum of changes to the Arrow 2 structure.

(c) There must be low additive drag and no adverse interference effects to increase the aircraft's external drag.

(c) A high total pressure recovery is required.

(e) Intake flow stability is required at all engine speeds from windmilling to maximum R. P. M.

The proposed solution is a multiple-shock ramp intake,with a swept-lip face placed outboard from the fuselage wall and thus directing the shock structure inwards.

This external ramp type intake reduces the wave drag and interference effects of shock waves in the airstream, and presents a knife edge, which is simply a faired extension of the Arrow 2 fuselage lines, to the external air stream.

The ramp angles are chosen to give no shock-boundary layer separation on either the ramp surface or the fuselage diverter plate under on-design and off-design conditions. This is accomplished by ensuring that the static pressure rise through attached oblique shocks, detached terminal shocks, and standing normal shocks, on any of the four ramps is less than the pressure rise required to separate a turbulent boundary layer. The pressure rise through all reflected shocks and standing shocks on the diverter plate satisfies the same requirements.

The ramp angles also determine the available pressure recovery and the intake throat area. A variable position fourth ramp is used to ensure reasonable intake performance throughout the complete Mach number range.

The capture area is designed to be a minimum consistent with giving sufficient mass flow for the engine and for engine cooling, so that the thrust minus additive drag will be a maximum. An analytical study indicates that 11.0 sq. ft. of capture area will be a near optimum for the Arrow 3 aircraft, but this figure requires confirmation from wind tunnel results. A small capture area will improve the nett thrust of the installation at speeds below the design Mach number, but at higher speeds will starve the engine.

The fuselage boundary layer is bled, through a fuselage diverter bleed, between the fuselage diverter plate and the fuselage wall in a similar manner to that on the Arrow 2. In the range $1.3<M<1.6$ the lip shock is capable of separating a turbulent boundary layer. Therefore the length and shape of the diverter plate is determined by the condition that at $M = 1.6$ the lip shock cannot reflect from it and form a standing normal shock wave on the diverter plate. This normal shock forms on the fuselage side, and the separated flow behind it is removed through the diverter bleed.

Because of the shock induced disturbance to the boundary layer on the diverter plate an inlet lip bleed is used to capture this disturbed low energy air and feed it to the air conditioning system.

Critical conditions for the intake occur at a Mach number of 2.5, when the terminal normal shock is strong enough to separate the turbulent boundary layer on the variable fourth ramp. Separation can be prevented if supersonic flow can be induced to take place in the throat so that the terminal normal shock is positioned internally. This induction of supersonic flow in the throat is known as "starting" the intake. It reduces the drag by swallowing the complete capture area, and sometimes improves the pressure recovery.

Starting the intake is a difficult procedure that requires fast response on the part of the variable ramp and of the variable by-pass actuators further down the intake duct. The inlet throat area must be suddenly increased until it can swallow the complete capture mass flow, and yet have an internal contraction ratio less than that required to choke the flow. To obtain these conditions it is proposed that at a Mach number of 2.5 the terminal normal shock be initially positioned at the inlet lip. This can be accomplished by programming the angle of the final ramp so that the constant area ducting to the throat adequately covers the contraction ratio requirements. This straight constant area portion of the ducting is introduced because even moderate turning of the flow upstream of the throat during supercritical operation gives complex internal shock conditions, with consequent severe velocity gradients and flow distortion in the duct.

During subcritical operation, to prevent the terminal normal shock moving forward off the ramp as a Mach number of 2.5 is approached, spillage is allowed to occur through a small slot cut upstream of the throat. This spillage is the excess air between full capture area and that entering the inlet. During supercritical operation only marginal spillage through the oblique shock structure is possible, and all excess flow must be taken through the inlet to prevent "buzz." This excess air must then by-pass the engine and be ejected axially, or be spilled through exit hatches in the air ducts.

3.2.3 Ejector and Engine By-Pass

An investigation into the design of a variable geometry ejector and surrounding structure for the Arrow 3 indicated that, although suitable solutions to this problem may be found, a long development program would be necessary. Therefore the decision was made that, for preliminary design purposes only, a fixed rear end geometry would be considered.

This fixed rear end geometry produces an aircraft which is reasonably efficient at supersonic speeds but inefficient under subsonic conditions. It severely penalises the aircraft range for all missions where a long subsonic cruise is called for, but the available range may be increased considerably if the use of a jettisonable ejector plug, such as that proposed for the Arrow 2, is adopted.

The fixed rear end geometry selected is based on an ejector throat diameter of 40 inches and exit diameter of 53 inches. This geometry is optimum for operation at a Mach number of 2.0 and penalises the aircraft performance under all other flight conditions. The pumping action of the ejector is such, however, that reasonable thrusts are available throughout the range $1.2<M<3.0$. A larger exit diameter would provide greater thrusts at $M=3.0$, but would seriously affect the rear end ground clearance during landings and take-offs.

The by-pass inlet area at the forward end of the engine is 450 sq. ins., and is based on the maximum excess flow through the intake under windmilling engine conditions for both subcritical and supercritical intake operation. Under maximum engine R.P.M. conditions all the by-pass air passes axially through the ejector, but at lower R.P.M. the excess air must be spilled from the by-pass between the by-pass inlet and the ejector. To obtain maximum nett thrusts this excess air should be expanded to ambient pressure and ejected axially, but the magnitude of the changes necessary to the Arrow 2 profile to accomplish this has led to the adoption of the simpler, but less efficient, scheme whereby the air is ejected laterally through the engine access doors. The area in the by-pass walls required to eject this excess air is 400 sq. ins.

3.3 Thermal Requirements

3.3.1 General

To enable the existing structure to be used for the Arrow 3 it is necessary for the structural temperatures and thermal stresses to be comparable with those of the Arrow 2. However, the maximum recovery temperature used in Arrow 2 design was 250°F., while flight at M = 3.0 on a standardday implies a stagnation temperature of 632°F.and a 90% recovery temperature of 562°F. This indicates that the existing structure can not be used for the Arrow 3 unless it is insulated to prevent its temperature rising above 250°F.

Insulation does not lower the steady state conditions that exist for any specified recovery temperature, but merely extends the time it takes for these conditions to occur. Therefore, if insulation is used to lower the temperature of a structure, the thermal conditions inthe structure will be transient, and the maximum temperatures attained dependent on the heating cycle as well as on insulation and structural parameters.

Apart from its effect on the maximum structural temperatures, other definite advantages accrue from the use of insulation. Thermal stresses decrease as the rate of heating is reduced,therefore the use of insulation will give lower thermal stresses than would occur in an uninsulated structure for the same rise in structural temperature. The amount of heat entering the aircraft is also reduced, and this eases the air-conditioning, fuel system, and other problems.

3.2 Insulation Requirements

The most severe heating conditions on the structure of the Arrow 3 occur when the aircraft takes off, accelerates rapidly to M = 3.0, and then cruises at this speed for the maximum time. Such a mission, detailed in Table 6, indicates that the aircraft is capable of sustained flight at M = 3.0 for approximately 15 minutes, and will provide the design condition for the insulation. As the time to accelerate to M = 3.0 is approximately 4 minutes, and investigation has shown that for an insulated structure the rate of change of recovery temperature is unimportant, a reasonable but conservative design figure is to assume M = 3.0 conditions for 20 minutes. The steady state wall temperature for this condition, assuming a 90% recovery factor and an emissivity of 0.9, is approximately 500°F.

The Arrow 2 maximum structural temperature is 250°F.: above this temperature the strength of the 7075 S aluminum alloy of which the primary structure is made falls off rapidly. However, the weight of insulation required is proportional to the temperature difference across the insulation, therefore any increase in the allowable structural temperature will show an appreciable weight saving. Examination has indicated that there are several aluminum alloys with satisfactory strength properties up to temperatures of 350°F., and the substitution of one of these for the existing 7075 S material will reduce the temperature differential from 250°F. without changing the structure. These materials are discussed in detail in section 3.4.1.

The above considerations lead to the adoption of a design criterion for the external insulation that it shall be chemically stable at temperatures up to 500°F., and that the temperature of the protected structure shall not exceed 350°F. after exposure to an M = 3.0 air stream for 20 minutes under condition corresponding to flight at 50,000 ft. or above.

This criterion does not apply to internal duct structure and structure subject to radiation from internal heat sources such as the engines. For such cases each insulation problem must be treated individually, although, in general, internal ducting will be subject to near stagnation air temperatures and receive no relief from radiant cooling.

The thickness of insulation required depends on:-

(a) its thermal conductivity.

(b) its heat storage capacity.

(c) the heat conductivity and heat storage capacity of the structure to be protected.

These factors are demonstrated in Fig. 2 where the variation of skin thickness with insulation thickness is plotted for flight at M = 3.0 at 60,000 ft. It is seen that as the skin thickness decreases the insulation thickness increases, the relationship being approximately of the inverse form. Therefore the insulation of very thin skins is not practical, and a lighter overall structure ensues if the skin thickness in such regions is increased.

Examination of probable insulation materials showed that there is very little variation in their specific heats. Assuming that the specific heat is independent of the insulation it can be shown that an important insulation parameter is ρ.k., the product of the density, ρ in.lb./ft.^{3}, and the thermal conductivity, k, in B.T.U./hr. ft.^{2} °F./ft. This parameter should be small for an efficient insulation, and analysis indicates that a value of approximately 1.0, or lower, will be satisfactory for the Arrow 3.

3.3.3 Structural

Generally it is proposed that the Arrow 2 structure be retained, but that the materials be changed from magnesium and 7075 S aluminum to higher temperature aluminum alloys, such as X2020-T6, which have satisfactory properties at temperatures up to 350°F. This structure will be covered externally by an insulation layer to prevent the temperatures exceeding the above value.

This principle can not be followed with compartments vented to atmosphere, and the engine nacelles. Where a compartment adjacent to the outside skin is vented to atmosphere both internal and external insulation would be required, and this would be structurally heavy. (A typical example of this is the bay containing the intake operating linkage.) The engines, also, are strong heat sources, and it would be prohibitive to prevent the external skin structure adjacent to them from rising above 350°F. Therefore it is proposed that such structures be fabricated of stainless steel and generally left uninsulated. In the case of the engine nacelles the stainless steel structure would reach temperatures of 700°F. to 800°F. at M = 3.0. The adjacent wing structure in this vicinity, however, would be of aluminum insulated to maintain a maximum temperature of 350°F.: this is practical

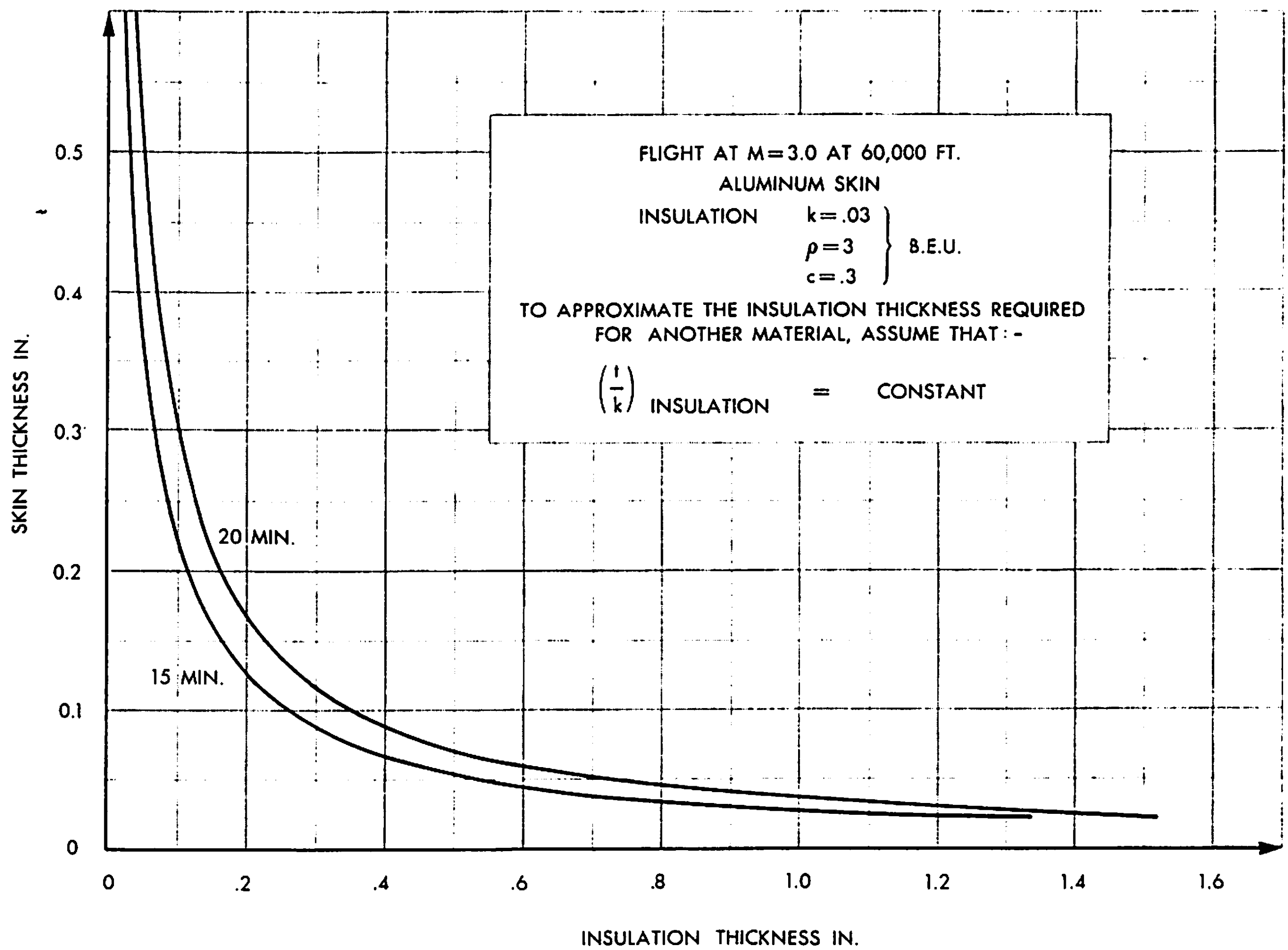

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FIG. 2 INSULATION REQUIRED TO MAINTAIN SKIN BELOW 350° F.

because wing skins are heavy gauge and provide a good heat sink. The equipment compartment between the engines, although fabricated of stainless steel, would also be insulated to protect the installed components from excessive temperatures.

3.3.4 Systems

Due to its being a large heat sink, and to the use of external insulation, the temperature of the fuel in the tanks will not rise appreciably. Therefore, the fuel may be used as a heat sink for the oil cooling system. Estimates indicate that JP 4 is a suitable fuel for the Arrow 3 and the fuel tank pressures need not be increased above the Arrow 2 values.

The oil cooling system, involving the alternator and constant speed unit cooling oil and the hydraulic fluid for systems, depends now entirely on fuel cooling at high Mach numbers. At low speeds oil to air heat exchangers are still required, but the air supply will be cut off at high speed and the oil to fuel heat exchangers used. To cater for short transient fuel temperature rises when throttling back from high speed, the installation of an oil to fuel cooler bypass and/or a water boiler is envisaged.

3.4 Structural Modifications

3 . 1 Structural Materials

The primary structure of the Arrow 2 is fabricated from 7075S aluminum alloy, with a portion of the fuselage skinning in magnesium and titanium alloys and with stainless steel in the engine section. Due to the rapid reduction in strength of 7075S-T6 at temperatures over 250°F., and the thickness of insulation required to prevent the structure exceeding this temperature, other alloys, having better elevated temperature properties, were investigated.

Three aluminum alloys were selected for investigation, namely X2020-T6, 2024-T81 and 2024-T86. The latter, although possessing slightly higher physical properties at room temperatures, has no better properties than 2024-T81 at elevated temperatures. Furthermore it is only available as sheet, therefore there is no advantage in using it in place of 2024-T81 which is available in both sheet and plate.

Pertinent physical properties of X2020-T6 and 2024-T81 are presented in Table 1 below for comparison. The values presented are typical and not guaranteed minima.

Other properties of these materials are roughly comparable, therefore because X2020-T6 appears to have the better physical characteristics it is selected as the preferable alloy and temper to substitute for 7075S-T6. As the physical properties fall off rapidly with temperatures greater than 350°F., an insulation is required on the outer skin to reduce the temperature to this value.

TABLE 1

PROPERTIES OF HIGH TEMPERATURE ALUMINUM ALLOYS

Aluminum Alloy	X2020-T6	2024-T81
Specific Gravity	2.71	2.77
Elastic Modulus, Tension, 75°F., p.s.i.	11.1×10^6	10.5×10^6
Elastic Modulus, Compression, 75°F., p.s.i.	11.4×10^6	10.7×10^6
Elastic Modulus, Tension 350°F., p.s.i.	9.7×10^6	9.7×10^6
Ultimate Tensile Stress at 75°F., p.s.i.	86,000	70,000
Ultimate Tensile Stress after 1/2 hr. exposure at 350°F., p.s.i.	67,500	61,000
Ultimate Tensile Stress after 100 hrs. exposure at 350°F., p.s.i.	57,500	53,000
Ultimate Tensile Stress, recovery after exposure at 350°F. for 1/2 hr., p.s.i.	81,000	70,000
Ultimate Tensile Stress, recovery after exposure at 350°F. for 100 hrs., p.s.i.	76,000	67,000
.2% proof stress at 75°F., p.s.i.	79,000	64,000
.2% proof stress after 1/2 hr. at 350°F., p.s.i.	65,000	53,000
.2% proof stress after 100 hrs. at 350°F., p.s.i.	60,000	50,000
.2% proof stress, recovery after 1/2 hr. at 350°F., p.s.i.	75,000	64,000
.2% proof stress, recovery after 100 hrs. at 350°F., p.s.i.	69,000	58,500
F[illegible]ture stress, under stress for 100 hr. at [illegible]00°F., p.s.i.	55,000	50,000
.1% creep stress after 10 hrs. at 300°F., p.s.i.	44,000	33,000

Where magnesium alloy skins are used on the fuselage of the Arrow 2 X2020-T6 aluminum alloy and insulation are used on the Arrow 3, since at the temperatures being considered, magnesium alloys are not efficient. From the forward station of the fuselage fuel tank, the air intake ducts are machined from aluminum alloy plate, rolled to shape, and insulated internally i.e., next to the air flow.

Stainless steel structure is used around the engine bays at the aft end of the fuselage, because of the thickness of insulation required to prevent an aluminum structure exceeding 350°F. It is also used for that portion of the air intakes forward of and including the moveable ramps and ramp operating linkage bay, because it avoids the use of excessive thicknesses of insulation and is easier to fabricate into irregular shapes. Apart from the equipment compartment between the engines, the stainless steel structure is not insulated.

Table 2 below contains a summary of the structural materials it is proposed to use, the maximum working temperatures of these materials, and the maximum temperatures at the outer surface of the insulation. It is seen that all insulated structure has an allowable maximum temperature of 350°F., and that two types of insulation are required. One has a maximum working temperature of 500°F., and the other, a maximum working temperature of 700°F.

TABLE 2

SUMMARY OF STRUCTURAL MATERIALS

COMPONENT	MATERIAL	MAXIMUM MATERIALS TEMP. °F.	MAXIMUM INSULATION TEMP. °F.
Fuselage Skins:—			
Aft of Radome to Aft Bulkhead of cockpit	X2020-T6	350	500
Air Intakes to Sta. 315	Stainless Steel	Int. - 700 Ext. - 500	Not insulated
Sta. 315 to Sta. 579	X2020-T6	350	500
Sta. 579 to Sta. 807	Stainless Steel	Ext. - 500 Int. - 700	Not insulated
Wing Skins	X2020-T6	350	500
F Skins	X2020-T6	350	500
Duct to Sta. 315	Stainless Steel	700	Not insulated
Duct - Sta. 315 to Sta. 579	X2020-T6	350	700

3.4.2 Insulation

3.4.2.1 Development of an Insulation

A check in the early stages of the investigation on available literature suggested no material that would provide a satisfactory insulation for the Arrow 3 aircraft, and indicated that little research work had been done on protecting light alloy structures during flight in the vicinity of Mach 3. Therefore a development program was started in the Metallurgical Department of Avro Aircraft to develop a suitable scheme. An answer to the insulation problem for the general external surfaces of the aircraft is thought to have been found, and work to evaluate the proposed system, as well as to develop alternative insulations, is continuing.

Early in the investigation it became apparent that although some information was available on the physical properties of materials in the temperature range considered, with few exceptions there was no information on material thermal properties. There was also very little equipment available for the determination of these values, so it became necessary to develop experimental techniques for determining thermal conductivities and then apply these techniques to any available materials that appeared promising. The list of materials tested was restricted by the requirement for chemical stability at 500°F., but included silicones, phenolics, polyurethanes, proprietary products, and formulations developed in the Metallurgical Department laboratories.

Results of this investigation indicated that only materials with a large proportion of their volume entrapped gas would have the necessary thermal characteristics. These materials, however, all had unsatisfactory mechanical properties either at elevated temperature or throughout the temperature range. Attempts to improve the mechanical properties of foamed and similar materials were unsuccessful.

A series of tests were next carried out on composite panels comprising a representative aluminum skin, fibreglass honeycomb insulating core, and a thin fibreglass outer layer. Initial results indicated that the fibreglass core has satisfactory mechanical properties but too high a thermal conductivity. This suggested that use of a foam filled honeycomb may provide a core material with the mechanical properties of the honeycomb and yet having thermal properties very similar to that of the foam. Thermal tests on such a panel proved this to be so. It is this foam filled core that it is proposed to use as the basic external insulation on the Arrow 3.

Suitable honeycomb core materials are available commercially from manufacturers such as Hexcell and Narmco, so supply will present no problem. The foam material to be used to fill the honeycomb has not yet been finally decided on. Several suitable materials are available, one being a room temperature vulcanizing foamed silicone rubber which is, however, slightly heavier than desired, and has little mechanical strength at the densities used. A more promising material at present being developed by the Metallurgical Department is a matrix of phenolic or glass micro-balloons bound together by a high temperature resin. This material can be prepared with very low densities and yet having reasonable mechanical properties.

A final decision has also to be made on the external coating required to protect the honeycomb core from erosion and other damage. As the aircraft structure is flexible the insulation itself must be capable of considerable deformation without bond failure or structural failure in the insulation core. This dictates that the protecting external coating should be a tough skin to prevent erosion, and yet have a low modulus of elasticity and large elongation. At present it is proposed to use glass fibre reinforced silicone rubber sheeting or a similar material for this purpose generally, although on special areas of very rigid non-buckling structure, such as the wing and fin leading edges, the silicone rubber may be replaced by stainless steel or titanium sheeting.

A structural test programme to confirm the practicability of the above insulation schemes is at present under way. At the same time the general insulation investigation is continuing. Flight in the Mach 3 region is interesting many aircraft firms in the United States, and considerable effort is being diverted to the insulation problems in that country. Therefore it may confidently be expected that by the time the Arrow 3 goes into production new insulating materials will be available which may provide a more elegant solution to the problem than that given above.

3.4.2.2 Application of Insulations

Figure 3 indicates the areas of the aircraft requiring insulation. These include:

(a) all the outer skins other than the air intakes and engine nacelles.

(b) the inner surfaces of the air intake ducts between Stations 315 and 579.

(c) the fuselage central equipment area between Stations 579 and 740.

Outer skin:- The outer skins generally are insulated with the non-structural insulation described in section 3.4.2.1, e.g. a foam filled honeycomb core and reinforced silicone rubber type outer skin. The core is bonded directly to the aluminum aircraft skins in situ under pressure, and cured using heat lamps. To ensure that peeling of the insulation or outer skin does not occur due to damage in flight, positive mechanical connections to the aircraft structure are made at suitable intervals.

Regions where the aircraft structure does not buckle under load, such as the wing leading edges and some of the thicker wing skins, will, where practical, have the silicone outer skin replaced by a thin stainless steel or titanium sheet. In these cases the insulation is bonded under pressure in an autoclave to the primary skin by components. Because of the stiffness of the insulation it has a structural effect on the structure it protects, and thus permits thinner skins to be used on the primary aircraft structure. To ensure the transfer of load to the insulation it is rigidly connected at its edges to the component by mechanical attachments.

Inner surfaces of the air intake ducts:- These ducts are subject to high pressures and air temperatures approaching 700°F. Due to the high temperature, the insulation developed for the outer wing can not be used. Instead, thermal protection is provided by a high temperature insulation such as Min - K. This is contained in the duct by a thin stainless steel liner which, as its temperature rises, expands against the insulation. The liner uses longitudinal and circumferential expansion joints to alleviate thermal stresses in it.

Fuselage central equipment area:- The fuselage structure in this area is stainless steel, and the wing structure is aluminum. To protect the wing structure, and prevent the ambient temperature in the equipment bay from rising too high, a blanket of a material such as Refrasil is applied outside the bleed air shroud.

3.4.3 Wings

3.4.3.1 Inner Wing

The only structural change envisaged for the inner wing is the extension of the existing No. 3 wing tank to include the unused space forward of the

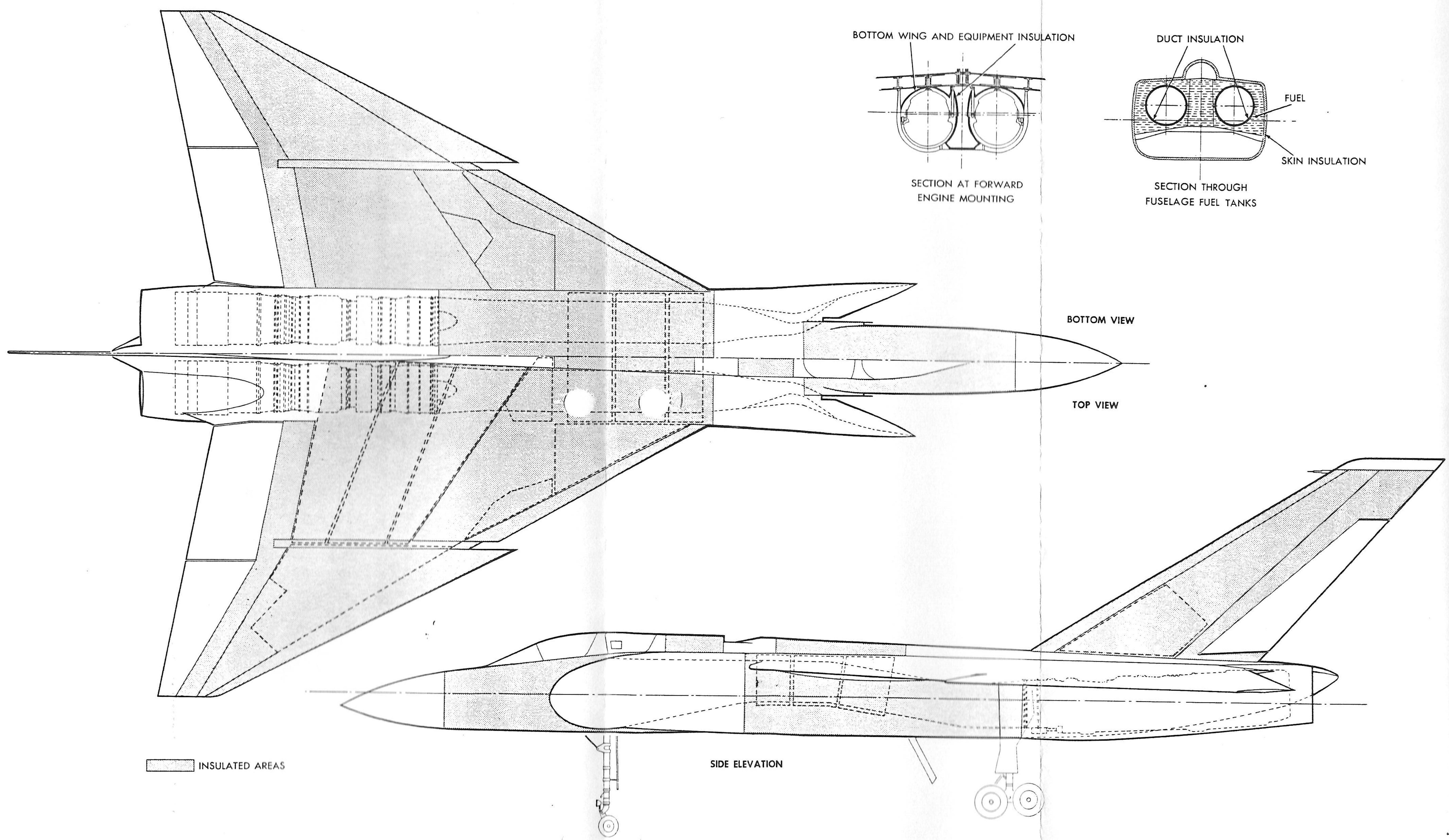

FIG. 3 ARROW 3 EXTERNAL AND INTERNAL INSULATION

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undercarriage bay as an integral fuel tank. Integral aluminum skin-stiffener panels replace the existing skins, and detail sealing changes are required. The new structure is illustrated in Figure 4. Although not shown on the illustration, all external wing surfaces are insulated.

3.4.3.2 Outer Wing

The new structure of the outer wing, and the region of it used as an integral fuel tank, are also shown on Figure 4. The geometry of the spars, ribs, and stringers, remains unchanged, but sculptured skins are needed to provide strength as a pressure vessel, cater for the additional roll inertia loads, and minimize the sealing problems. Deleting one skin joint on the undersurface allows similar skin panels to be used for the top and bottom surfaces, and provides stringer continuity. The panel joint is along the centre spar forward, and the skins are machined in the flat developed state and formed before assembly. Sections of ribs forming part of the tank boundaries are sculptured to facilitate sealing. Forged fittings, similar to those used for the inner wing tanks, are used for shear connections between stringers and ribs.

3.4.4 Fin

The fin structure, as modified for partial use as an integral fuel tank, is shown in Figure 5. Because of the increased span and tip chord some structural re-design has been necessary, but existing spar geometry has been retained near the root so as to avoid major modifications to the fin attachment box and surrounding structure. Integral skin-stiffener panels are proposed, using only two skin panels per side instead of five. The only panel joint is along Spar 10. Skins are machined in the flat developed state and formed before assembly. The root rib, and sections of rib 4 forming the tank, are now sculptured to facilitate sealing. The machined skin panels are aluminum alloy, and the whole fin surface is insulated.

Control cables and other services pass through the fuel storage region in sealed pipes to prevent fuel leakage.

3.4.5 Control Surfaces

Because of the small heat capacity of the control surfaces due to the thin skins, and the consequent necessity to use relatively thick insulation with an aluminum structure, the control surfaces have been redesigned as stainless steel structure.

3.4.6 Fuselage

3.4.6.1 Nose

The fuselage nose section geometry and structure, as illustrated in Figure 6, is changed considerably from the Mark 2 aircraft due to the new intake configuration. The intake requires that the fuselage side walls in the region of the front and rear cockpits be increased 9 inches per side at the maximum half breadth line, so that cross sections through the cockpit are now almost circular. This allows extra console width and a better positioning of some cockpit instruments and controls. Because of the increased width, a larger radome, accommodating a larger radar dish, can be incorporated if required.

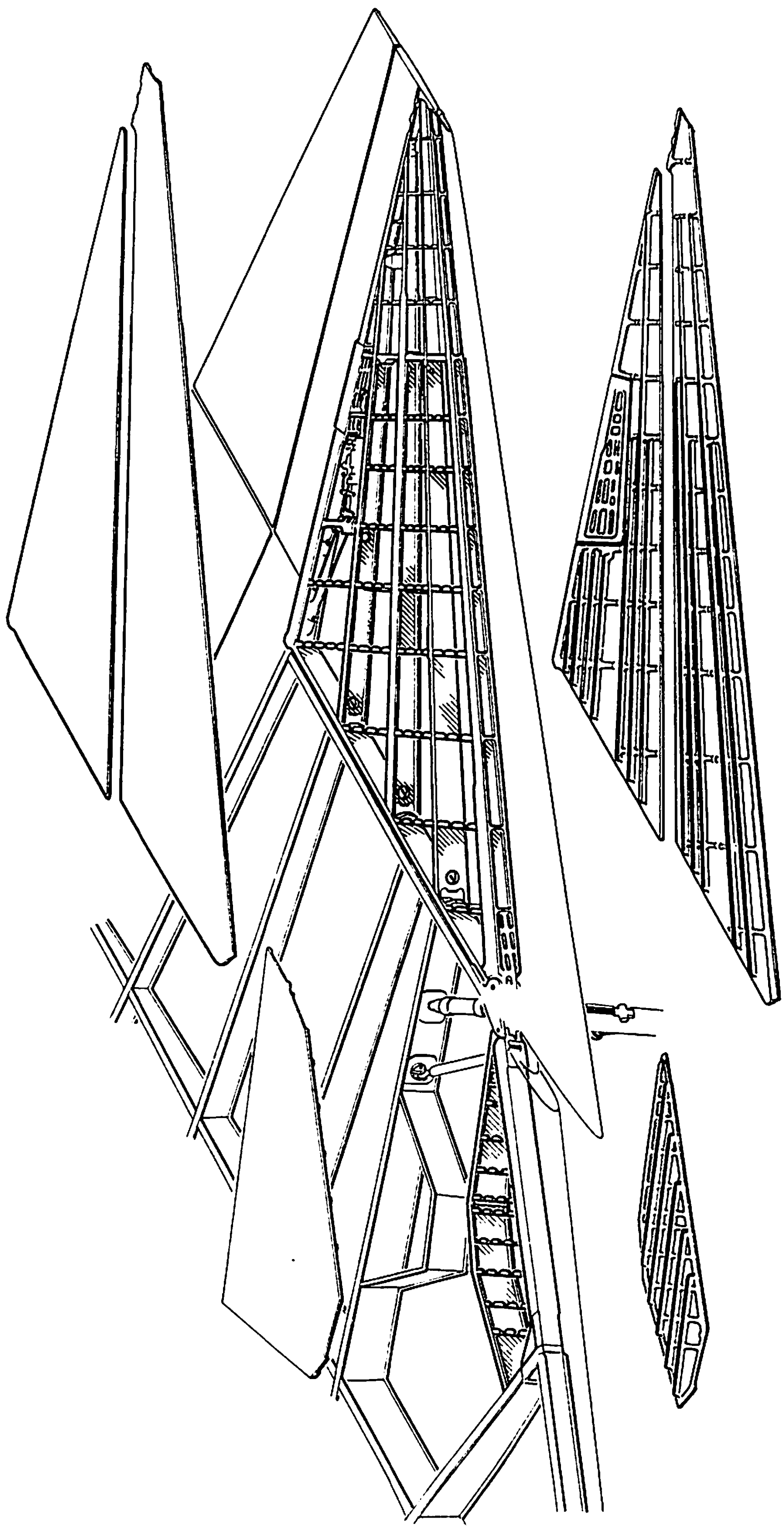

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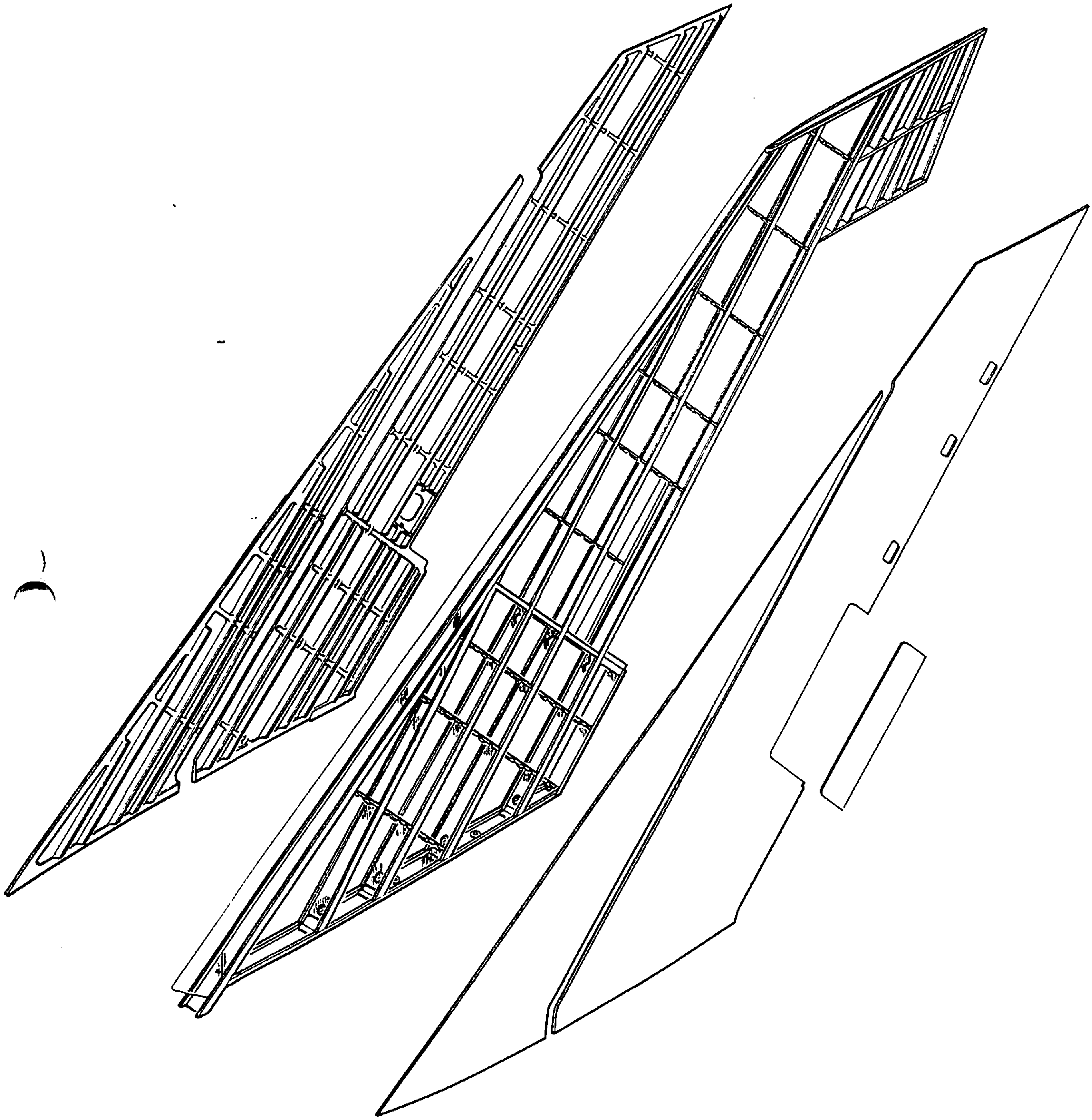
FIG. 4 ARROW 3 WING—MODIFIED STRUCTURE FOR FUEL STORAGE



MODIFIED STRUCTURE TO USE FIN SPACE FOR EXTRA FUEL

3429-105-1

FIG. 5 ARROW 3 FIN—MODIFIED STRUCTURE FOR FUEL STORAGE

The nose shear load is transferred to the centre section fuselage partly by a torque box similar to that on the existing aircraft, and partly by the air conditioning bay walls which have been made load carrying. The new torque box is, however, further aft than the box section on the existing aircraft because of the intake geometry. Its position, on either side of the air-conditioning bay, is shown in the illustration of the primary structure (Figure 6).

Primary bending loads are carried as end loads by the fuselage longerons. The top longerons are continuations of the cockpit edge members, and diffuse the end loads into the upper skins of the fuselage centre section. The nose lower longeron end load is transferred through the skin of the torque box, and transmitted to the lower longeron of the centre section as in the existing aircraft.

Fuselage frames in the intake region and aft to Station 337 support the moveable ramp loads, and provide balance paths for the duct pressure in the flat panel regions.

The new intake configuration and the centre section duct layout combine to reduce the available width for the air-conditioning bay by approximately 5 inches per side. However, the bay is lengthened by 22 inches so that the volume is approximately the same. The ram air duct, bringing air from the shock diverter plate to the air-conditioning system, is made integral with the side wall structure to give a compact and efficient installation.

For temperature considerations, and because of the difficulty in insulating the intake shell, basically steel structure is used forward of station 337 and outboard of the air-conditioning walls. The cockpit walls and radar nose skins are of insulated aluminum alloy.

3.4.6.2 Intake

The Mach 3 performance requirement dictated that a variable geometry inlet be incorporated in the Arrow 3 design. The intake and variable ramp finally selected are illustrated in Figures 6 and 7. The intake is a swept nose configuration with the compression ramps outboard. The intake throat then projects well inboard into the fuselage cross-section, avoiding the necessity to increase the fuselage width in this region and permitting the external shape to be nearly elliptical. This gives an efficient structure, and provides sufficient room between the duct wall and outside skin for the variable ramp mechanism.

The leading edge of the inlet starts at the same fuselage station as the existing ramp on the Arrow 2 aircraft, that is, approximately at Station 160. Since the ramp on the new inlet is approximately 80 inches long, as against 43 inches for the existing ramp, the throat is moved aft about 37 inches. To retain as long a diffuser as possible, the start of the 37 inch diameter circular duct section is moved aft from Station 330 to Station 337.

There are three fixed ramps followed by a fourth variable ramp at the throat. The variable ramp is made in two sections, one hinged at the front at the junction of the third and fourth ramps, and one hinged at the rear downstream of the throat. There is an overlapping sliding joint between

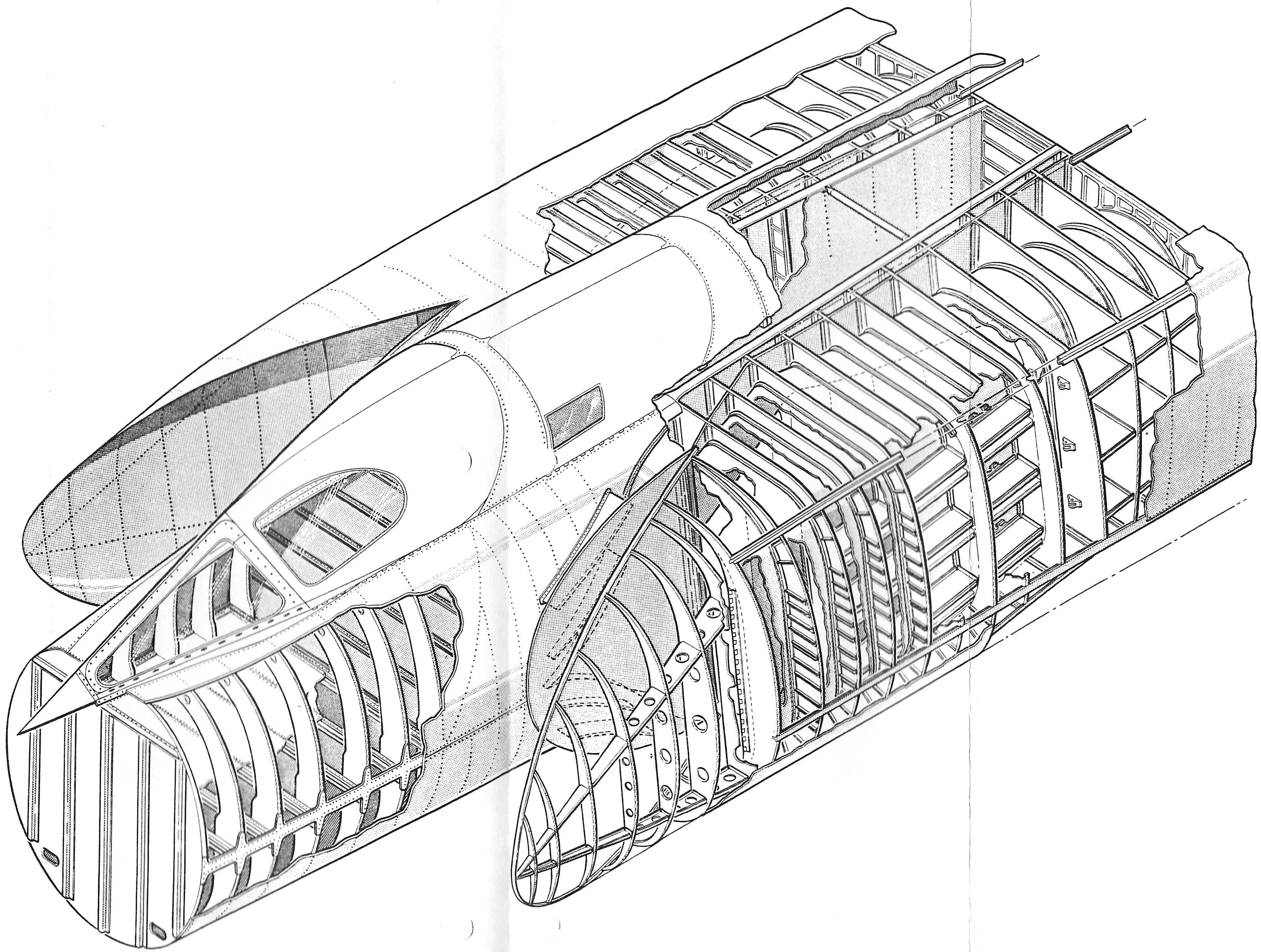


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FIG. 6 ARROW 3 FRONT FUSELAGE WITH REDESIGNED INTAKES

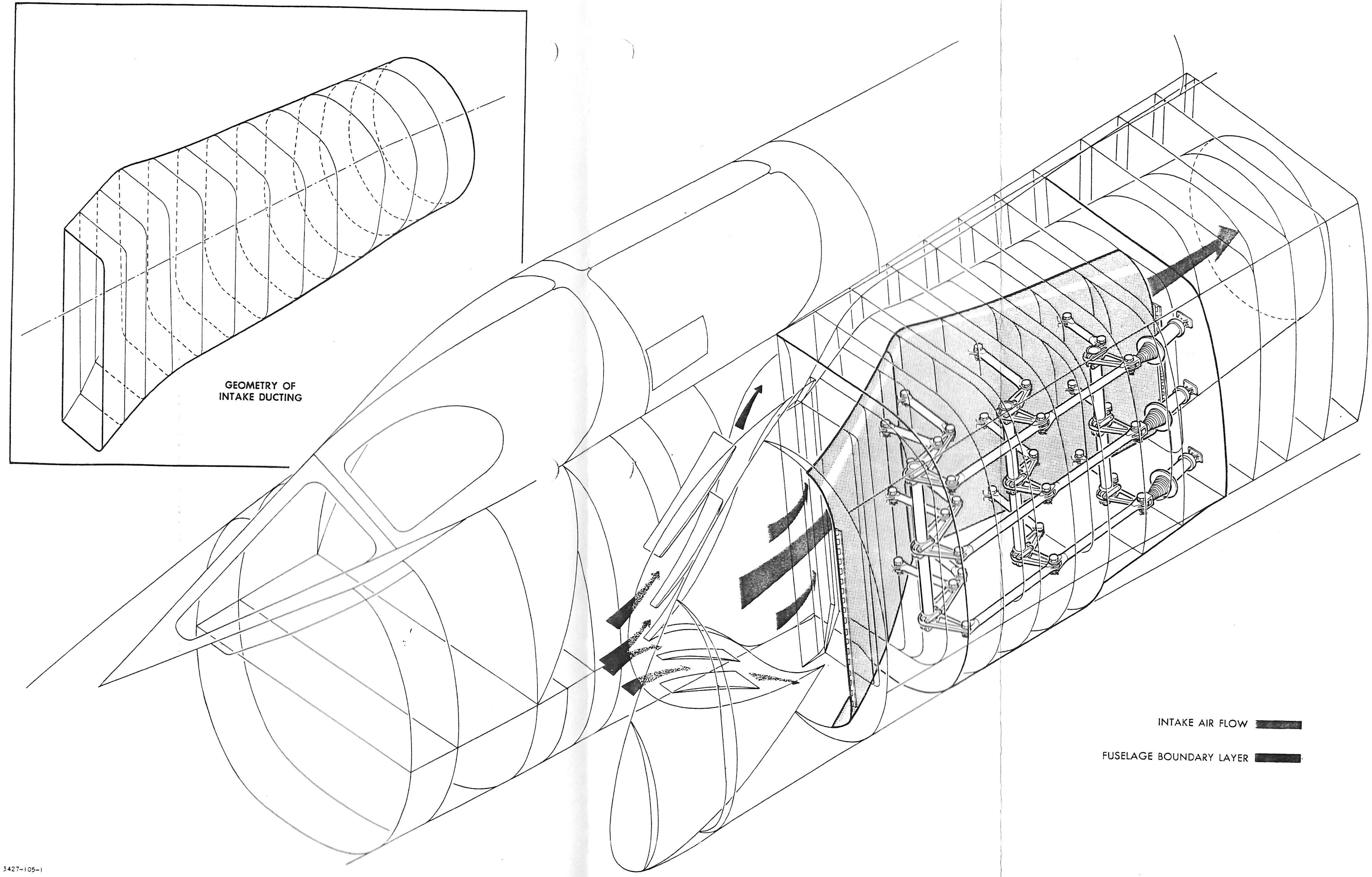

FIG. . VARIABL. RAMP AND OPERATING MECHANISM the two sections at the centre. The geometry of the moveable portion of the ramp is arranged to give a smooth area distribution along the diffuser portion of the duct throughout the design Mach number range.

At high Mach Numbers the mechanism operating the variable ramp is subjected to high temperatures; therefore it is designed to have a minimum number of bearings. The system consists of bellcranks and connecting rods mounted in three banks 16 inches apart. Plain bearings, using a solid lubricant, are proposed because of the high temperatures. Each bank is moved by a separate hydraulic jack.

In order to reduce the high pressure loads on the moveable ramp and operating mechanism during high speed flight, air is bled off the intake in the region of the rear hinge and ducted to the mechanism bay. This bleed air decreases the pressure differential across the ramps, thus reducing the loads on the mechanism. The jacks are housed in a separate insulated bay immediately aft of the ramp operating linkage: this ensures that the hydraulic system temperatures do not exceed allowable values.

3.4.6.3 Centre Fuselage

By simple small changes in geometry, and the use of sculptured structure, the centre fuselage from Station 337 to Station 469 is redesigned as a three cell integral fuel tank. The new structure is illustrated in Figure 8. Adjacent primary structure, such as the heavy frame at Station 485, the fuselage side rib, the auxiliary spar, and the main attachments to the inner wing, remain unchanged. All fuselage frames and fuel tank pressure bulkheads forward of Station 414 are now full width across the fuselage section, but remain normal to the fuselage datum. Aft of Station 414, to Station 469, the frames are normal to the duct sloping axis. In plan view the duct centre lines are now parallel extensions of the engine centre lines. This means that all intersections of the duct with the frames will be circular and at right angles, greatly simplifying the machining operations and also easing the sealing problem.

The proposed centre fuselage section of the duct is a large diameter cylinder with integral, rectangular sectioned, stiffening rings. The inside diameter of the duct structure is now 37.0 inches, so that a minimum internal diameter of 36.0 inches will be available after the duct insulation is installed. Each duct section has flanged ends for bolted assembly and incorporates a circumferential seal. Fuselage frames are located by, and attached to, the integral stiffening rings. Each fuel tank pressure bulkhead locates on a thickened section of the duct cylinder between two stiffening rings and is attached by two circumferential lines of bolts with sealing grooves between the mating surfaces. This restricts the leak path to six circumferential lines of bolts and the flanged joints along the duct length.

The pressure bulkheads are basically "I" section to provide two lines of fastenings around the fuselage and duct walls, and the armament bay roof. The bulkhead is manufactured in two sections, with a sealed joint across the duct horizontal centre line.

The frames have a vertical joint at the aircraft centre line, and a horizontal joint across the duct centre line. The frame flanges are slotted to allow stringer continuity, and the webs cater for fuel flow and surge requirements.

The armament bay roof panels form the bottom panels of the fuel tanks. They are structure instead of fairing, eliminating some existing Arrow 2 tubular structure. The geometry changes from the Mark 2 aircraft give a panel shape which is a constant section from Station 337 to Station 414 and is simple to produce and seal. The panel stiffeners are bonded onto the outside surface of the tank. This avoids cutouts in the shallow sections of the frames, and allows fastenings outside the tank for ancilliary equipment.

3.4.6.4 Rear Fuselage

The design of the rear fuselage is based on the existing Arrow 2 lines. The tunnel geometry is altered somewhat to cater for the mass flow requirements of the bypass air, and some local changes in the outer profile are required to take care of the larger afterburner of the Iroquois Series 3 engine. A fixed geometry ejector, with a disposable plug for increased subsonic performance, is used as in the Arrow 2. The main structural changes are in the deletion of the existing bypass gills just ahead of the engine, and the substitution of bypass air overflow doors aft of the engine intake. These new doors are incorporated into the main engine access doors.

The rear fuselage structure (Station 485 aft) is fabricated basically of stainless steel. Because stainless steel is used, no insulation is required on the outside surfaces, but the underside of the wing and the inside of each bypass duct are insulated to form an insulated equipment bay.

The parabrake installation is revised to cater for the larger diameter engines.

3.4.7 Undercarriage

3.4.7.1 General

The increase in all up weight between the Arrow 2 and Arrow 3, together with the consequent increase in landing speeds, imposes rather severe operating conditions on the undercarriage of the Arrow 3. Therefore considerable modifications to the Arrow 2 undercarriage may be necessary before it can be adopted. The modifications, however, appear to present no critical problems.

The effect of increased weight on the aircraft structure can be minimized by a reduction in the dynamic reaction factor. This factor is related to the shock absorber characteristics, and can be reduced by an increase in stroke and/or an increase in efficiency. To support the heavier aircraft, either larger tires or higher tire pressures are required, and if the latter are adopted an increase in the permissible runway unit construction index over that used in Arrow 2 design will be necessary. The additional weight and higher landing speeds also require the brakes to have larger energy absorption characteristics, and this calls for a redesigned brake unit. The undercarriage and wheel structures also require strengthening.

The effect of increased temperature is not important. Owing to the large heat capacity of the undercarriage units, their structural and tire temperatures remain well below the maximum value of 350°F. permitted for the general structure. Therefore no difficulties due to thermal effects are expected.

It is proposed that the undercarriage modifications be based on the use of an increased runway unit construction index, but agreement to do this will first be obtained from the relevant authorities. This proposal is not impractical, because several present day aircraft use far higher tire pressures than the Arrow 2. It is also possible to considerably increase the efficiency of the shock absorbers over those of the Arrow 2, and this will be used to reduce the dynamic reaction factor to the desired value for the Arrow 3. The brakes themselves present no problem. Considerable advances that have been made in the use of new materials in brake design permit the efficiency of brakes to be increased considerably. Use of these materials is at present in the laboratory stage, but it is expected that they will be used in production within the predicted time schedule of the Arrow 3 development. Their use permits the required energy absorption by a brake system very little larger than that of the Arrow 2, Because of the higher landing speeds, the use of higher pressure "Low Profile" tires will also be adopted.

This approach will have to be changed if the use of an increased runway unit construction index is not acceptable. In this case considerable redesign of the main undercarriage may be necessary.

3.4.7.2 Nose Undercarriage

The nose undercarriage presents few problems. Developments in material properties since its design for the Arrow 2 permit it to be modified to carry the Arrow 3 loads by retaining the existing physical dimensions and merely changing the materials of which it is constructed. However the wheel may have to be strengthened to meet the increased loads.

3.4.7.3 Main Undercarriage

A change in materials is not sufficient to enable the main undercarriage units to be used on the Arrow 3. Certain increases in sectional sizes of the leg and bogie, as well as a strengthening of the wheels themselves, are also necessary to enable the increased loads of the Arrow 3 to be borne. These modifications, however, do not affect the basic principles of the undercarriage design and operation.

3.5 Fuel Storage, Capacity, and System

For the Arrow 3, about 190 cubic feet of unused internal space in the aircraft is converted into extra storage space for fuel, still keeping the aircraft aerodynamically clean and stable. This means an estimated 1,192 gallons of extra fuel can be stored in the centre fuselage, inner wing, outer wing, and the fin. Fuel in these locations satisfies balance and combat centre of gravity aft requirements. The new tanks are shown on the structure illustrations, Figures 4, 5 and 8.

Based on a 15% enclosed space deduction for structure, fuel system equipment, fuel expansion, and trapped fuel, the capacities of the new tanks are:-

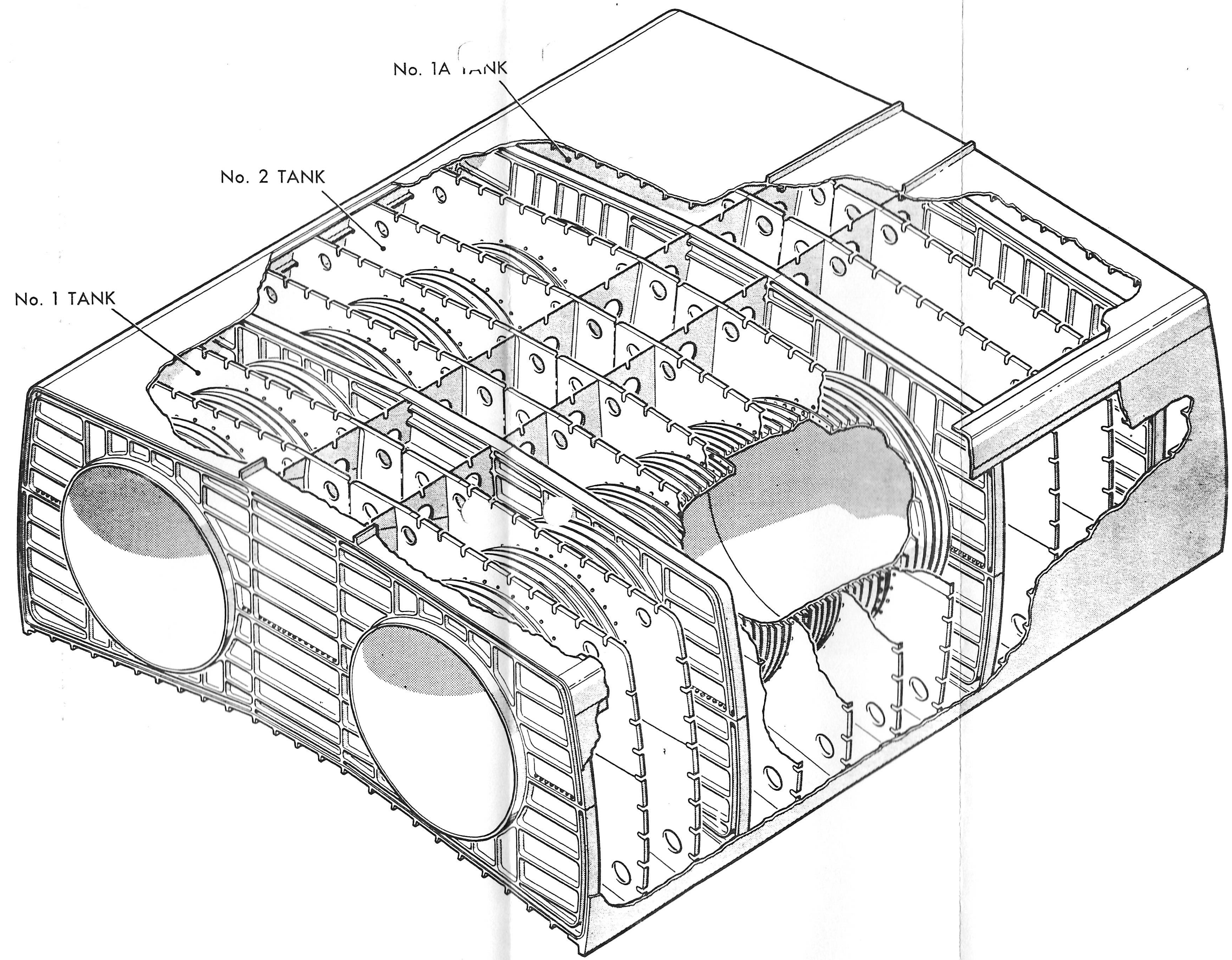

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FIG. [illegible] ARROW 3 CENTRE FUSELAGE

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Centre Fuselage	1,198 gallons	
Inner Wing Cell	96 gallons	(48 gals. per side)
Outer Wing	308 gallons	(154 gals. per side)
Fin	95 gallons	
Total Capacity	1,698 gallons	

The new centre fuselage, however, deletes the No. 1 and No. 2 bag tanks, so that the increased capacity is 1,698 - 506 = 1,192 gallons. The total Arrow 3 capacity is then 3,684 gallons.

The new centre fuselage tank is divided into 3 cells. The middle cell, and the fin tank, supply one engine, and the front and rear cells feed the other engine. This allows an approximately symmetric feed sequence to both engines, and satisfies balance requirements. The Arrow 2 fuel system is extended to integrate the proposed new tanks into a revised fuel system. The general use of sculptured structure simplifies the new fuel system and reduces the feed, flow, access, servicing and sealing problems. The new system is illustrated in Figure 9.

The fuel system illustrated assumes an in-flight refueling requirement for the Arrow 3. A vertical rising probe located ahead of the pilot's canopy is proposed for this system since the increased fuselage nose width dictated by the new intakes gives space for a probe and a feed pipe running aft into the fuel system as shown.

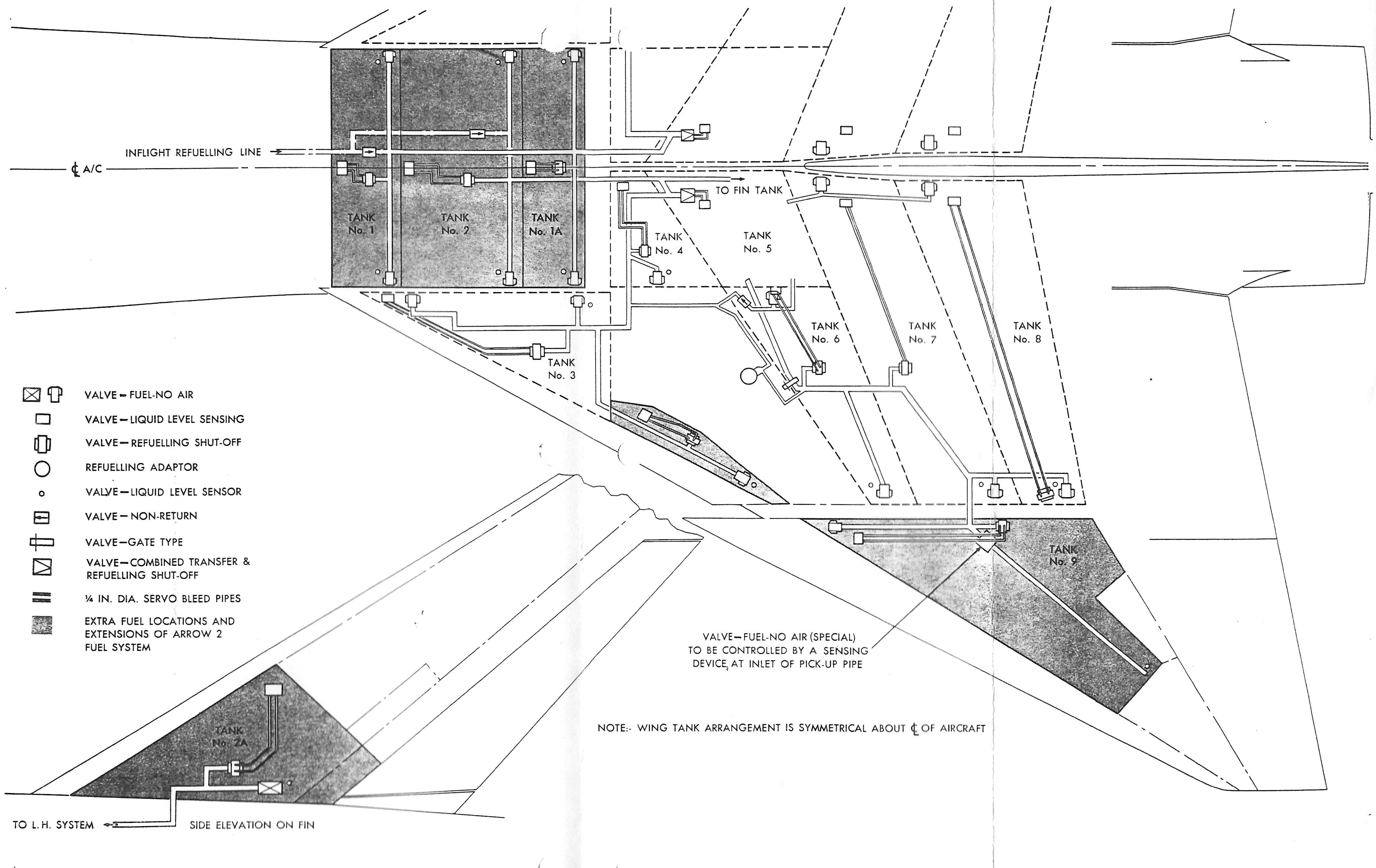

FIG. 9 ARROW 3 FUEL SYSTEM

3.6 Systems Modifications

3.6.1 General

The maximum operating temperature of the Arrow 3 hydraulic systems determines whether Arrow 2 equipment can be used without modification, or whether a new line of equipment has to be developed. The results of a preliminary investigation indicates that, in general, the Arrow 2 equipment is satisfactory, but this must be confirmed by a more complete investigation which includes a detailed study of the system components.

The Arrow 2 systems have a maximum operating temperature of 250°F. to 275°F., and are installed in a structure which has a maximum temperature generally of 250°F.. The Arrow 3 structure has a maximum temperature generally of 350°F., but due to the insulation this temperature is a transient condition, not a saturation condition, and the air temperature inside the structure, as well as much of the structure, will be less than 350°F.. The hydraulic systems are a good heat sink, and estimates, based on the length of time the structural temperatures will be above 250°F., indicate that the higher structural temperatures of the Arrow 3 will have negligible effect on the systems temperatures. Therefore, it is proposed that the Arrow 2 hydraulic systems generally be used without change for the Arrow 3.

Should further investigation show that the system temperatures are larger than at present expected, high temperature systems will have to be used. Such systems are at present in use on other aircraft, notably the B-58

Hustler, and are covered by standard specifications. A system operating at temperatures up to 400°F. is in the U.S.A.F. Mil. Spec. Class III Temperature Classification, and would be the most likely replacement. It uses a synthetic hydraulic fluid, neoprene based seals, and nickel plated fittings. Change to such a system would only take place if it could not be avoided, and would be dependent on the result of tests on Arrow 2 systems in an Arrow 3 environment.

The major modification in the Arrow 2 systems to suit them for the Arrow 3 is the change to oil-fuel heat exchangers for high Mach number flight. The necessity for this change is stated in Section 3. 3. 4.

3. 6. 2 Flying Control Hydraulic System

No changes in control surface areas, angles of travel, or rates of travel, are expected, therefore the Arrow 2 flying control hydraulic system is adopted unchanged. Particular attention will, however, have to be paid to shrouding and insulation of the actuator mechanisms where they are in areas vented to atmosphere.

3. 6. 3 Utility Hydraulic System

No fundamental changes are at present proposed to the utility hydraulic system of the Arrow 2 to suit it for use in the Arrow 3. The sizes of several components may, however, require to be increased.

The speed brake and nose wheel steering sub-circuits require no modification. The design loads on the armament sub-circuit are determined by sea level flight conditions, therefore this circuit also should require no alteration. The undercarriage actuating sub-circuits remain unchanged unless a modified main undercarriage is required (See Section 3.4.7), when the main undercarriage retraction gear may require redesign. The wheel brake circuits require no changes apart from the larger brakes necessary to compensate for the higher aircraft weights.

3.6.4 Intake Ramp Actuation Sub-circuit

An additional hydraulic circuit is added to the Arrow 3 to operate the intake ramp actuators. The intake ramp position is a speed dependent variable, therefore the circuit must be of the continuous operating type. If a suitable accumulator is installed it may be possible to couple this circuit to the utility hydraulic circuit. Alternatively, it can be connected directly to the flying control circuit. Further investigation of the effects of this addition to the existing circuits is required, but it is expected that only minor systems modifications will be necessary.

3.6.5 Air Conditioning

An investigation of the air-conditioning requirements of the Arrow 3 indicates that only one major change in the Arrow 2 system is necessary. This change is an increase in the boiler capacity by approximately 40% to allow for the higher recovery temperatures associated with Mach 3.0 flight. Development will also be required on some of the bought out equipment, to meet the higher pressure and temperature requirements.

3. Weight Analysis

3. 7. 1 General

The basis of estimation for the Arrow 3 aircraft weight has been to develop the current weights for the Arrow 2 version on a "weight penalty concept" to meet the performance requirements, environmental conditions, and modifications, as outlined throughout this brochure. It is thought that this form of estimation is realistic and will predict a weight very close to that attainable in practice.

The weights are based on the assumption that there are no major changes in equipment between the Arrow 3 and the Arrow 2.

2 Increments in Non-Structural Weight

During design development the requirements for non-structural items have been well defined. The following weight increments are based on these requirements.

Power Plant and Services:-

Brochure weight of the Orenda Series 3 Iroquois, 5,445 lb. per engine.

Estimated deletion for divergent nozzle called up in Series 3 Iroquois brochure but not used on Arrow 3 is 450 lb./engine.

Nett increase over Series 2 is 495 lb./engine.

Estimated fuel system increase for additional tankage, 180 lb.

Engine mounting and services increment, 70 lb.

Additional Internal Fuel:-

Available volume for tankage allows for 9,300 lb. of additional internal fuel in 2 outer wing tanks, 2 new inner wing tanks, revised centre-section fuselage tanks, and fin tankage.

Equipment, Fixed and Removable:-

Preliminary air-conditioning requirements indicate a 100 lb. increase in system weights. Based on increases to Arrow 2 equipment since the figure quoted for March 1st., 1958 (See Table 3), a nominal increase of 150 lb. for the remainder is allowed for, assuming no major changes in the types of equipment carried by the Arrow 3.

Useful Load Less Fuel:-

Residual fuel consistent with the additional tankage is estimated at 160 lb. Water for air-conditioning is increased by 100 lb. as indicated by preliminary requirements.

Nominal increases are allowed for engine oil and fire extinguishing fluid.

Insulation:-

Preliminary studies on the insulating problem indicate that the total weight of insulation and attachments required to limit the skin temperature to 350°F. is of the order of 2,950 lb.

3.7.3 Gross Weight Estimation

An estimate of the gross weight, based on the use of growth factors and allowing for changes in materials and in design, gives an all up weight of 82,250 lb., which is within 1/4% of the value determined from a more detailed analysis. This weight was used to derive a stressing weight for preliminary design purposes, and as it agrees very well with the later, and more detailed analysis, has been retained for all structural design.

3.7.4 Stressing Weight

For a gross weight of approximately 82,250 lb., and a full internal fuel load of 28,738 lb., a logical combat or stressing weight with 50% fuel gone is 67,881 lb. This has been rationalized, for design purposes, to a stressing weight of 68,000 lb.

3.7.5 Allowable Limit Load Factor

The Arrow 2 stressing weight is 47,000 lb., and the allowable limit load factor at this weight varies from 7.33 for subsonic flight to 6.0 for flight at Mach 2.0. Factoring these values to allow for the increase in stressing weight, increase in structural material, change in material properties, and structural modifications, the allowable limit load factors for the Arrow 3 are at least 5.5 in subsonic flight and 5.2 at M = 3.0.

3.7.6 Increments in Structural Weight

The incremental increases in structural weight are based on a preliminary detail analysis of the modified structure. Details are as follows:

Undercarriage:-

Using approximate methods based on R.A.E. Structures Reports 80 and 198, but modified to conform with present day practice, an increase in weight of 350 lb. is estimated for a landing weight of 59,111 lb. and take-off weight of 82,250 lb.

Flying Controls Group:-

Based on increases in this group during Arrow 2 development from March 1st., 1958, a weights increment of 320 lb. is allowed.

Structure:-

From preliminary scheme drawings covering major structural changes as outlined in this brochure, and based on consideration of material changes and the modified loading conditions, the increase in structural weight is 2,204 lb. This is made up as follows:

Outer Wing:- Modifications to the existing structure to permit its use as a fuel cell, allowing for effects of fuel pressure loads, sealing, and accommodation of fuel system, gives an increased weight of 150 lb. Although flight case primary loads are similar to those on the Arrow 2, landing cases become more critical in the region of the transport joint.

Inner Wing:- Increases to compression skin gauges as indicated by a preliminary stress office investigation, modifications for additional fuel cells, and allowances for undercarriage accommodation. Weight increment is 250 lb.

Fin and Rudder:- Modification to the existing structure due to the increased fin size and for use of the fin root as a fuel cell, and allowing for the effects of fuel pressure loads and system changes, gives a weight increment of 260 lb.

Fuselage:- Forward of Sta. 255 - The increased structure required for fairing in to the new intakes and the additional weight of materials due to canopy modifications gives a weight increment of 250 lb. This includes a nominal allowance for nose undercarriage load increases.

Sta. 255 to Sta. 485 - Modification to the fuselage fuel cells, intake re-design, and allowances for the increased intake duct temperatures and pressures, gives a weight increment of 650 lb. There is an extensive use of steel in the variable ramp and intake regions, and a replacement of magnesium skins with aluminum. The additional fuel capacity also requires considerably increased skin gauges.

Sta. 485 to Sta. 591.625 - Allowances for the increased intake duct temperatures and pressures and a nominal skin gauge increase for the external skin gives a weight increment of 100 lb.

Sta. 591.625 to Sta. 742.5 - The addition of air spillage doors to the engine access doors, and the general change to a stainless steel structure, gives a weight increase of 280 lb.

Sta. 742.5 aft - The change to stainless steel structure, and allowances for the new ejector, gives a weight increase of 250 lb.

Marry Up:- In addition to the above increments an additional increase in weight of 14 lbs. has been allowed for small quantities not included in the above detail summary, such as structural joints, etc.

3.7.7 Weight Summary and Breakdown

Table 3 presents a preliminary breakdown of the Arrow 3 weight with increments over the Arrow 2 components as of March 1st., 1958. A gross weight of 82, 392 lb. is shown for the maximum internal fuel condition. This agrees substantially with the figure of 82, 250 lb. developed for initial design purposes. All figures quoted are preliminary only, and will be varied as detail design progresses, however, the values in general are felt to be realistic at this stage.

TABLE 3

WEIGHT SUMMARY OF ARROW 3

Description	Arrow 2 Weight March 1st., 1958	Arrow 3 Estimated Weights	
		ΔW	W
Outer Wing	2,624.15 lb.	150 lb.	
Inner Wing	7,397.74 lb.	250 lb.	
Fin and Rudder	1,034.75 lb.	260 lb.	
Fuselage - Forward of Sta. 255	2,571.78 lb.	250 lb.	
Sta. 255 to 485	1,700.73 lb.	650 lb.	
Sta. 485 to 591.625	1,151.34 lb.	100 lb.	
Sta. 591.625 to 742.5	1,574.10 lb.	280 lb.	
Sta. 742.5 Aft	1,036.95 lb.	250 lb.	
M[illegible]ry Up	51.61 lb.	14 lb.	
STRUCTURE	19,143.15 lb.	2,204 lb.	21,347 lb.
INSULATION + ATTACHMENT		2,950 lb.	2,950 lb.
LANDING GEAR	2,584.25 lb.	350 lb.	2,934 lb.
Engine and Accessories	9,186.78 lb.	990 lb.	
Fuel System	716.33 lb.	180 lb.	
Engine Mountings	132.38 lb.	20 lb.	
Engine Services	765.18 lb.	50 lb.	
POWER PLANT AND SERVICES	10,800.67 lb.	1,240 lb.	12,040 lb.
FLYING CONTROLS GROUP	1,792.80 lb.	320 lb.	2,113 lb.
Air Conditioning System	856.00 lb.	100 lb.	
Remainder	8,069.18 lb.	150 lb.	
EQUIPMENT FIXED AND REMOVABLE	8,925.18 lb.	250 lb.	9,175 lb.
AIRCRAFT BASIC WEIGHT	43,246.05 lb.	7,314 lb.	50,560 lb.

TABLE 3 (Continued)

Description	Arrow 2 Weight March 1st., 1958	Arrow 3 Estimated Weights ΔW	W
AIRCRAFT BASIC WEIGHT	43,246.05 lb.	7,314 lb.	50,560 lb.
Crew	390.00 lb.		
Oil	138.97 lb.	25 lb.	
Engine Fire Extinguishing Fluid	25.00 lb.	10 lb.	
Missiles	1,728.00 lb.		
Residual Fuel	218.40 lb.	160 lb.	
Oxygen Charge	13.39 lb.		
Water for Air-Conditioning	285.00 lb.	100 lb.	
USEFUL LOAD	2,798.76 lb.	295 lb.	3,094 lb.
OPERATIONAL WEIGHT EMPTY	46,044.81 lb.	7,609 lb.	53,654 lb.
Maximum Internal Fuel	19,438.00 lb.	9,300 lb.	28,738 lb.
A.U.W. MAXIMUM INTERNAL FUEL	65,482.81 lb.	16,909 lb.	82,392 lb.
Maximum External Fuel + External Tank	4,248.00 lb.		4,248 lb.
A.U.X. MAX. EXTERNAL + INTERNAL FUEL	69,730.81 lb.	16,909 lb.	86,640 lb.
COMBAT WEIGHT (1/2 Int. Fuel)			68,023 lb.
NORMAL LANDING WEIGHT (1/4 Int. Fuel, less missiles)			59,111 lb.

4 PERFORMANCE OF THE ARROW 3

4.1 Propulsion Performance

For performance purposes the nett thrust of the propulsion system is based on the pressure recovery of the intake shock system coupled with an assumed duct loss of 6%. The additive drag is derived by assuming that during subcritical intake operation all spillage takes place through the terminal normal shock wave. This assumption is conservative.

Engine performance is estimated from unpublished data supplied in the form of non-dimensional curves for the Iroquois 3 by the engine manufacturers. Ejection characteristics are based on theoretical considerations.

Predicted values of the installed nett thrust and the fuel consumption for various engine and flight conditions are given in Figs. 10 to 15 inclusive.

4.2 Drag

The drag estimate of the Arrow 3 is based on the predicted values for the Arrow 2. Accurate drag estimation is a complicated process which involves a considerable amount of work. The Arrow 2 values were obtained from theoretical considerations, as well as the results of wind tunnel tests and free flight models, and flight test results on the Arrow 1 have indicated that they are of the right order and may be conservative. Therefore it is considered that the use of Arrow 2 drag curves, extrapolated to a Mach number of 3.0 and then corrected to allow for any changes in the configuration between the Arrow 2 and the Arrow 3, will give far more reliable values than are likely to be obtained by other than the most complex methods of analysis.

The corrections used to convert the Arrow 2 values to Arrow 3 values are a function of the Mach number. Subsonically the nett correction is that due to the Arrow 3 intakes, while at supersonic speeds allowances are made for the effects of intake and afterbody changes, movement of the centre of gravity, and other changes. The supersonic allowance for all effects other than the intake incremental additive drag is estimated to be a reduction in the Arrow 2 drags varying with Mach number from 0 to 20% in the range $1.0 \leqslant M \leqslant 1.7$, and a constant 20% reduction for Mach numbers greater than 1.7. The intake incremental additive drags are derived from theoretical considerations, and added to the drag values obtained by the above method. This intake incremental additive drag is zero at Mach numbers greater than 2.5, because here the intake operates supercritically and swallows the design capture mass flow. Results of this drag analysis are presented in carpet form in Figs. 16, 17, and 18.

It is thought that this method of analysis gives realistic drags which should be attainable with little difficulty. As an example of the derivation, consider the conditions when the aircraft is flying at a height of 60,000 feet and Mach number of 2.0. Assuming a centre of gravity at 32% m.a.c., a saving of 50 drag counts on the Arrow 2 results is necessary to give the predicted Arrow 3 values. An area rule investigation into the effect of increasing the ejector unit diameter to the 53 inches of the Arrow 3 indicates a saving of 90 drag counts, and as the Arrow 3 intakes have a smoother area distribution than the Arrow 2 intakes, the total saving will be larger still. However, due to limitations in the method of estimation, it is thought that these results must be treated with considerable reserve, and a saving of 50 drag counts has been taken as a realistic value. It is possible that further drag reductions can be made, e.g. if stability considerations permit the leading edge droop to be removed there will be an additional saving of about 10 drag counts.

4.3 Performance Weights

The weights used for performance estimates are those given in Section 3.7 Table 3. The A. U. W. with maximum internal fuel is 82,392 lb., combat weight with half the maximum internal fuel is 68,023 lb., and operational weight empty 53,654 lb.

Predicted Performance

Predicted values of the Arrow 3 performance are given in Tables 4 to 10 inclusive, and Figures 19 to 24 inclusive following. As stated in the preceding text the considerations on which these estimates are based are felt to be generally realistic, therefore, it is confidently expected that these predicted values will be attainable in practice.

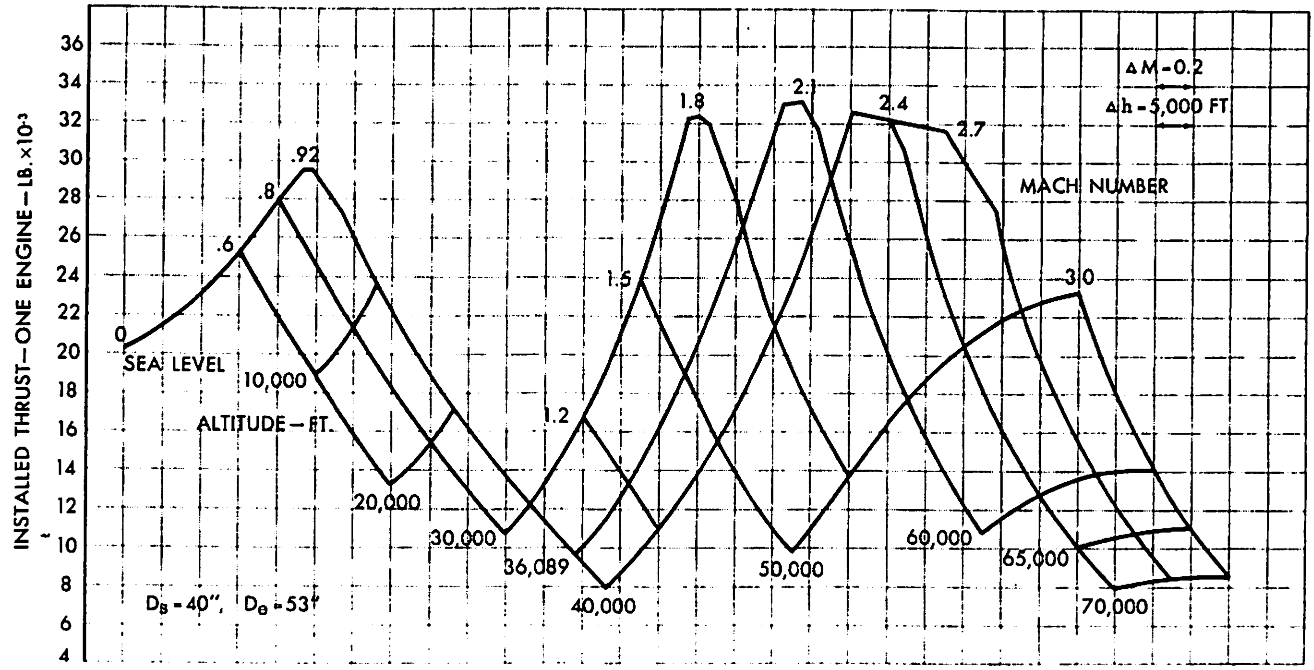

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FIG. 10 SERIES 3 IROQUOIS ENGINE-INSTALLED THRUST, A/B LIT

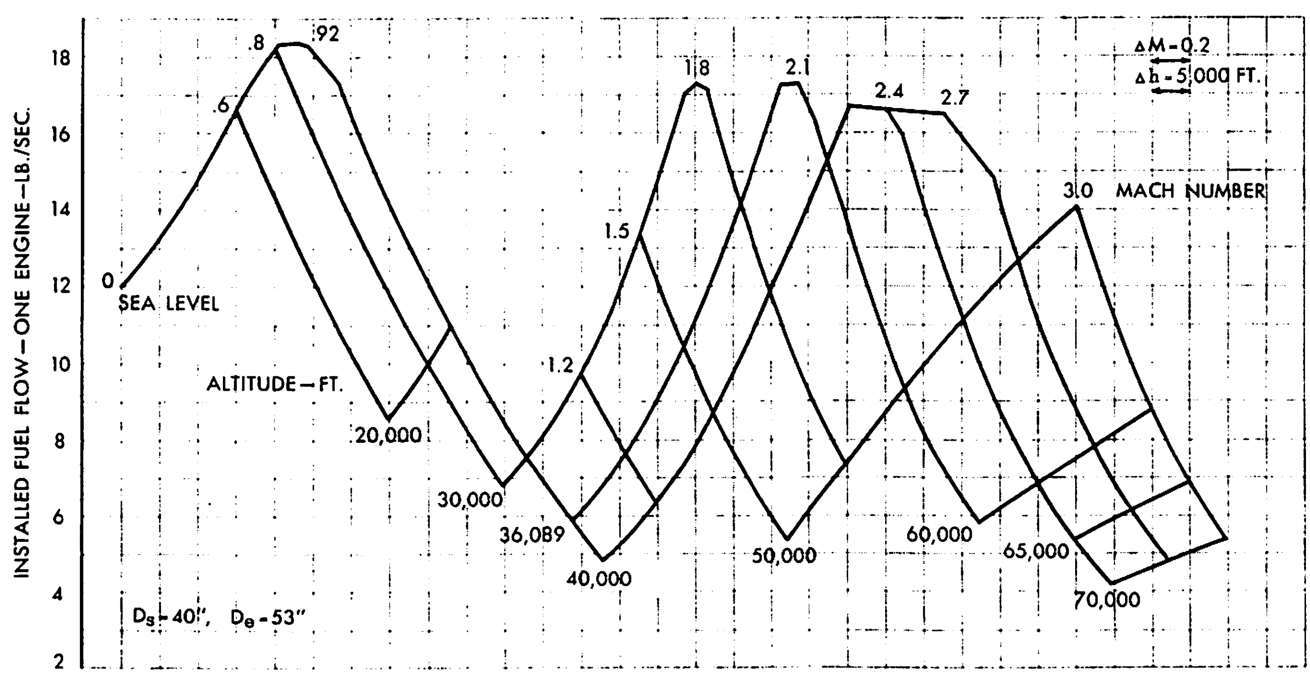

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FIG. 11 SERIES 3 IROQUOIS ENGINE-INSTALLED FUEL FLOW, A/B LIT

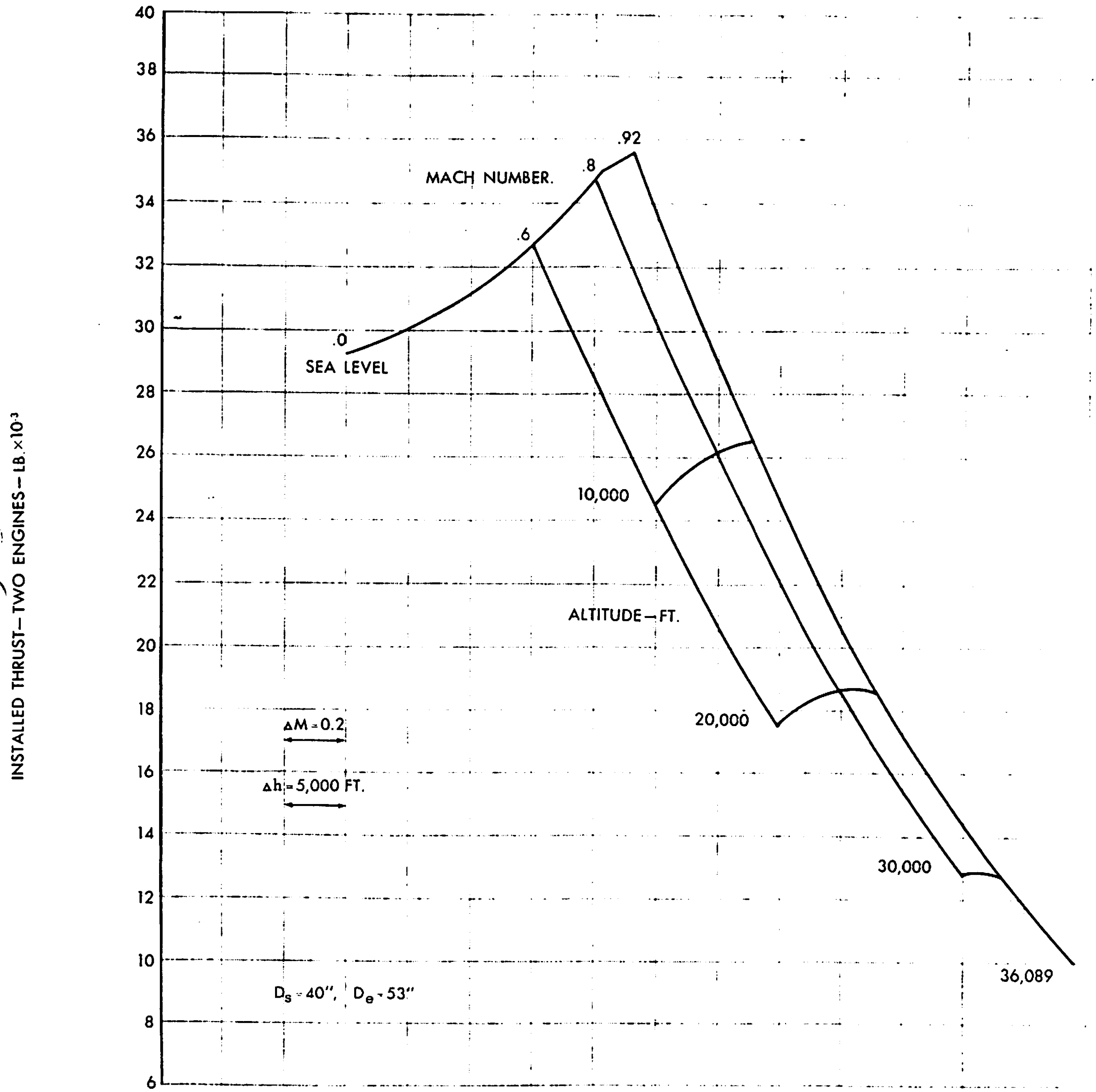

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FIG. 12 SERIES 3 IROQUOIS ENGINE—MAXIMUM INSTALLED THRUST, A/B UNLIT

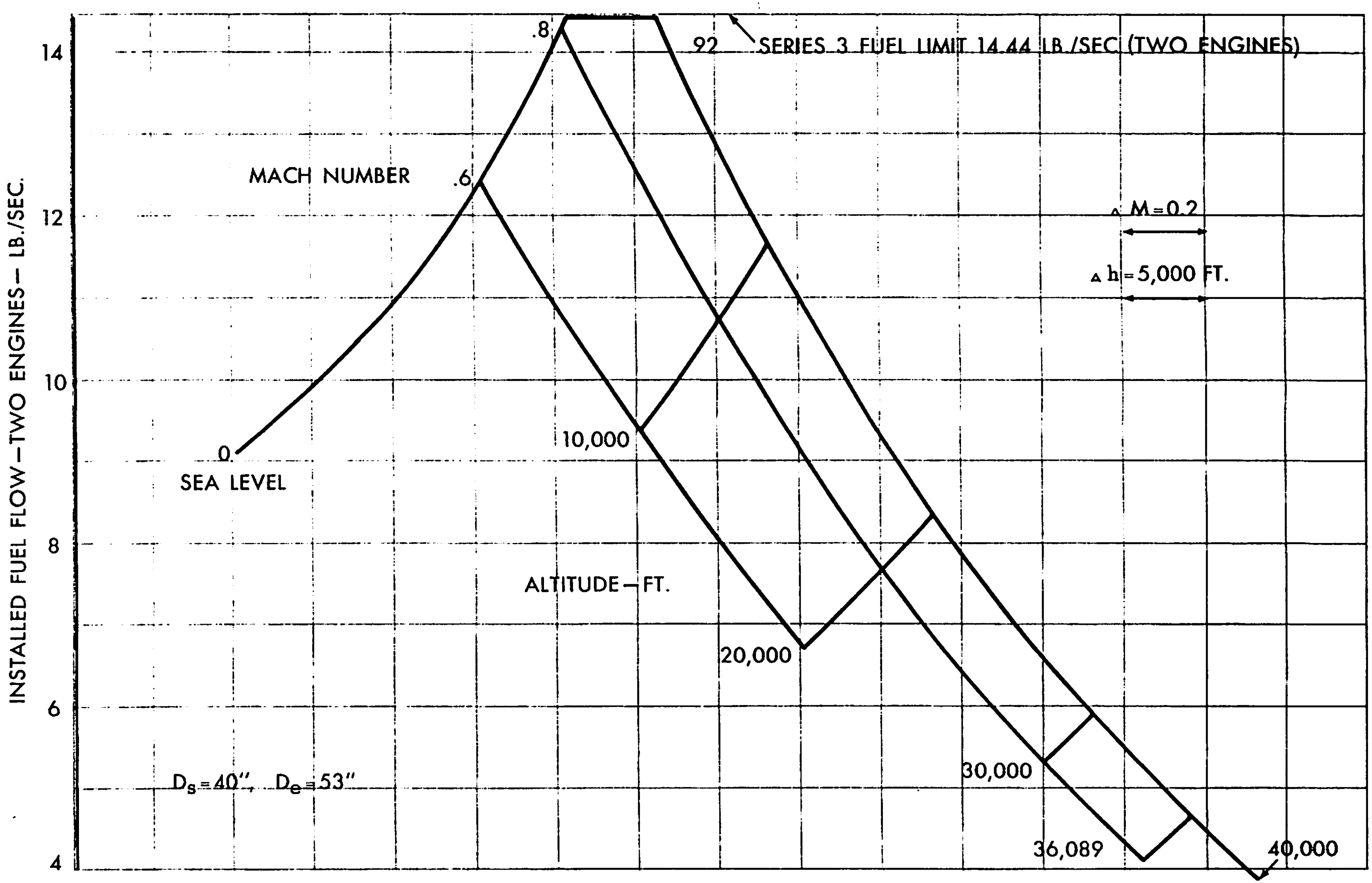

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FIG. 13 SERIES 3 IROQUOIS ENGINE—INSTALLED FUEL FLOW, MAXIMUM RPM, A/B UNIT

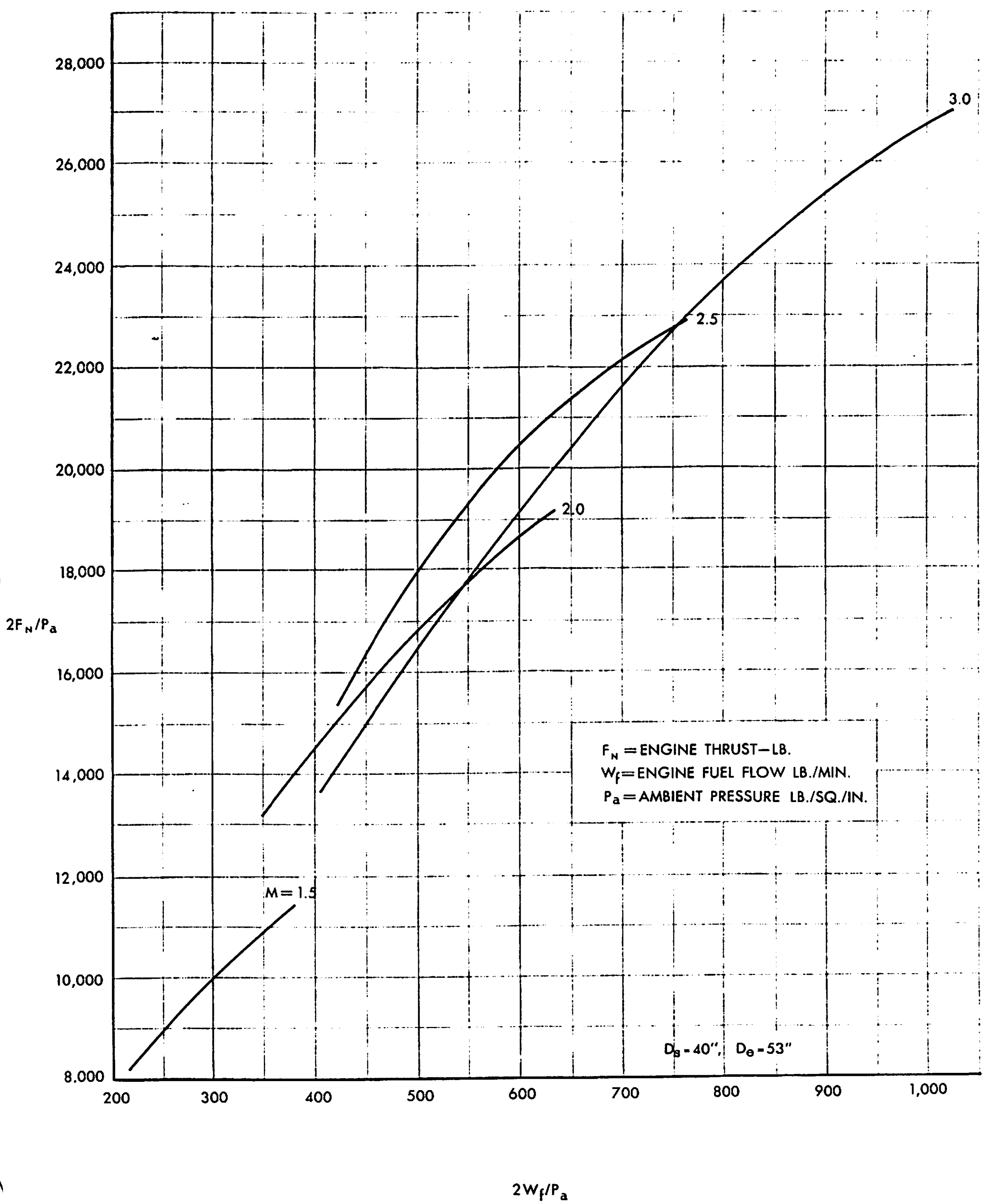

FIG. 14 SERIES 3 IROQUOIS ENGINE—PARTIAL THRUST, A/B LIT

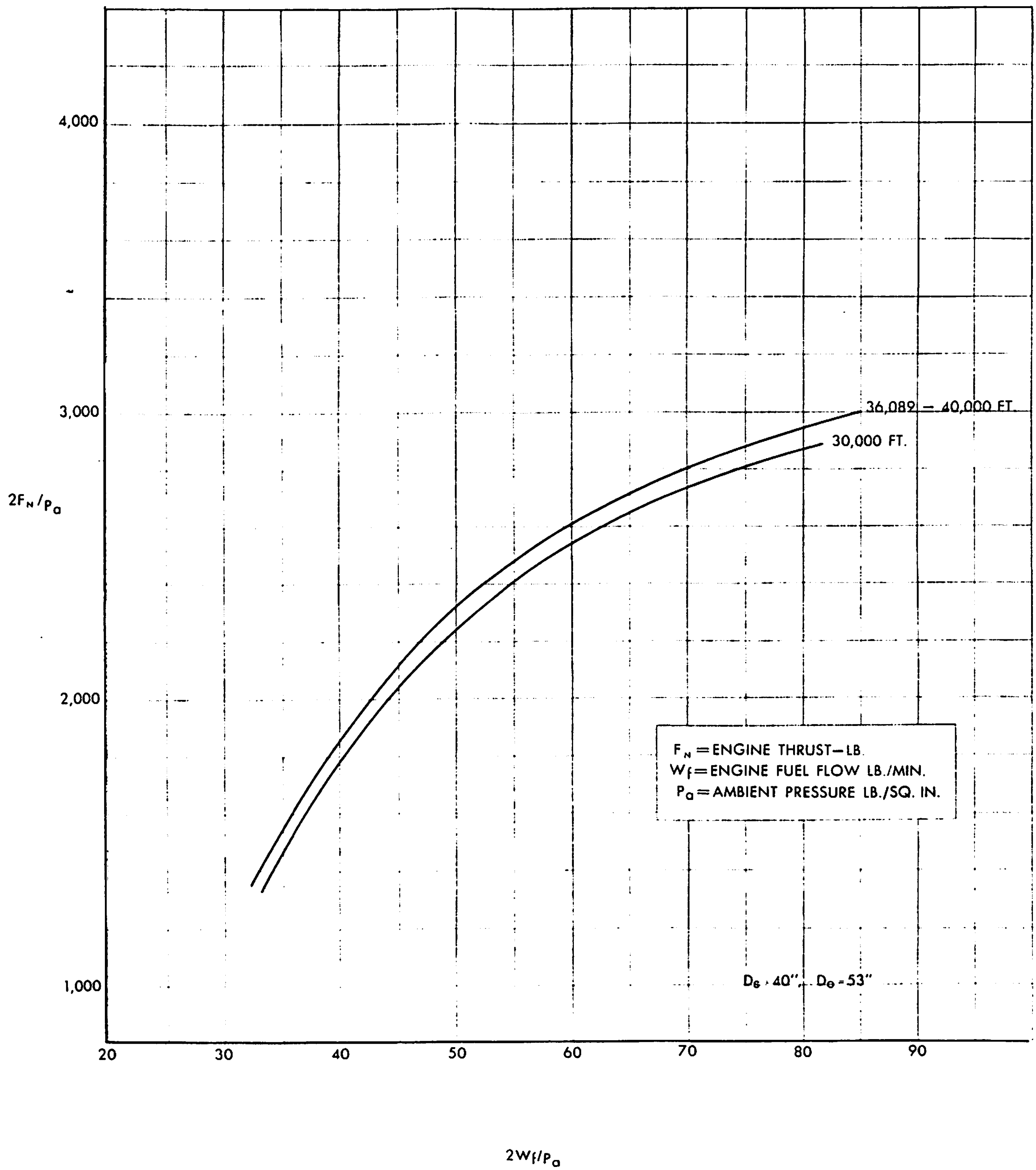

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FIG. 15 SERIES 3 IROQUOIS ENGINE—PARTIAL THRUST AT M = .92, A/B UNLIT, ALTITUDE 30,000 TO 40,000 FT.

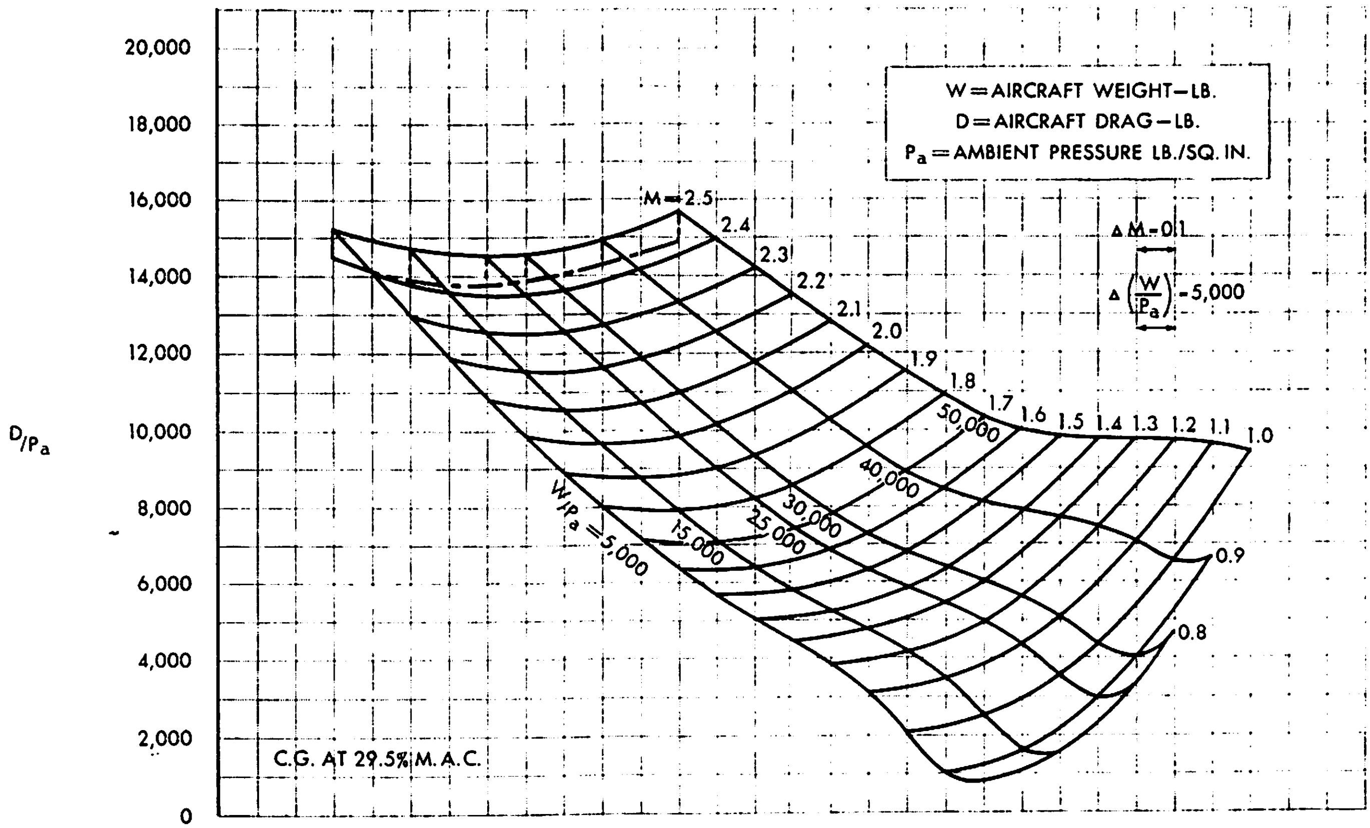

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FIG. 17 ARROW 3—D/Pa v W/Pa AND MACH NUMBER, NO AILERON DEFLECTION

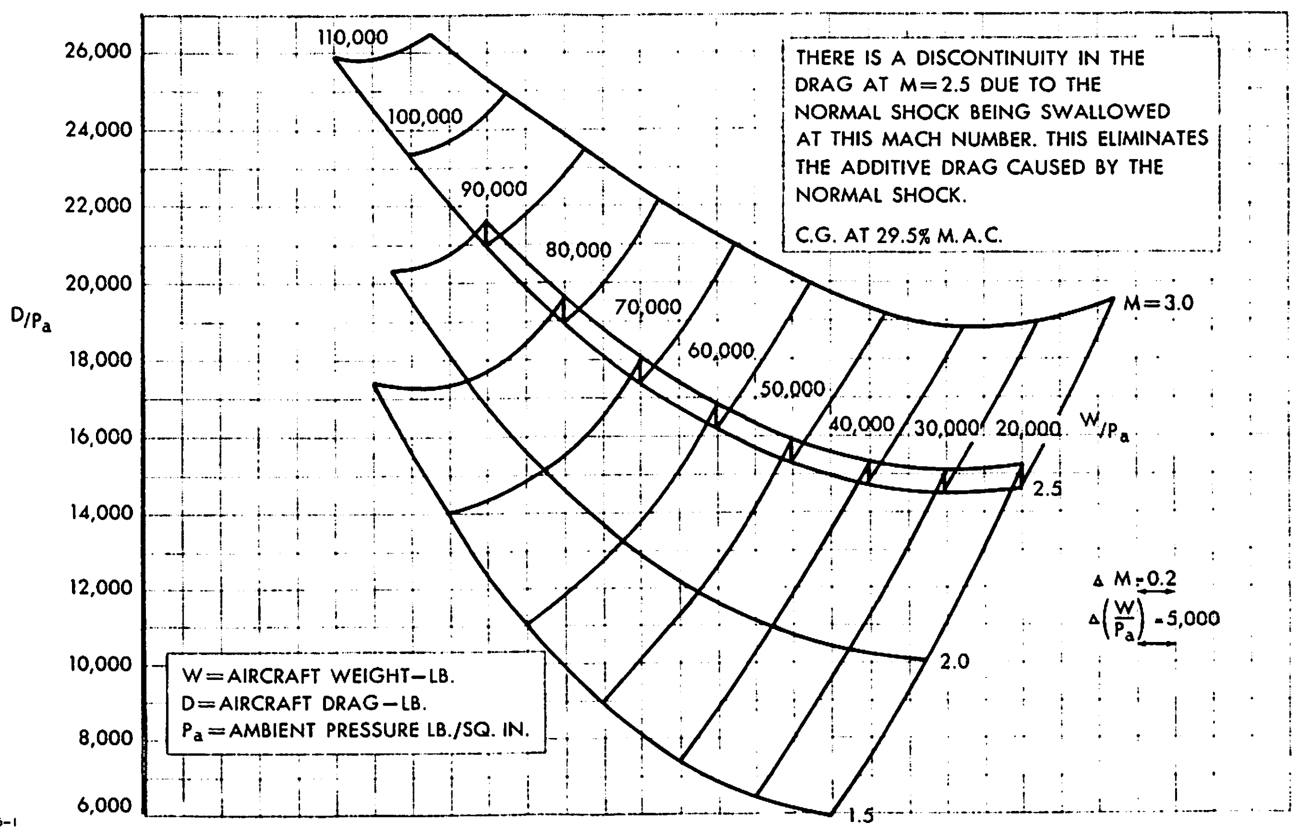

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FIG. 18 ARROW 3—D/Pa v W/Pa AND MACH NUMBER, 4° AILERON DEFLECTION ABOVE 45,000 FT.

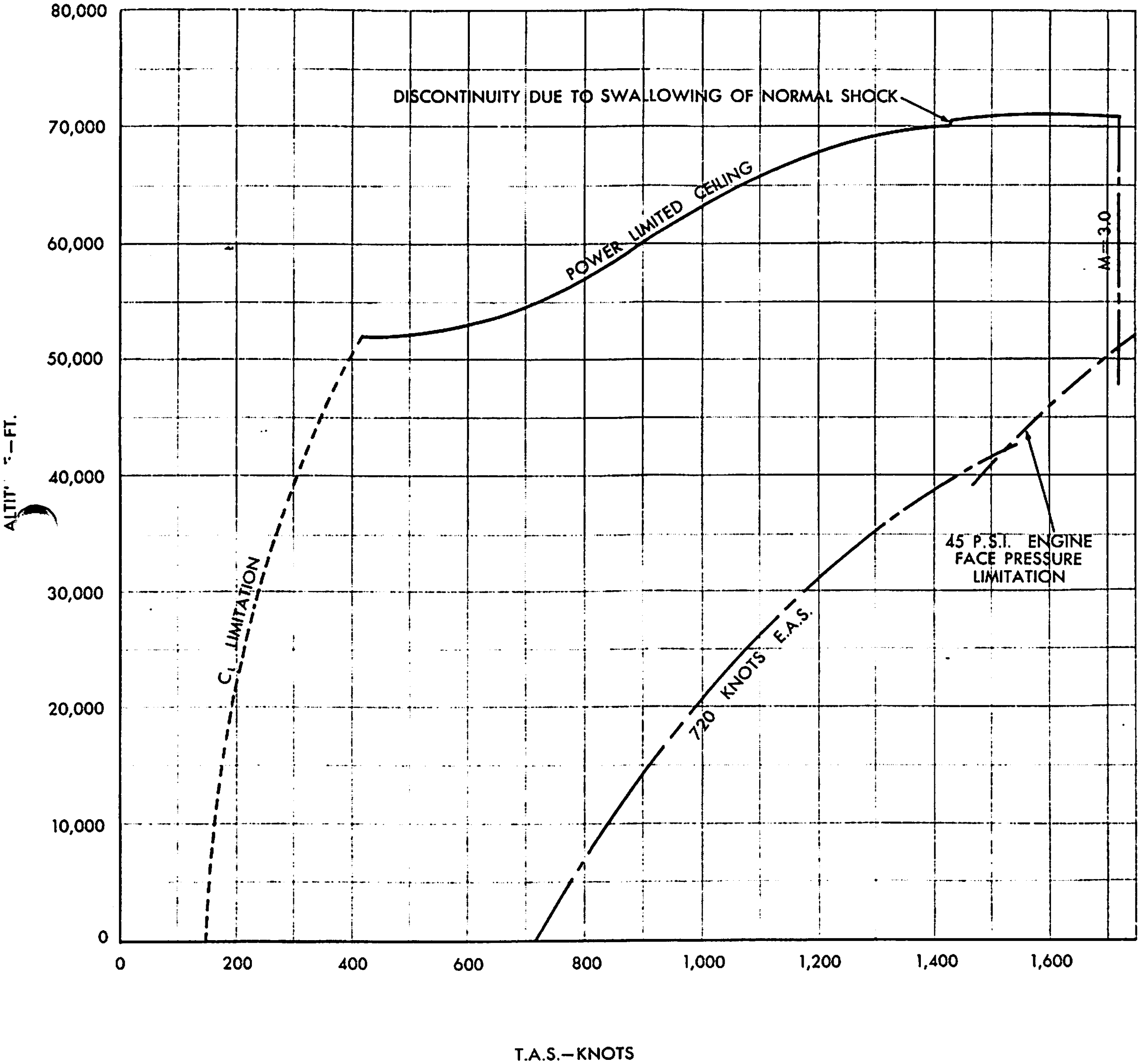

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FIG. 19 MAXIMUM LEVEL SPEED AT COMBAT WEIGHT

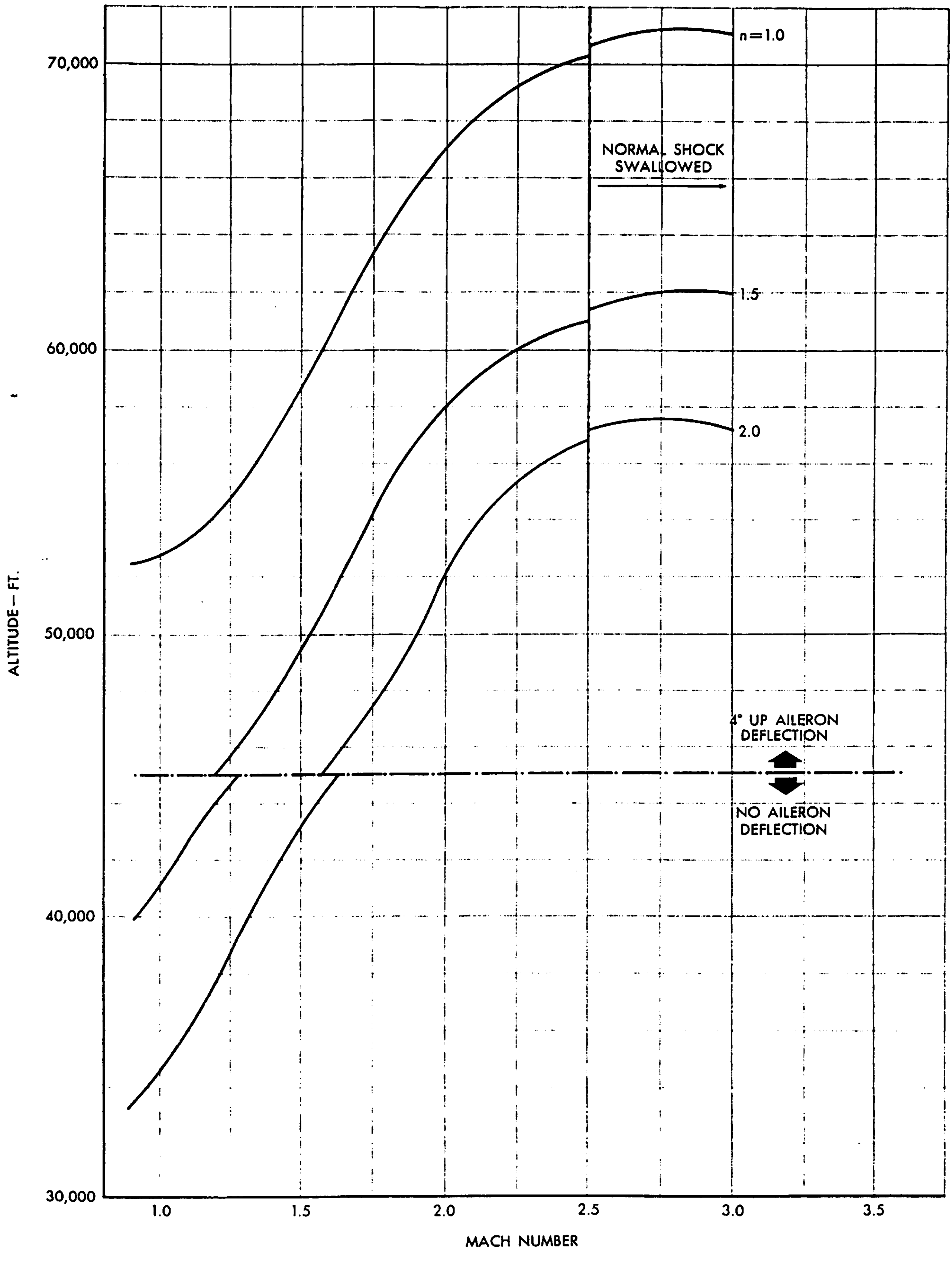

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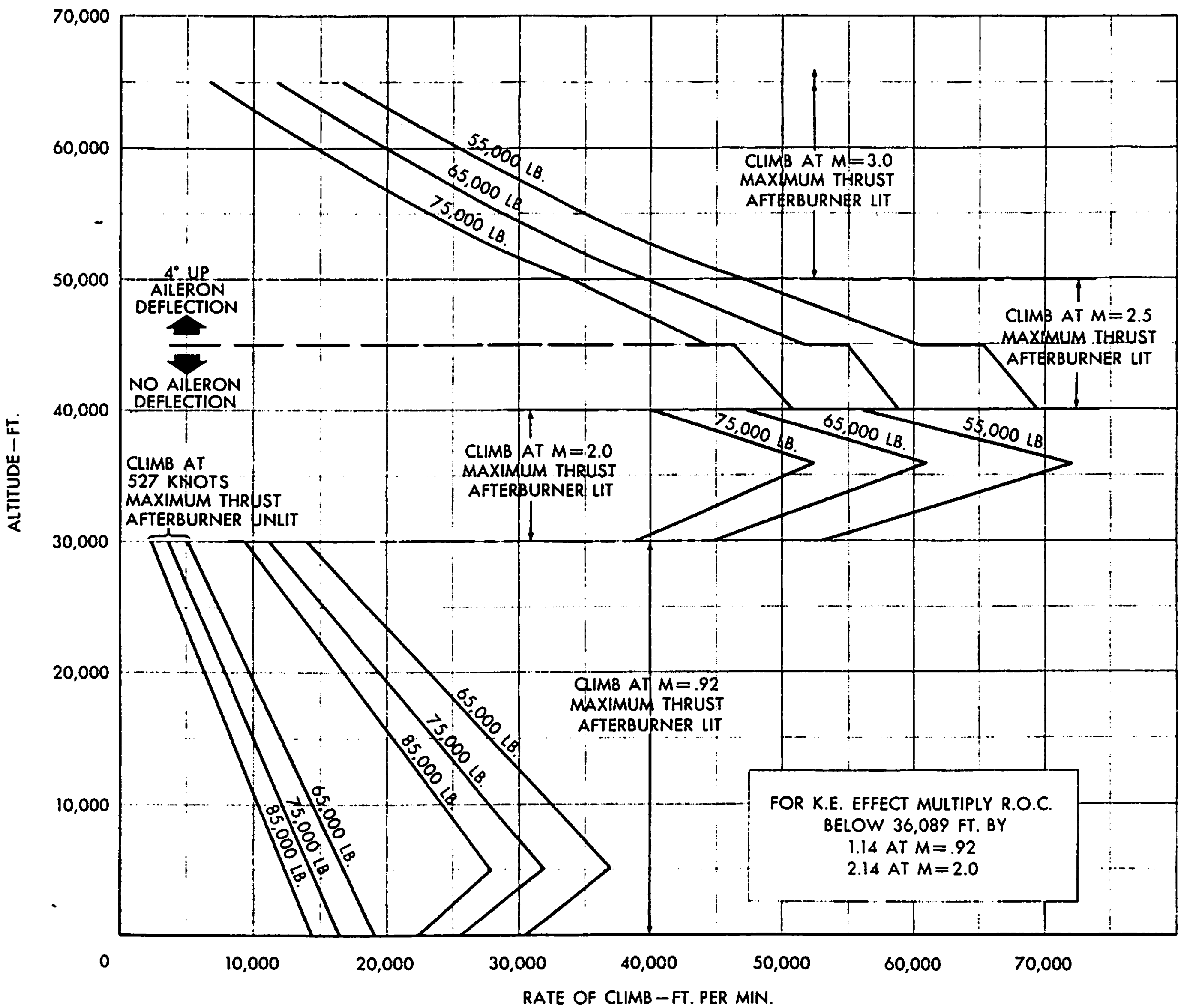
FIG. 20 ARROW 3 MANOEUVRABILITY-AVAILABLE STEADY G'S AT COMBAT WEIGHT, MAXIMUM THRUST, A/B LIT



3445-105-1

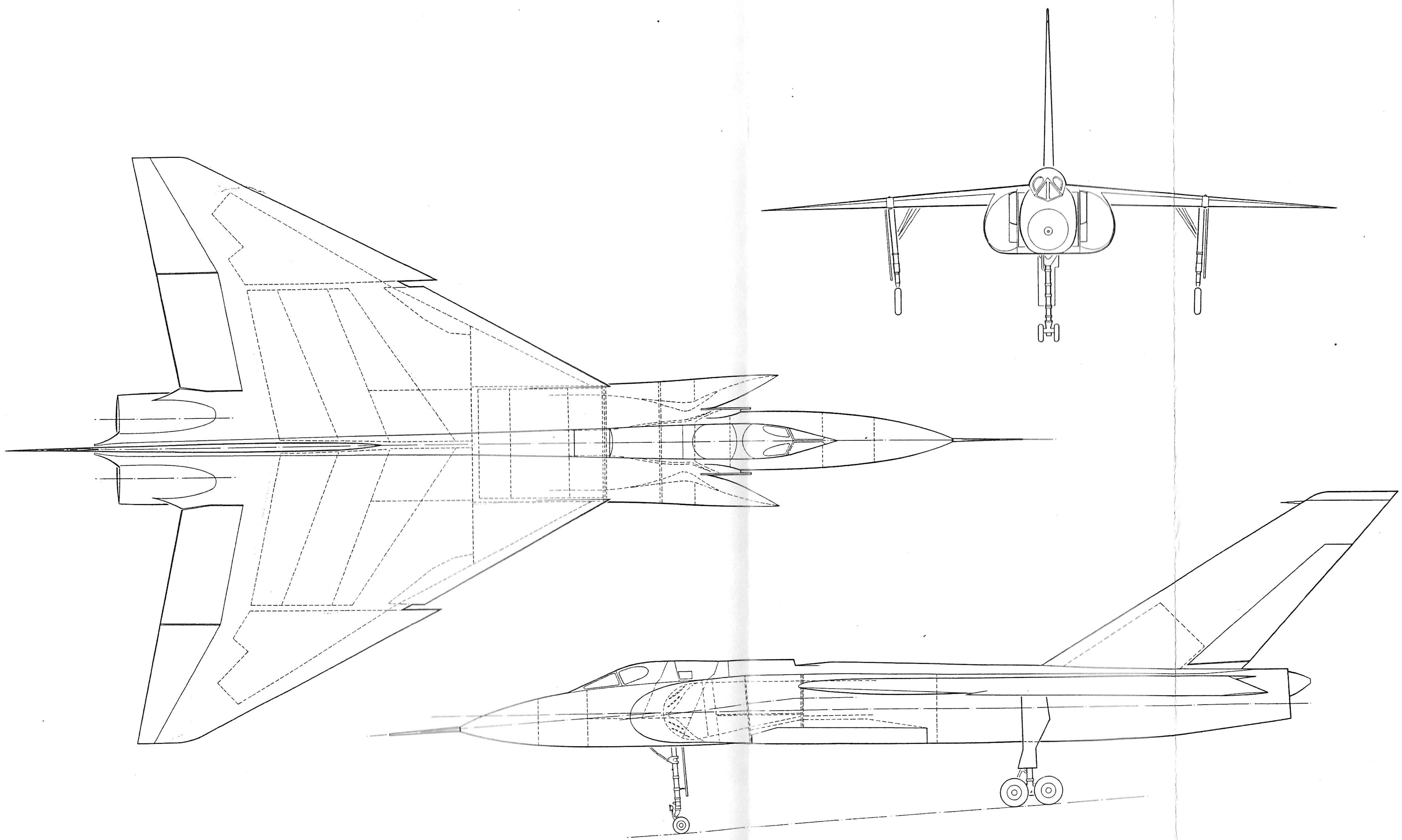
FIG. 22 STEADY RATE OF CLIMB

TABLE 4

ARROW 3 LOADING AND PERFORMANCE UNDER I.C.A.O. STANDARD ATMOSPHERIC CONDITIONS

WEIGHT	
Take-off weight with full internal fuel	82,392 lb.
Operational weight empty	53,654 lb.
Combat weight (1/2 full internal fuel)	68,023 lb.
Normal landing design weight (MIL-S-5701)	59,111 lb.
Wingloading at take-off weight	67.2 lb.
Power loading at take-off weight (A/B lit)	2.03 lb./lb. Thrust
SPEED	
True airspeed in level flight at sea level at combat weight, maximum thrust, A/B lit.	720 kts. T.A.S. *
True airspeed in level flight at 65,000 ft. at combat weight, maximum thrust, A/B lit	1,720 kts. T.A.S. *
CEILING	
Combat celing at combat weight, rate of climb 500 ft./min. with maximum thrust, A/B lit, at M=3.0	70,500 ft.
RATE OF CLIMB	
Steady state rate of climb at sea level at combat weight.	
(1) Maximum thrust, A/B unlit, at 527 kts.	18,400 ft./min.
(2) Maximum thrust, A/B lit, at M=.92	29,400 ft./min.
Steady state rate of climb at 65,000 ft. at combat weight, maximum thrust, A/B lit, at M=3.0	10,300 ft./min.
TIME TO HEIGHT	
Time to reach 65,000 ft. and M=3.0 from engine start, maximum thrust, A/B lit	8.00 min.
MANOEUVRABILITY	
Combat load factor at combat weight, maximum thrust, A/B lit at M=3.0 at 65,000 ft.	1.37

* Placarded speeds



3426-105-1

FIG. 1 ARROW 3—THREE VIEW G.A.

TABLE 8

ARROW 3 MAXIMUM RADIUS MISSION WITH R.C.A.F. RESERVES

Condition	Distance N.M.	Time Min.	Fuel Lb.	A/C Wt. Lb.
Start weight				82,392
Engine start		.5	100	82,292
Take-off to unstick, max. thrust, A/B unlit	-	.6	340	81,952
Acc. to 527 Kts. at S.L., max. thrust, A/B unlit	6.5	1.1	810	81,142
Climb at 527 Kts. to 30,000 ft., max. thrust, A/B-unlit	34.3	4.0	2,060	79,082
Cruise out at M=.92 at 30,000 ft.	296.0	32.8	6,300	72,782
Acc. to M=2.0 at 30,000 ft., max. thrust, A/B lit	26.8	1.98	2,860	69,922
Climb at M=2.0 to 40,000 ft., max. thrust, A/B lit	1.9	.14	270	69,652
Acc. to M=2.5 at 40,000 ft., max. thrust, A/B lit	12.5	.60	1,160	68,492
Climb at M=2.5 to 50,000 ft., max. thrust, A/B lit	4.6	.21	355	68,137
Acc. to M=3.0 at 50,000 ft., max. thrust, A/B lit	27.0	1.02	1,550	66,587
Climb at M=3.0 to 65,000 ft., max. thrust, A/B lit	20.0	.7	808	65,779
Combat at M=3.0 at 65,000 ft., max. thrust, A/B lit		5.0	4,200	59,851
Descend to optimum cruise altitude of 35,000 ft. at idle thrust	108.0	7.1	130	59,721
Cruise back at M=.92 at optimum cruise altitude of 38,000 ft.	321.6	36.6	4,835	54,886
Loiter over base at 38,000 ft.		15.0	1,770	53,116
Descend to S.L. at idle thrust		6.6	340	52,776
Land with reserves for 5 min. loiter at S.L.		5.0	850	51,926
TOTAL	859.2	118.95	28,738	

NOTE: (a) 1,728 lb. missile fired at the end of combat.
(b) Allowance is made for deceleration during descent after combat.
(c) Combat radius 430.N.M.
(d) If A/B is required for take-off the loss in combat radius is approximately 8 N.M.

TABLE 5

ARROW 3 LOITER DASH MISSION WITH REFUELLING, MIL-C-5011a RESERVES

Conditions	Distance N.M.	Time Min.	Fuel Lb.	A/C Wt. Lb.
Start weight				82,392
Fuel allowances for starting engines, take-off, and acc. to climb speed - 2 min. at normal power		2.0	1,130	81,262
Climb on Course at 527 Kts to 30,000 ft., max. thrust, A/B unlit	25.5	3.0	1,600	79,662
Cruise at M=.90	564.9	64.6	8,940	70,722
Loiter at max. endurance speed	-	123.0	15,068	55,654
Flight refuelling				82,392
Acc. to M=2.0 at 30,000 ft., max. thrust, A/B lit	31.0	2.30	3,480	78,912
Climb at M=2.0 to 40,000 ft., max. thrust, A/B lit	2.6	.17	336	78,576
Acc. to M=2.5 at 40,000 ft., max. thrust, A/B lit	142.0	.69	1,385	77,191
Climb at M=2.5 to 50,000 ft., max. thrust, A/B lit	5.4	.24	425	76,766
Acc. to M=3.0 at 50,000 ft., max. thrust, A/B lit	31.4	1.17	1,850	74,916
Climb at M=3.0 to 65,000 ft., max. thrust, A/B lit	26.0	.92	1,100	73,816
Combat at M=3.0 at 65,000 ft.	-	5.0	4,400	69,416
Descend to optimum cruise altitude of 38,000 ft. at idle thrust	111.0	7.25	141	69,275
Cruise back at 38,000 ft. at M=.92	590.0	67.2	10,521	58,754
Fuel allowances for reserves and landing - 5% of initial fuel plus 20 min. max. endurance at S.L.		20.0	5,100	53,654
TOTAL	1,402	297.54	57,476	

NOTE: (a) In order to improve the subsonic cruise performance jettisonable nozzle inserts have been fitted. With these inserts in position it is not possible to use the after burners.

(b) Missiles held throughout flight.

(c) Allowance is made for deceleration during descend after combat.

(d) Combat radius 701 N.M.

(e) 75 min. loiter instead of 123 if it is necessary to take off with afterburners lit (i.e. no nozzle inserts).

TABLE 7

ARROW 3 DASH MISSION WITH MIL-C-5011a RESERVES

Condition	Distance N.M.	Time Min.	Fuel Lb.	A/C Wt. Lb.
Start weight				82,392
Fuel allowance for starting engines, take-off, and acc. to climb speed - 2 min. at normal power plus 1 min. at max. power, A/B lit		3.00	2,660	79,732
Climb on course at M=.92 to 30,000 ft. max. thrust, A/B lit	11.4	1.30	2,020	77,712
Acc. to M=2.0 at 30,000 ft., max. thrust, A/B lit	28.8	2.14	3,260	74,452
Climb at M=2.0 to 40,000 ft., max. thrust, A/B lit	2.1	.15	305	74,147
Acc. to M=2.5 at 40,000 ft., max. thrust, A/B lit	13.4	.64	1,300	72,847
Climb at M=2.5 to 50,000 ft., max. thrust, A/B lit	5.0	.22	400	72,447
Acc. to M=3.0 at 50,000 ft., max. thrust, A/B lit	28.8	1.09	1,740	70,707
Climb at M=3.0 to 65,000 ft., max. thrust, A/B lit	22.4	.78	940	69,767
Cruise out at M=3.0 at 65,000 ft.	163.0	5.73	3,660	66,107
Combat at M=3.0 at 65,000 ft., max. thrust, A/B lit		5.00	4,400	61,707
Descend to optimum cruise altitude of 38,000 ft. at idle thrust	108.0	7.10	136	61,571
Cruise back at M=.92 at optimum cruise altitude of 38,000 ft.	166.9	19.00	2,817	58,754
Fuel allowance for reserves and landing - 5% of initial fuel plus 20 min. max. endurance at S.L.		20.00	5,100	53,654
TOTAL	549.8	66.15	28,738	

NOTE: (a) Missiles held throughout flight.

(b) Allowance is made for deceleration during descent after combat.

(c) Combat radius 275 N.M.

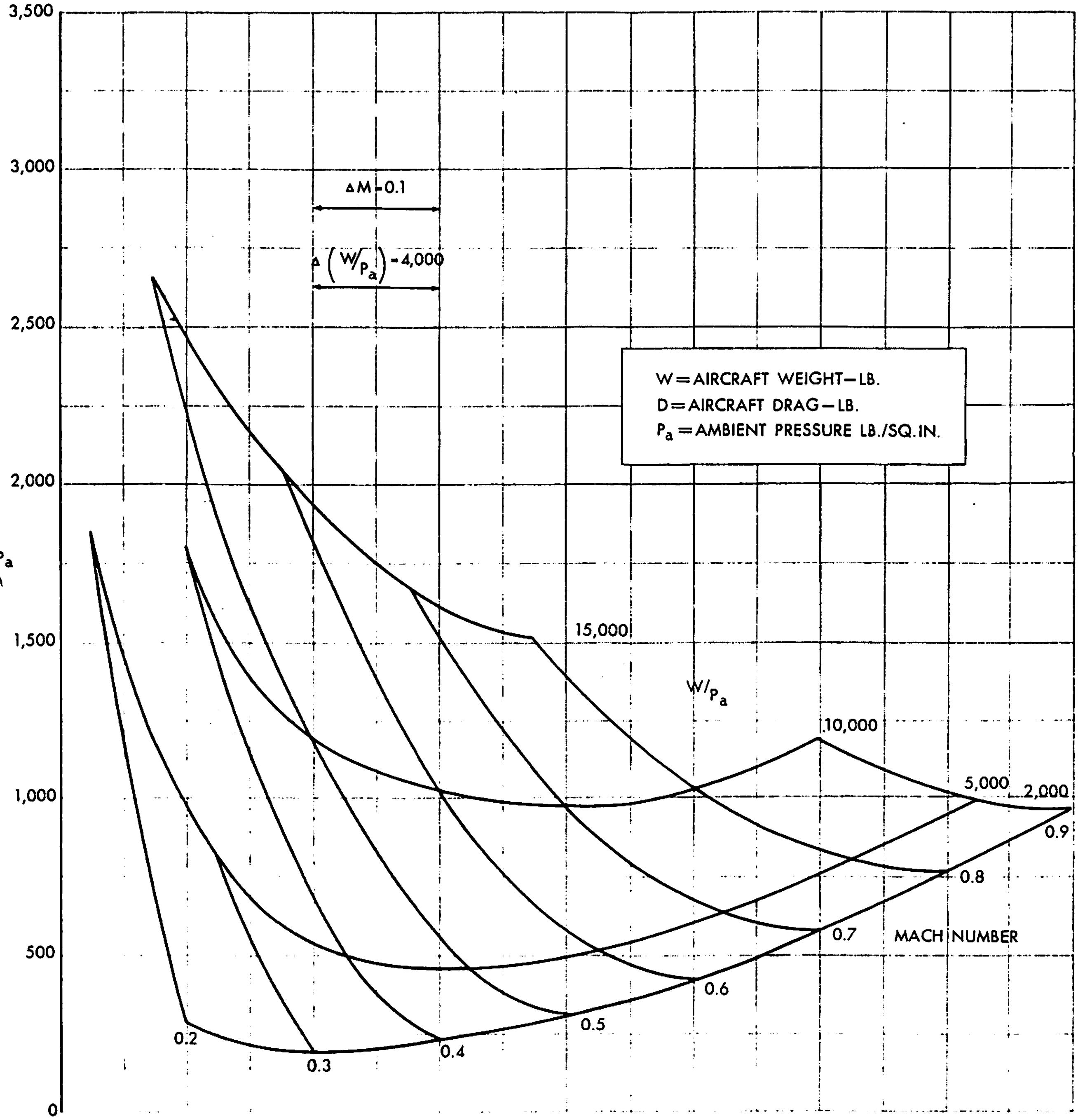


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FIG. 16 ARROW 3—D/Pa v W/Pa AND MACH NUMBER, LOW MACH NUMBERS

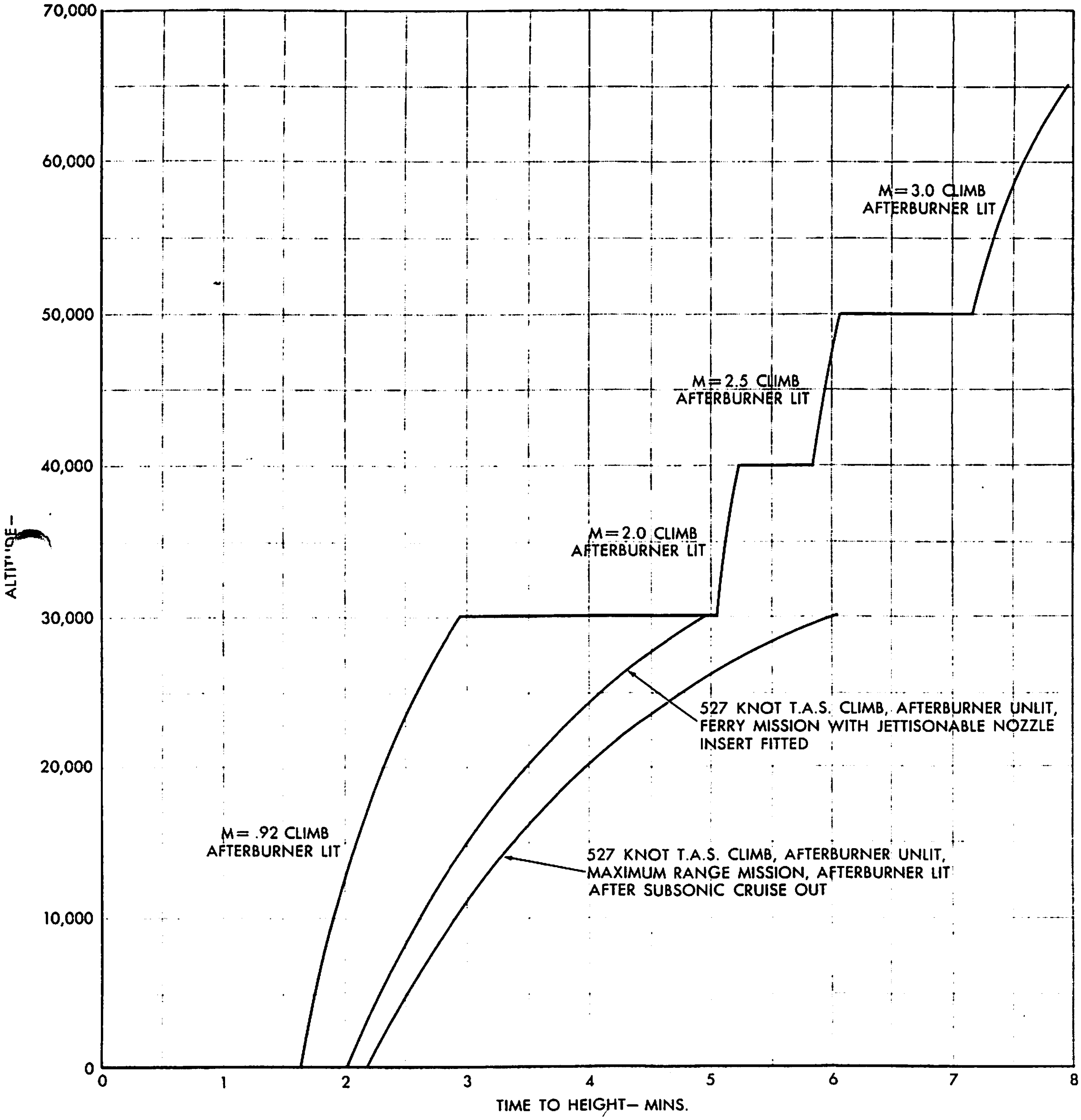

FIG. 21 TIME TO HEIGHT FROM ENGINE START

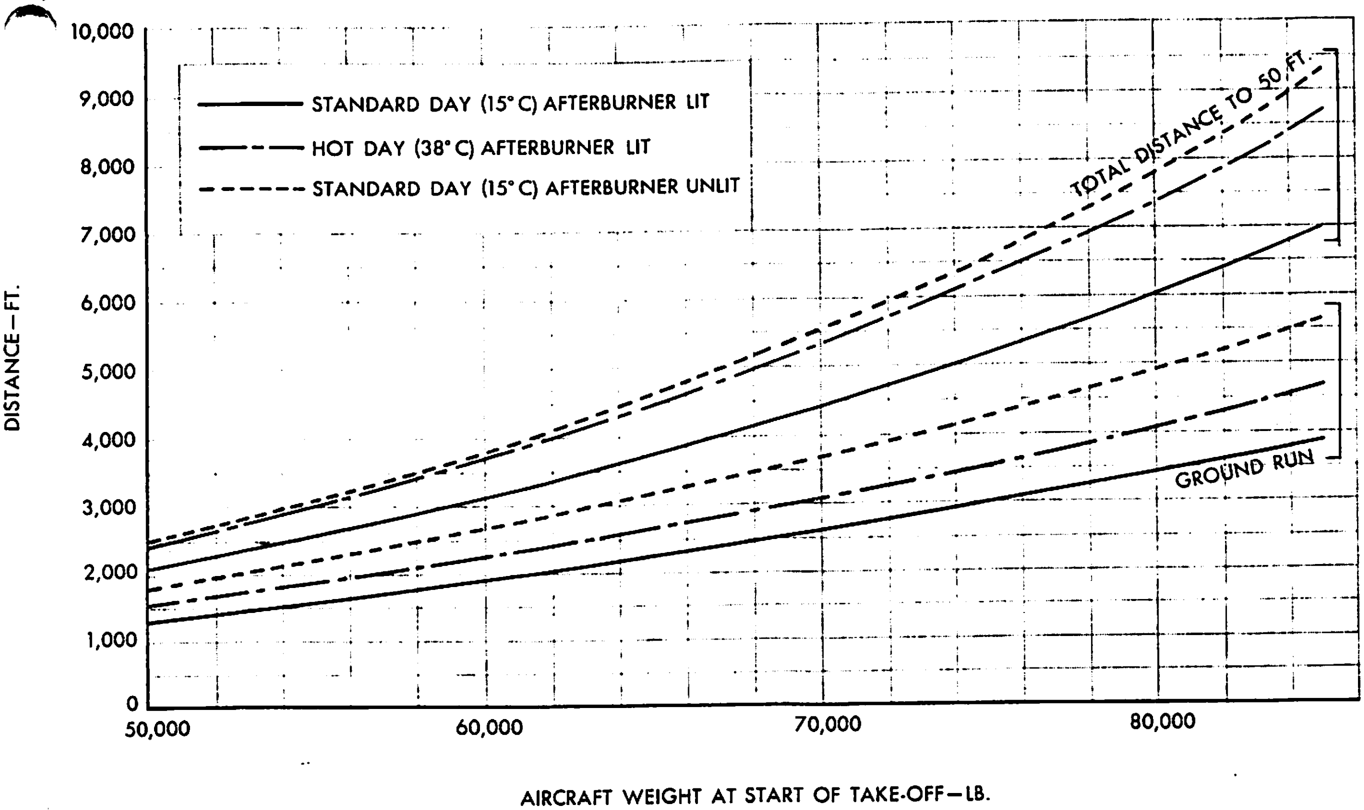

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FIG. 23 TAKE-OFF DISTANCE AT S.L.

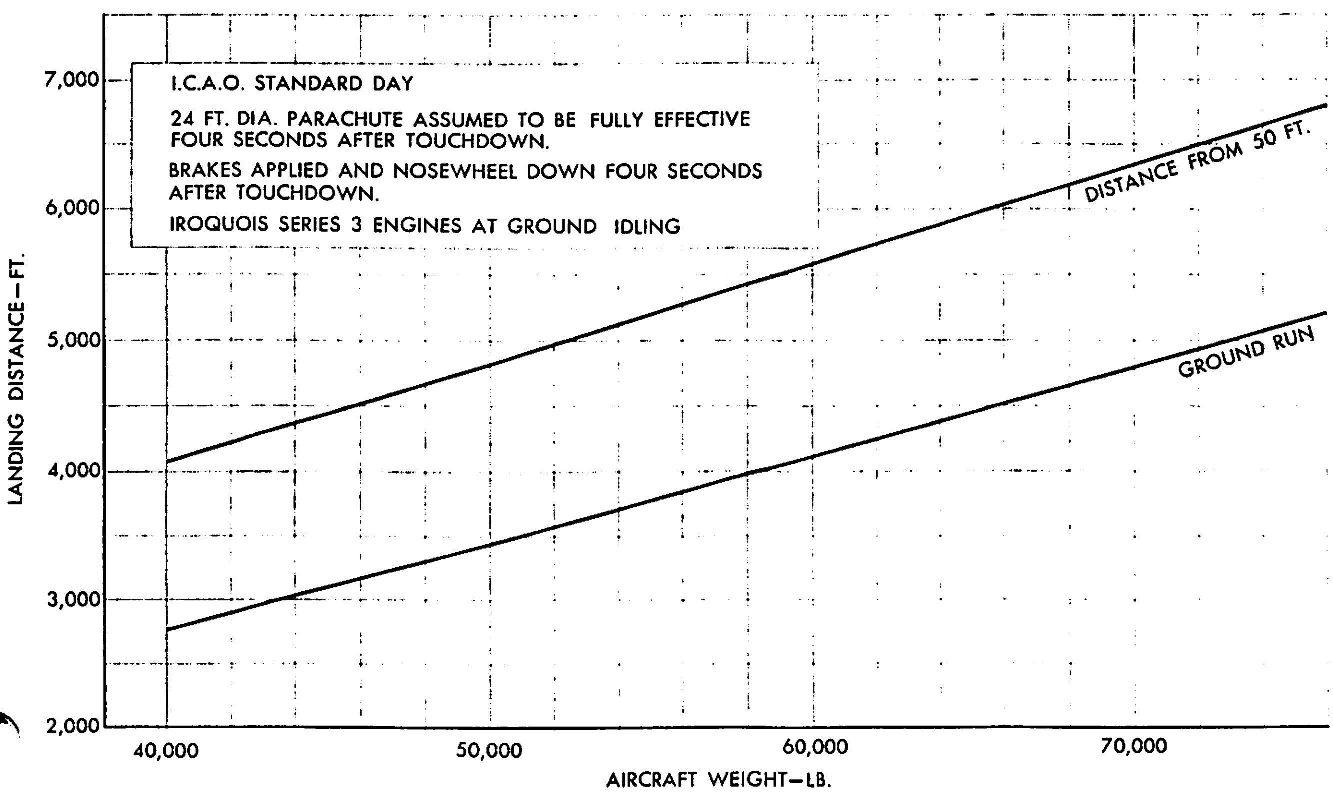

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FIG. 24 LANDING DISTANCE AT S.L.

TABLE 4 (Cont'd)

TAKE-OFF DISTANCE	
Take-off distance over a 50 ft. obstacle at S. L. at take-off weight	
(1) Maximum thrust, A/B lit, Standard Day	6,500 ft.
(2) Maximum thrust, A/B unlit, Standard Day	8,500 ft.
LANDING DISTANCE	
Landing Distance over a 50 ft. obstacle at S. L. at normal design landing weight	5,650 ft.
MISSIONS	
Mission details are given in Tables 5 to 10 inclusive.	
Combat radii of action:-	
›iter Dash Mission with refuelling, MIL-C-5011a reserves	701 N. M.
gh Speed Mission with R. C. A. F. reserves	389 N. M.
Dash Mission with MIL-C-5011a reserves	275 N. M.
Maximum Radius Mission with R. C. A. F. reserves	430 N. M.
Maximum Radius Mission with MIL-C-5011a reserves	332 N. M.
Range:-	
Ferry Mission using full internal fuel only, R. C. A. F. reserves	1,707 N. M.

TABLE 6

ARROW 3 HIGH SPEED MISSION WITH R.C.A.F. RESERVES

Condition	Distance N.M.	Time Min.	Fuel Lb.	A/C Wt. Lb.
Start Weight	-	-	-	82,392
Engine Start	-	.50	100	82,292
Take-off to Unstick, max. thrust, A/B lit	-	.39	620	81,672
Acc. to M=.92 at S.L., max. thrust, A/B lit	4.9	.76	1,530	80,142
Climb at M=.92 to 30,000 ft., max. thrust, A/B lit	11.5	1.32	1,930	78,212
Acc. to M=2.0 at 30,000 ft., max. thrust, A/B lit	29.2	2.16	3,120	75,092
Climb at M=2.0 to 40,000 ft., max. thrust, A/B lit	2.2	.16	294	74,798
cc. to M=2.5 at 40,000 ft., max. thrust, A/B lit	13.5	.65	1,255	73,543
imb at M=2.5 to 50,000 ft., max. thrust, A/B lit	5.0	.22	380	73,163
Acc. to M=3.0 at 50,000 ft., max. thrust, A/B lit	29.1	1.10	1,675	71,488
Climb at M=3.0 to 65,000 ft., max. thrust, A/B lit	23.0	.80	915	70,573
Cruise at M=3.0 at 65,000 ft., partial A/B.	271.0	9.50	5,700	64,873
Combat at M=3.0 at 65,000 ft., max. thrust, A/B lit		5.00	4,200	58,945
Descend to optimum cruise altitude of 38,000 ft. at idle thrust	108.0	7.10	130	58,815
Cruise back at M=.92 at optimum cruise altitude of 38,000 ft.	281.4	32.10	3,929	54,886
Loiter over base at 38,000 ft.		15.00	1,770	53,116
Descend to S.L. at idle thrust		6.60	340	52,776
Land with reserves for 5 min. loiter at S.L. at max. endurance speed		5.00	850	51,926
TOTAL	778.8	88.36	28,738	

OTE: (a) 1,728 lb. missiles fired at the end of combat.

(b) Allowance is made for deceleration during descent after combat.

(c) Combat radius is 389 N.M.

TABLE 9

ARROW 3 MAXIMUM RADIUS MISSION WITH MIL-C-5011a RESERVES

Condition	Distance N.M.	Time Min.	Fuel Lb.	A/C Wt. Lb.
Start weight				82,392
Fuel allowance for starting engines, take-off and acc. to climb speed - 2 min. at normal power		2.0	1,130	81,262
Climb on course at 527 Kts to 30,000 ft., max. thrust, A/B unlit	34.7	3.96	2,160	79,102
Cruise at M=.92 at 30,000 ft.	202.0	22.40	4,600	74,502
Acc. to M=2.0 at 30,000 ft., max. thrust, A/B lit	27.5	2.04	2,980	71,522
Climb at M=2.0 to 40,000 ft., max. thrust, A/B lit	1.9	.14	290	71,232
Acc. to M=2.5 at 40,000 ft., max. thrust, A/B lit	12.8	.62	1,250	69,982
Climb at M=2.5 to 50,000 ft., max. thrust, A/B lit	4.8	.21	378	69,603
Acc. to M=3.0 at 50,000 ft., max. thrust, A/B lit	27.5	1.05	1,660	67,944
Climb at M=3.0 to 65,000 ft., max. thrust, A/B lit	20.7	.73	880	67,064
Combat at M=3.0 at 65,000 ft., max. thrust, A/B lit		5.0	4,400	62,664
Descend to optimum cruise altitude of 38,000 ft. at idle thrust	108.0	7.10	136	62,528
Cruise back at M=.92 at optimum cruise altitude of 38,000 ft.	223.9	25.40	3,774	58,754
Allowance for reserves and land - 5% of initial fuel plus 20 min. max. endurance at S.L.		20.00	5,100	53,654
TOTAL	663.8	91.01	28,738	

NOTE: (a) Missiles held throughout flight.

(b) Allowance is made for deceleration during descent after combat.

(c) Combat radius 332 N.M.

TABLE 10

ARROW 3 FERRY MISSION USING FULL INTERNAL FUEL ONLY

Condition	Distance N.M.	Time Min.	Fuel Lb.	A/C Wt. Lb.
Start Weight				82,392
Engine Start		.5	100	82,292
Take-off to unstick max. thrust, A/B unlit		.6	340	81,952
Accelerate to 527 Kts., max. thrust, A/B unlit	6.5	1.1	810	81,142
Climb to 30,000 ft. at 527 Kts., max. thrust, A/B unlit	25.5	3.0	1,525	79,617
Cruise climb to 36,000 ft. at M=.9	1,675	192.5	23,243	56,374
Loiter over base at 36,000 ft. to 38,000 ft.	-	15.0	1,500	54,874
Descend to S. L. at idle thrust	-	6.6	340	54,534
Land with reserves for 5 min. loiter at S. L.	-	5.0	880	53,654
TOTAL	1,707	224.3	28,738	

NOTE: (a) Missiles carried throughout flight.

(b) In order to improve the subsonic cruise performance, jettisonable nozzle inserts have been fitted. With these inserts in position it is not possible to use the afterburners.

(c) If the afterburners are required for take-off the ferry range is 1,445 N.M. with a jettisonable external tank.

5. REFERENCE

1 Iroquois Series 3 Programme Proposal - Orenda Engines Ltd. brochure B9-57-1.