FILE IN VAULT

C-105 STRUCTURE

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C-105 STRUCTURE

THIS REPORT COVERS IN A BROAD MANNER THE MAJOR STRUCTURAL ASPECTS OF THE C-105 AIRCRAFT UNDER THE FOLLOWING HEADINGS:-

1. LOAD ANALYSIS
2. SHAPES
3. INTERNAL PRESSURES
4. LANDING GEAR
5. HEAT
6. FATIGUE
7. NEW MATERIALS

INTRODUCTION

The aerodynamic requirements as applied to the C-105 has resulted in a very difficult structure problem.

To meet these very rigid geometric requirements and produce a light weight structure has required an almost complete change in the concept of stressing and structural arrangement. There is no place for conservative assumptions to short cut analysis. In fact, in most cases it is impossible to make conservative assumptions, for in many cases extra material can be as detrimental to the structure as insufficient material. The work is characterised by rational analysis with close control on distorted shapes.

Load Analysis

The problem of analysing complex low aspect ratio wings and fuselage combination must be solved for on a rational basis. The fuselage or centre section can no longer be assumed to be a rigid support but bends under load, thereby influencing the load distribution to a large extent. Fig. 1 illustrates this point.

It is therefore evident that the fuselage wing combination must be analysed as a complete complex redundant structure. The multi-spar, stressed skin arrangement with a large cut-out for the landing gear was realised to be highly redundant and an analysis was developed, built around the use of high speed digital computors.

It was decided to calculate directly for stresses and to compute displacements from known stresses. The wing was considered to act as a plate (a two dimensional problem) rather than as a beam (a one-dimensional problem) as is customary for straight or swept wings with relatively small taper.

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AIRCRAFT BODY DISTORTION

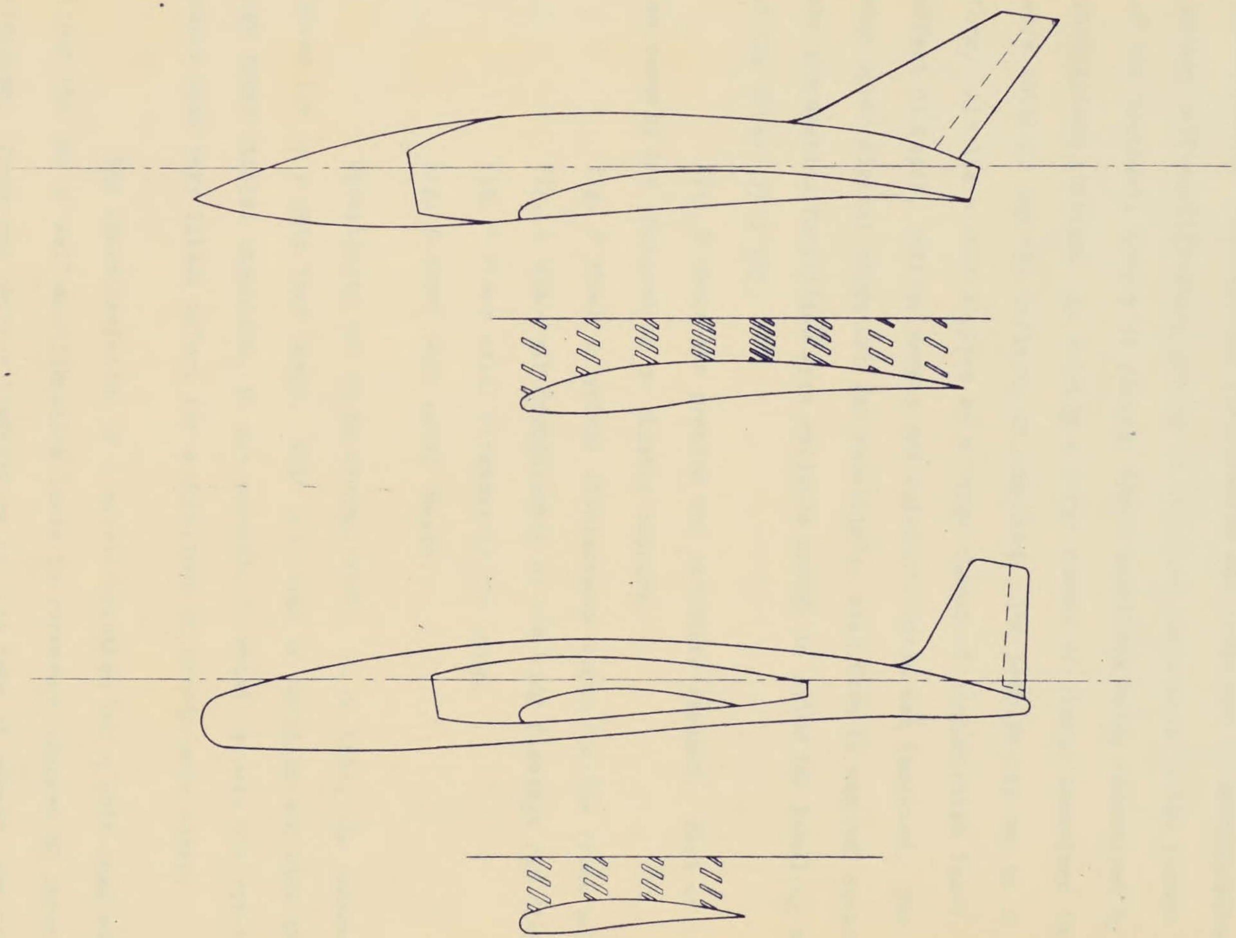

FIG 1

In brief, the method of analysis assumes an approximate moment distribution so as to satisfy the equilibrium conditions exactly but leaving strains still incompatible. This assumed distribution was corrected by several properly chosen self-equilibrating groups of internal loads akin to the stress functions of the Southwell theory of plates, their magnitude being determined by the Castigliano theorem. By solving a large number of linear equations the stresses were obtained and the deflections computed. The problem was set up in a matrix form. Air loads were applied as a large number of concentrated loads. Due effect of taper, oblique panels and Poisson's ratio was included. The use of high speed digital computors was essential; even then it was not apparent that the computation facilities were reliable enough to allow the handling of matrices of the order 173 X 173.

Fig. 2 shows the problem and structure assumed. Part of the structure was removed and replaced by an elastic support.

Fig. 3 shows a typical displacement curve for the front and rear spar.

Fig. 4 shows rib displacement at the wing fuselage rib displacement.

Fig. 5 shows axial stresses in the wing.

Fig. 6 shows skin panel shears.

Versatility had to be accomplished. To do this, the unknowns were solved for on a unit load basis. With this type of solution and with the use of high speed digital computors, it was possible to obtain stress distributions and shapes with very little effort for a multitude of aerodynamic cases.

The characteristics of a matrix solution and a unit load solution allows the use of self-equilibrating loads to represent changes of structure stiffness. There are obvious limitations to this type of repair, and complete versatility in relation to design changes is not possible.

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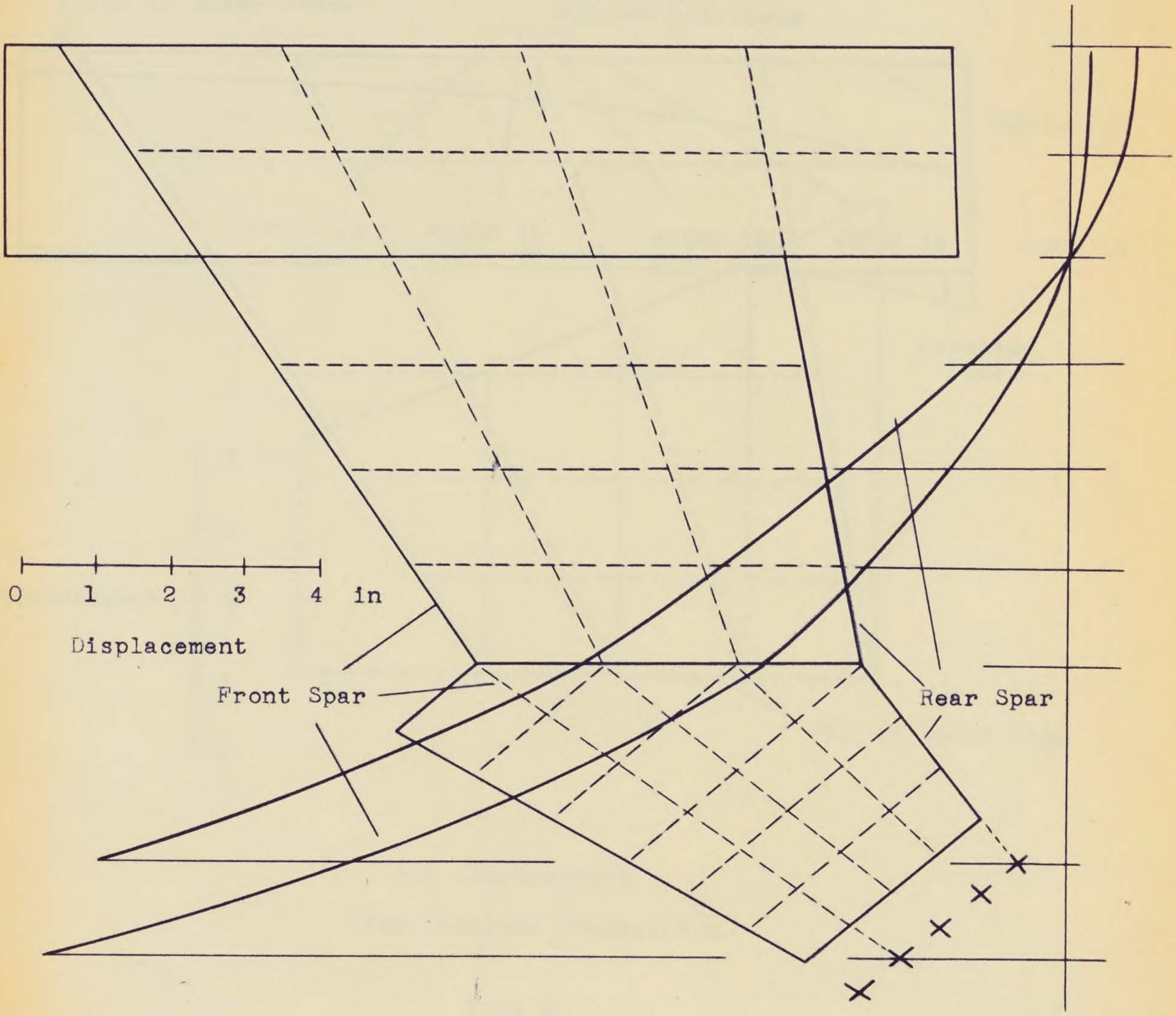
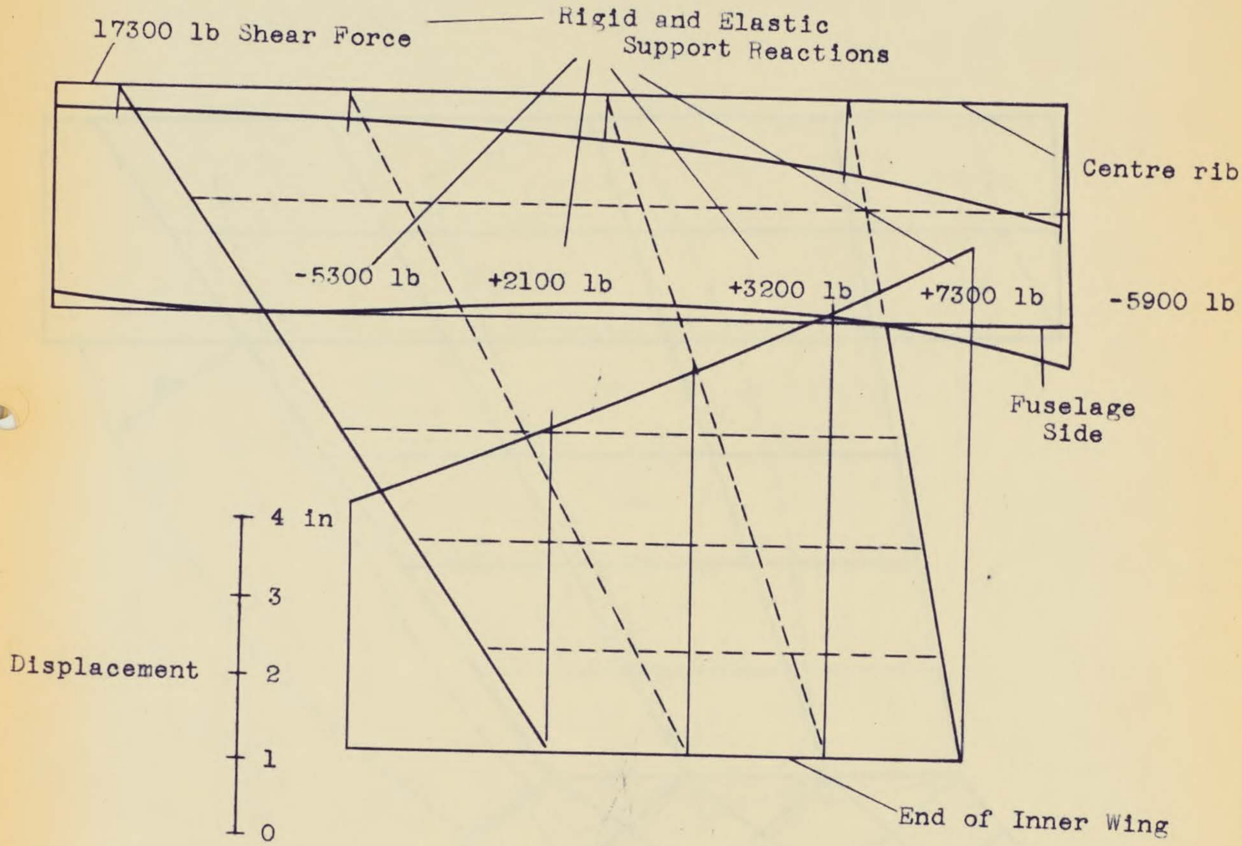


Fig. 3

Front and Rear Spar Displacement



Rib Displacement

Wing Fuselage Interaction

Fig. 47

ULTIMATE SPAR AXIAL STRESSES

DENOTED BY σ_s

SCALE

$0 \quad 1 \quad 2 \quad 3 \quad 4 \times 10{,}000 \ LB/IN^2$

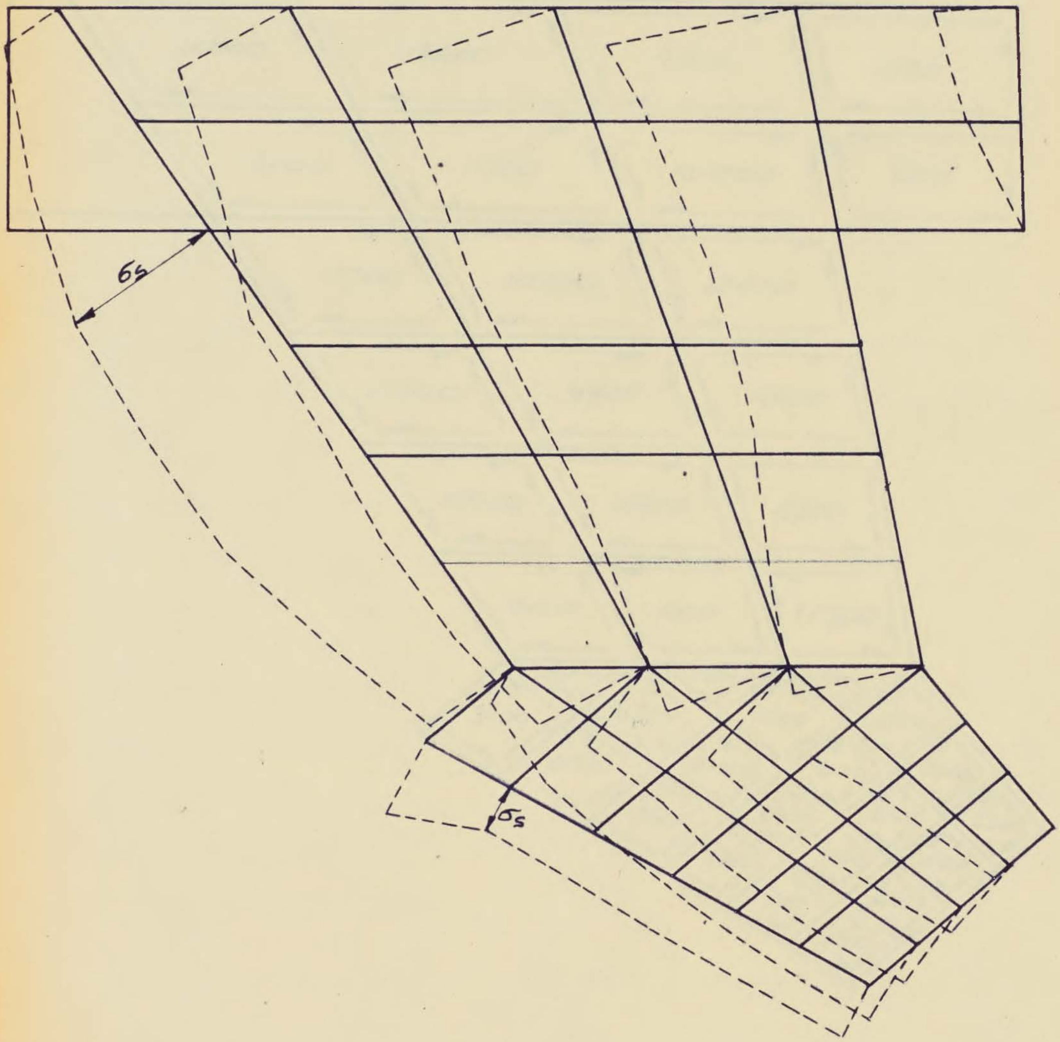


FIG. 5.

ULTIMATE PANEL SHEAR STRESSES, LB/IN.^{2}

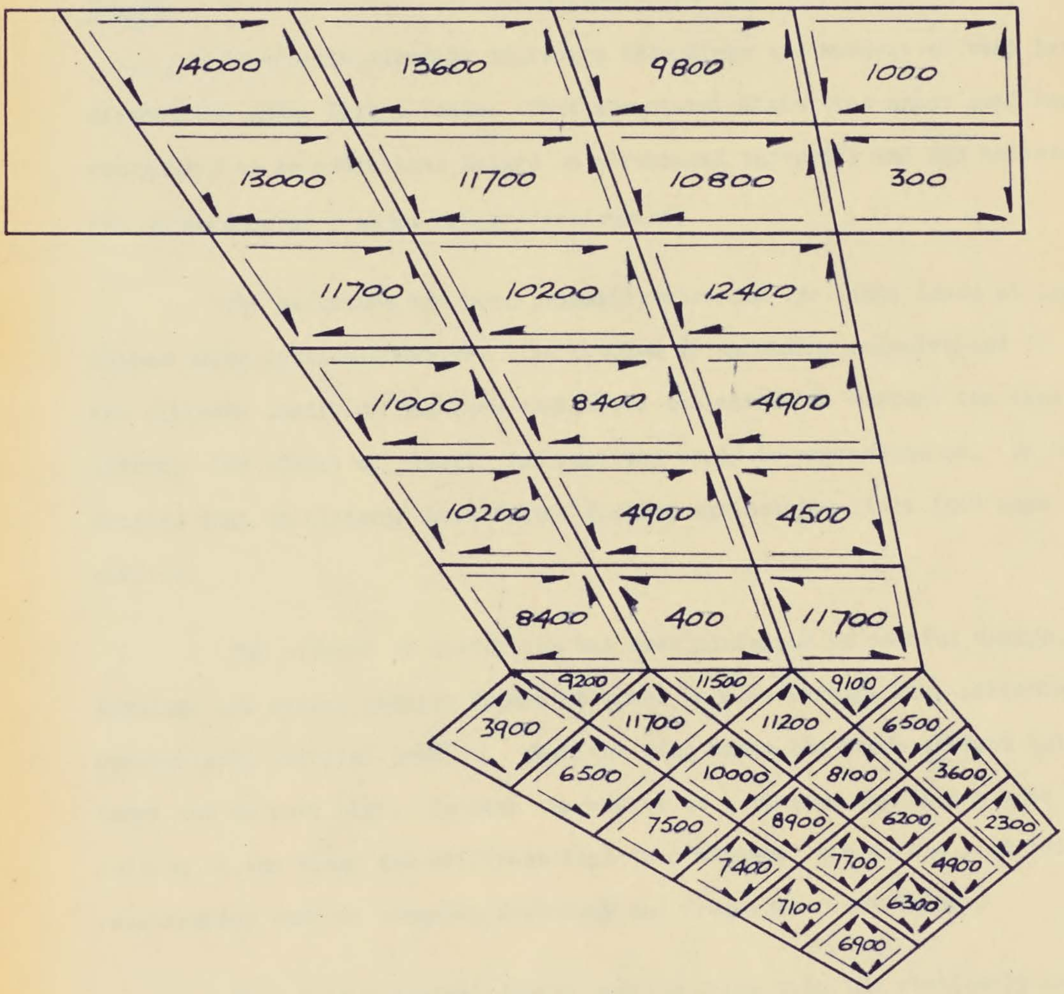


FIG. 6.

This weak point had to be remedied in some way. The major structural arrangements and scantlings had to be closely approximated by using broad assumptions and very simplified strain energy analysis. This work was done prior to the matrix analysis and kept up to date using results from the matrix.

It has been found that the two types of analysis complement each other and provide a very good check.

Shapes

Supersonic aircraft employing thin wings are subject to very large distortions under flight loads. This structural distortion under load has been recognised as an additional hazard to structural integrity and has necessitated its acceptance as a major design requirement.

The structure has been primarily stressed for limit loads at limit shapes using yield allowables. The problem of extending calculations to cover the ultimate condition has been considered too severe to warrant the time and effort. The effect of plastic deformations would be unpredictable. It has been decided that an ultimate load factor applied against the limit load case would suffice.

The effects of distortion has been minimised by careful design. Fuselage and centre section frames attached to a deflecting wing presented a particularly critical problem. When the wing bends the loads induced into a frame can be very high. To keep the effect to a minimum the frames were pin jointed at the wings and stiffness kept to a minimum, compatible with all other requirements such as longeron stability and frequency requirements.

Fig. 7 is a typical centre section frame with the statically determined bending moments as well as the induced bending moments shown.

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The frames are supported by struts from the wings. These struts increase the distortion effects but decrease the bending moments on the frames due to statically determined loads. The overall effect of the struts is however advantageous.

The struts also increased the frequency of the frame to a satisfactory level and improved the primary longeron stability problem.

A more difficult analytical problem regarding shapes is the transition problem between a deflecting structure as shown in Fig. 7 and a more rigid type of former forward of the wing in the area of the intake ducts and fuselage fuel tanks. Since it was not possible to make this type of former flexible enough to absorb the distortion, it was necessary to design a heavy frame at Station 485 to reduce distortions in that area to an acceptable value. Fig. 8 shows the shape of the fuselage along the centre line of the fuselage and along the side of the fuselage. The restraining effect of Former 485 is shown.

Internal Pressures

Supersonic aircraft are literally flying pressure vessels. In the past the only significant pressures from a structural point of view have been in the cockpit and fuel tanks. Now the complete aircraft structure is subject to variations of pressure and depressions of quite high magnitude.

This fact has influenced the structural design greatly, and has required a great deal of ingenuity in order to handle the problem with a minimum of weight penalty.

The analytical problems have not been severe, although it has proven to be a particularly interesting problem in several respects. It has opened a

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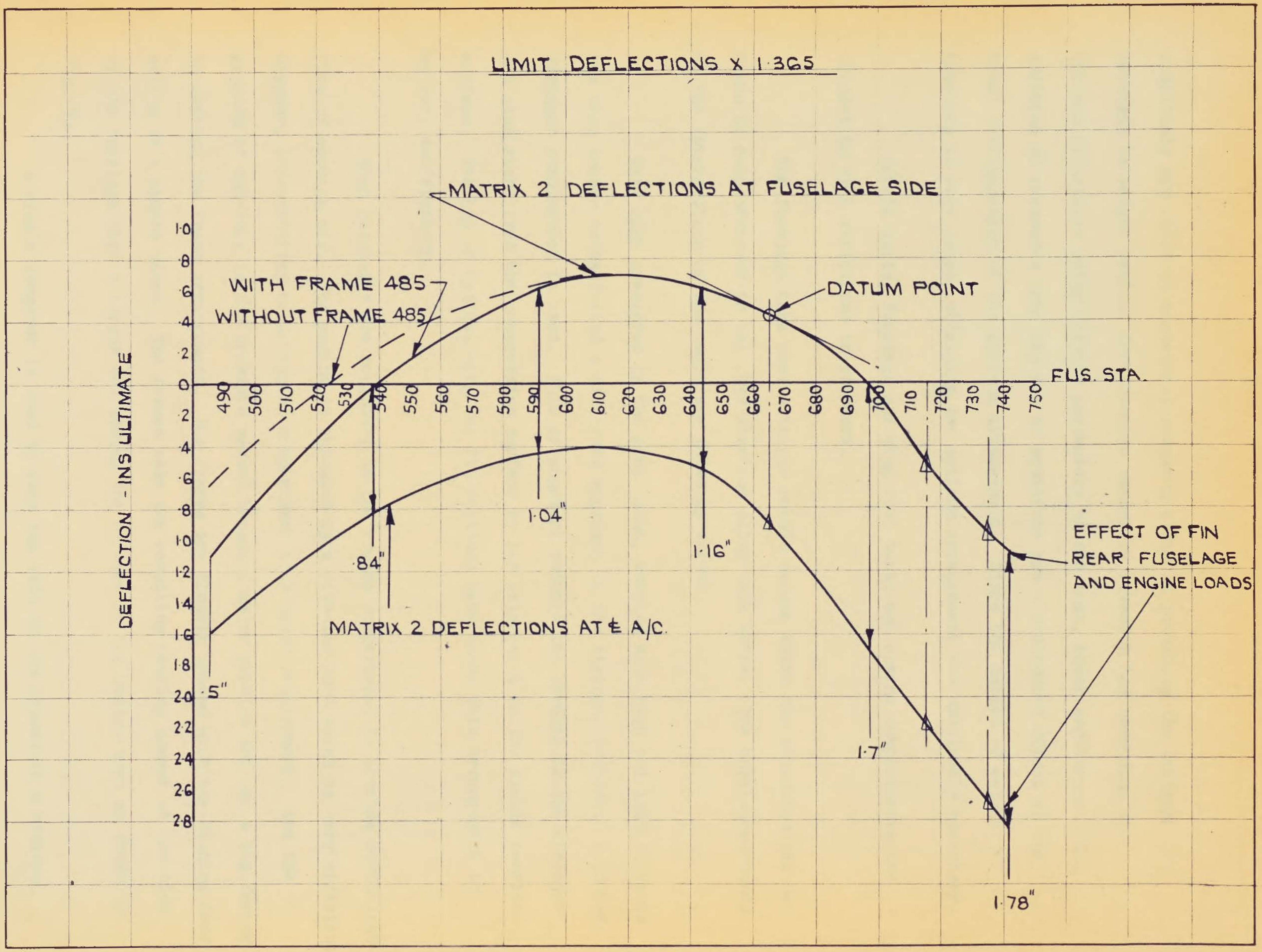

LIMIT DEFLECTIONS x 1·365
MATRIX 2 DEFLECTIONS AT FUSELAGE SIDE
WITH FRAME 485
WITHOUT FRAME 485
DATUM POINT
FUS. STA.
DEFLECTION - INS ULTIMATE
1·0
·8
·6
·4
·2
0
·2
·4
·6
·8
1·0
1·2
1·4
1·6
1·8
2·0
2·2
2·4
2·6
2·8
490
500
510
520
530
540
550
560
570
580
590
600
610
620
630
640
650
660
670
680
690
700
710
720
730
740
750
·5"
·84"
1·04"
1·16"
1·7"
1·78"
MATRIX 2 DEFLECTIONS AT ℄ A/C.
EFFECT OF FIN
REAR FUSELAGE
AND ENGINE LOADS

FIG. 8

completely new field in structural research aimed at providing the optimum strength to weight ratio. Considerable amount of research has been done on optimum structures using varying parameters of end load, shear and torque. The addition of pressures into the set of equations has a pronounced effect on the final configuration of the optimum arrangement. Since the amount of weight involved is very large, obtaining the optimum arrangement was absolutely necessary.

Intake ducts, fuselage and wing fuel tanks and engine compartments are subject to high variations in pressure.

The fuselage tank uses a triple bubble scheme where the pressures can be taken by hoop stresses in the .032 Aluminum Alloy tank skins. The unbalanced loads at the intersection of the radii are taken by struts.

Very high pressures in the wing tanks, coupled with high end load stresses and skin shears necessitated a different approach to the stringer problem. A posted stringer combination is used. This arrangement reduces the bending in the stringer and also stabilises the compression surface by its attachment to the stable tension surface. Testing of typical wing and fin sections have shown this arrangement to be very satisfactory.

High pressures are necessary in the engine compartment to provide sufficient flow of cooling air. The engine is shrouded by a circular duct which is very efficient. However, accessibility to an engine compartment is a major requirement, and the problem of cut-outs in the pressure vessel became a major problem and was a big factor in setting the frame arrangement. The frames are closely spaced with the closing door acting as a pinned panel. The frames take the resulting bending moment at the side of the fuselage with a torque box taking the off-set in the centre area as shown in Fig. 9.

A single longeron is used to keep the cuts in the frames to a minimum.

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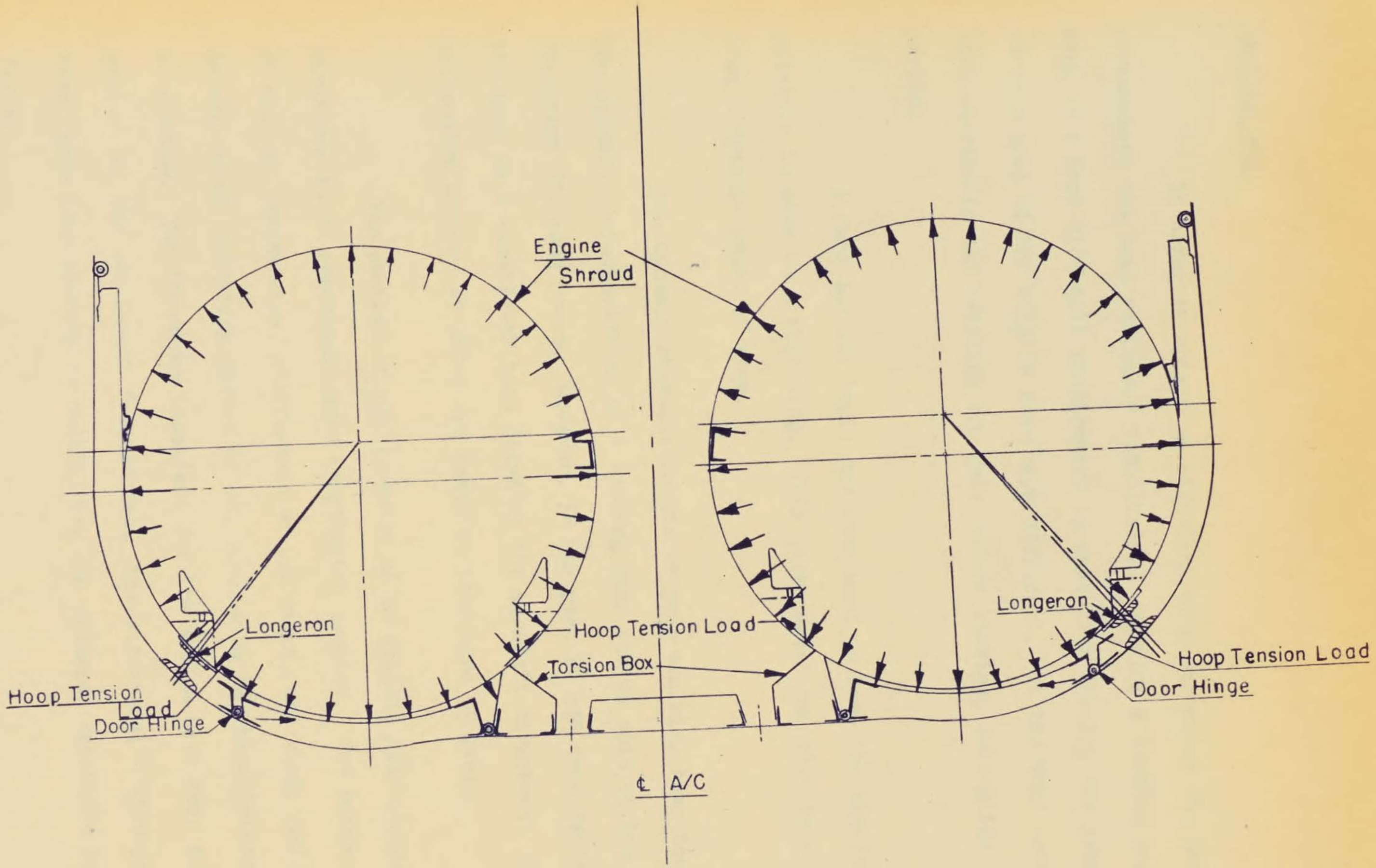

SECTION AT LIGHT FORMERS
IN WAY OF DOORS

FIG. 9.

Landing Gear

In order to obtain an efficient wing structure the landing gear arrangement has been seriously complicated. Stowing a landing gear in a thin wing is a very difficult requirement to meet, but forcing the gear to stow in a certain area of the wing is even more difficult. It has been accomplished by both shortening and twisting the gear before stowing, during the retraction period.

A fore and aft bogie has been used to keep the wheels as narrow as possible to stow in a thin wing. This arrangement has also been advantageous from a dynamic point of view.

The dynamic characteristics of gear and wing have been a major problem. The dynamic characteristics of a landing gear during spin up has been a well known phenomena for many years. However, in the past it has been satisfactory to assume the gear as a cantilever beam fixed at the aircraft structure. Recommended magnification factors of ANC-2 are based on this type of gear.

The problem of the dynamics of an elastic gear mounted on an elastic structure is an enormously more complicated problem. The method for calculation of natural frequencies, stiffnesses of U/C strut, and wing, has been developed. An equivalent vibrating system of U/C strut-wing has been introduced with six degrees of freedom. The frequency equations for the system have been solved by iteration method for the two lowest frequencies. The structural - aerodynamic damping has been taken into account by multiplying the undamped responses by a calculated decay factor.

Design parameters also had to be altered to derive the optimum dynamic characteristics. Metering pins, spring controlled valves, geometric arrangement of fore and aft bogie can be used to alter the forcing function due to spin-up and

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thereby significantly alter the dynamic characteristics.

Fig. 10 shows the idealised structure assumed for dynamic calculations and its distorted shape.

One of the most interesting results of these calculations is the fact that the critical case for the gear was found to be a dynamic case acting upwards and forwards when the coefficient of friction is very low. This is due to the fact that the gear is in fact cantilevered from the main wing box. An increase in the coefficient of friction has the effect of directing the net reaction through the centre of the main wing box.

Fig. 11 shows a typical forcing function with the resulting displacement function steps in the forcing function as typical of a fore and aft bogie, where the aft wheel hits and proceeds to "spin up" before the forward wheel hits. The forcing function differs greatly depending on the coefficient of friction and the attitude of the aircraft.

Heat

The range of aerodynamic heating on the C-105 aircraft still allows the use of aluminum alloys in an efficient manner.

It has been conceded that during the temperature manoeuvre, the design acceleration on the aircraft may be reduced. This is in line with elevator effectiveness which reduces with increased altitude and in line with the prime mission of the aircraft.

The temperature case does not therefore cause a serious weight penalty to the aircraft.

It is important however to realize that the performance of the aircraft may improve during its lifetime. There is therefore a programme underway to

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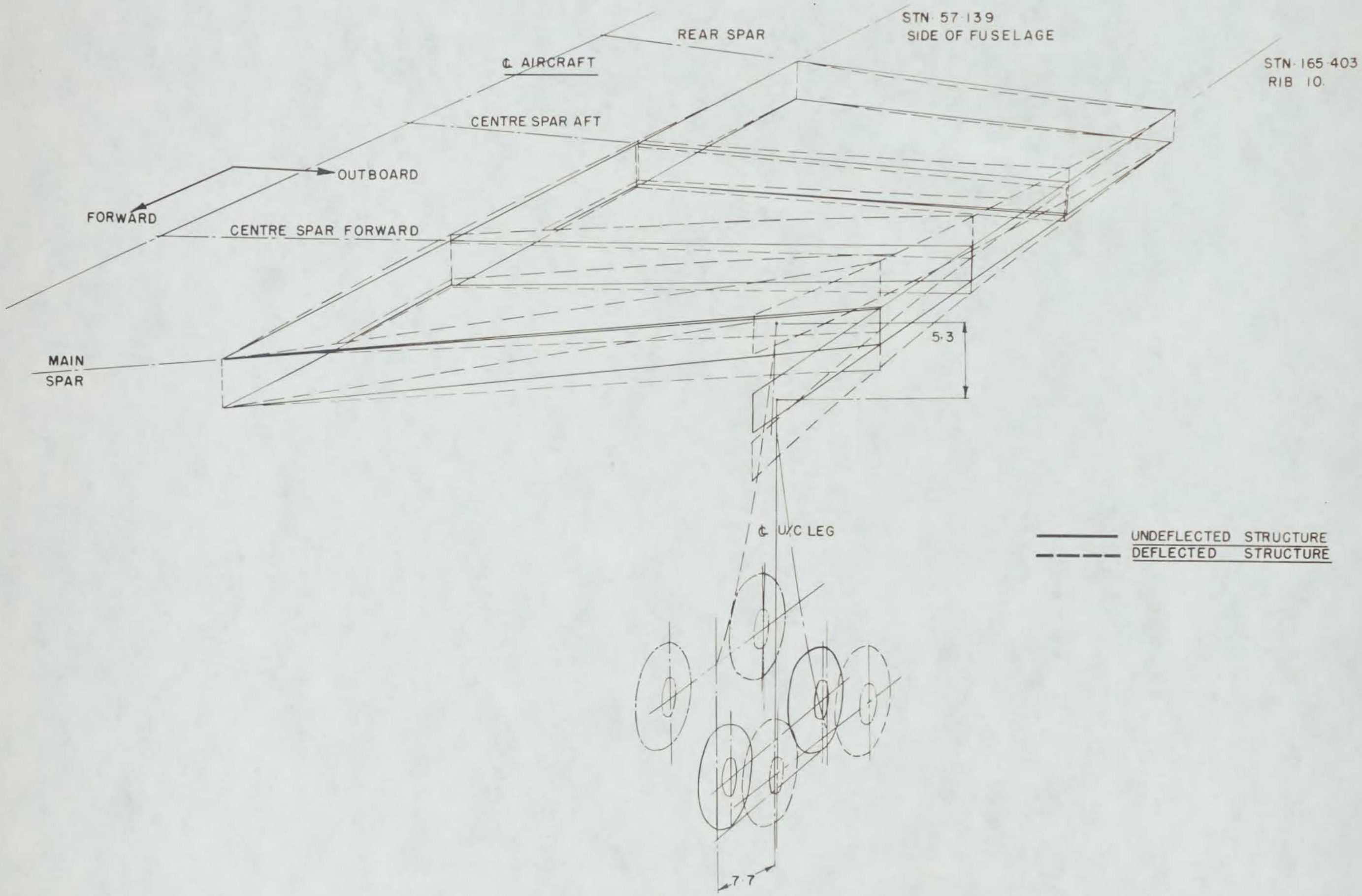

DISTORTION OF LANDING GEAR AND STRUCTURE

FIG 10

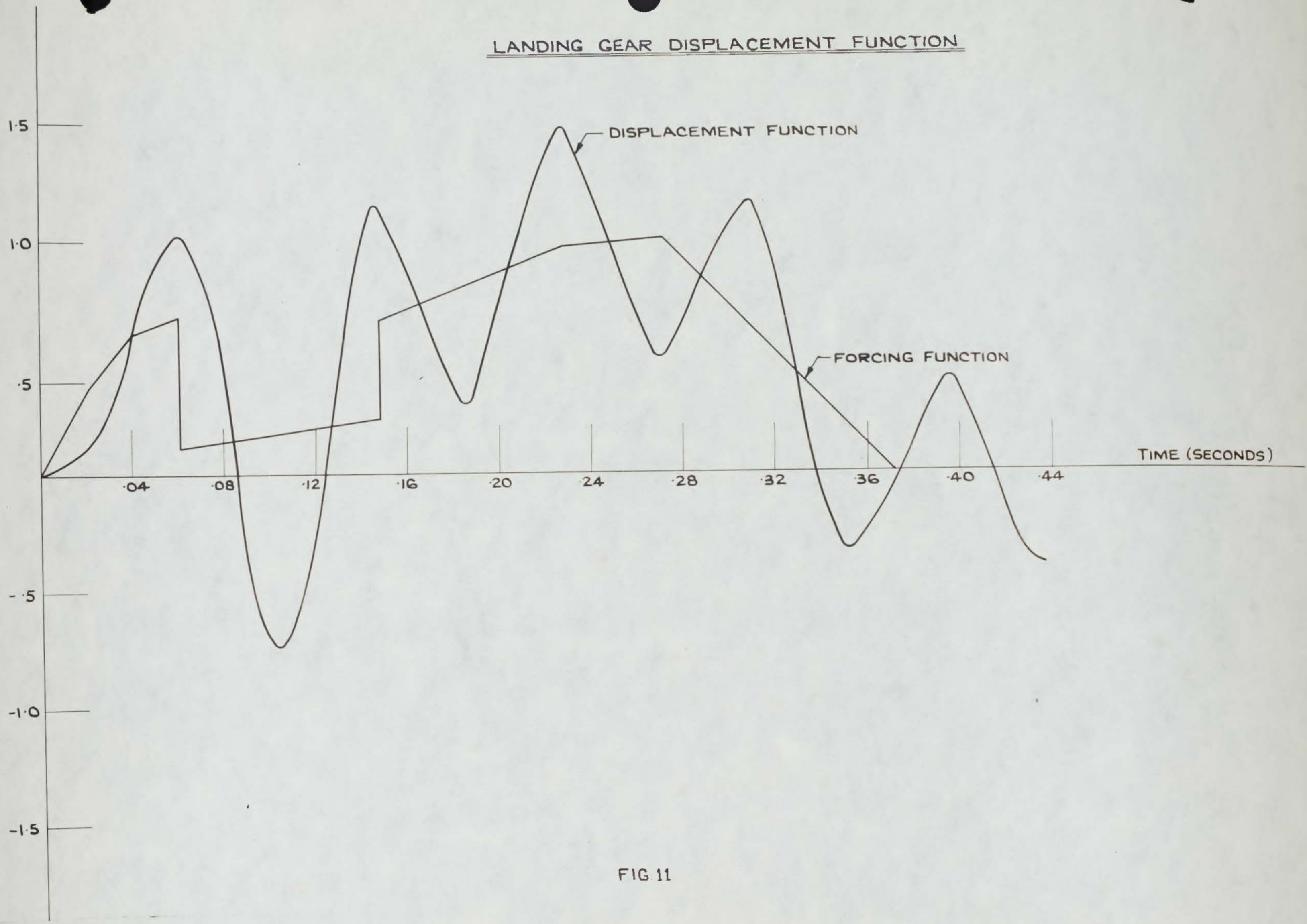
LANDING GEAR DISPLACEMENT FUNCTION
DISPLACEMENT FUNCTION
FORCING FUNCTION
1·5
1·0
·5
- ·5
-1·0
-1·5
·04
·08
·12
·16
·20
·24
·28
·32
·36
·40
·44
TIME (SECONDS)

FIG 11

estimate the allowable aircraft accelerations at higher Mach. numbers. Analytical work as well as flight test probing will be necessary to obtain this goal.

Fig. 12 is a graph of ultimate tensile strength plotted against temperature for 75ST6 plate and 24ST3 plate.

From this curve it is clear that 75ST6 material for a given aircraft weight will allow considerably higher aircraft accelerations for all flight conditions having a temperature of 270° F or lower. Above this temperature 24ST3 will allow higher aircraft accelerations.

It has been agreed that even with possible aircraft performance improvements, the great majority of flying hours will be at flight conditions below 270° F. For this reason 75ST6 has been used extensively.

In order to safely probe for higher temperature conditions the effects of transient temperature conditions as well as creep will have to be carefully analysed.

The determination of transient temperatures and thermal stresses in wings proceeds as follows.

The aeroplane is assumed to fly initially at $M = 1.0$ which, in the stratosphere, produces an equilibrium temperature of 0° F approximately. One of the conditions which cause the highest thermal stresses is a dive in which the aeroplane accelerates to $M = 2.0$ within 34.5 seconds (1.1 G acceleration) and continues at that speed. The adiabatic wall temperature is 250° F. Assuming a heat transfer coefficient h = Btu/hr. ft.^{2}° F from boundary layer to skin, neglecting the effect of finite conductivity of the skin, and assuming a fairly good thermal bond between skin and web (h = 1000), curves of temperature variation

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with time at various points of the web and curves of temperature distribution in the web at various times have been calculated (Fig. 13 & 14). The maximum thermal stress occurs when the temperature drop in the structure is maximum, i.e. at 178 seconds after beginning of acceleration. The maximum thermal stress is approximately 26,000 psi (tensile) at the web centre line.

Manufacture of test specimen is under way. Extensive testing will be done using a typical wing box with internal fuel and pressures. The flow of heat and the stresses will be measured with the box under varying applied external loads. Heat will be applied under a very close time control.

Fatigue

The presence of heat and high internal pressures aggravated the fatigue problem considerably. It is generally agreed that one cannot afford a weight penalty to improve the fatigue life of a fighter aircraft. It is however, more important than ever that the best fatigue life to weight ratio be obtained and that the structure has a uniformity of fatigue life, i.e. there should be no point where the stress concentration factors greatly exceed the average factor.

With this philosphy in mind, specimen testing of joints, fittings etc. has been inaugurated in the early design stage of the aircraft. A considerable amount of development testing was found necessary and the results incorporated into the aircraft.

A particularly interesting example is in relation to joints in thick wing skins. A low aspect ratio wing is subject to quite high chordwise bending stresses. The typical spanwise lap joint proved to be particularly bad from a fatigue point of view. The offset of load from the skin to the joint plate causes very high local stresses in both plates.

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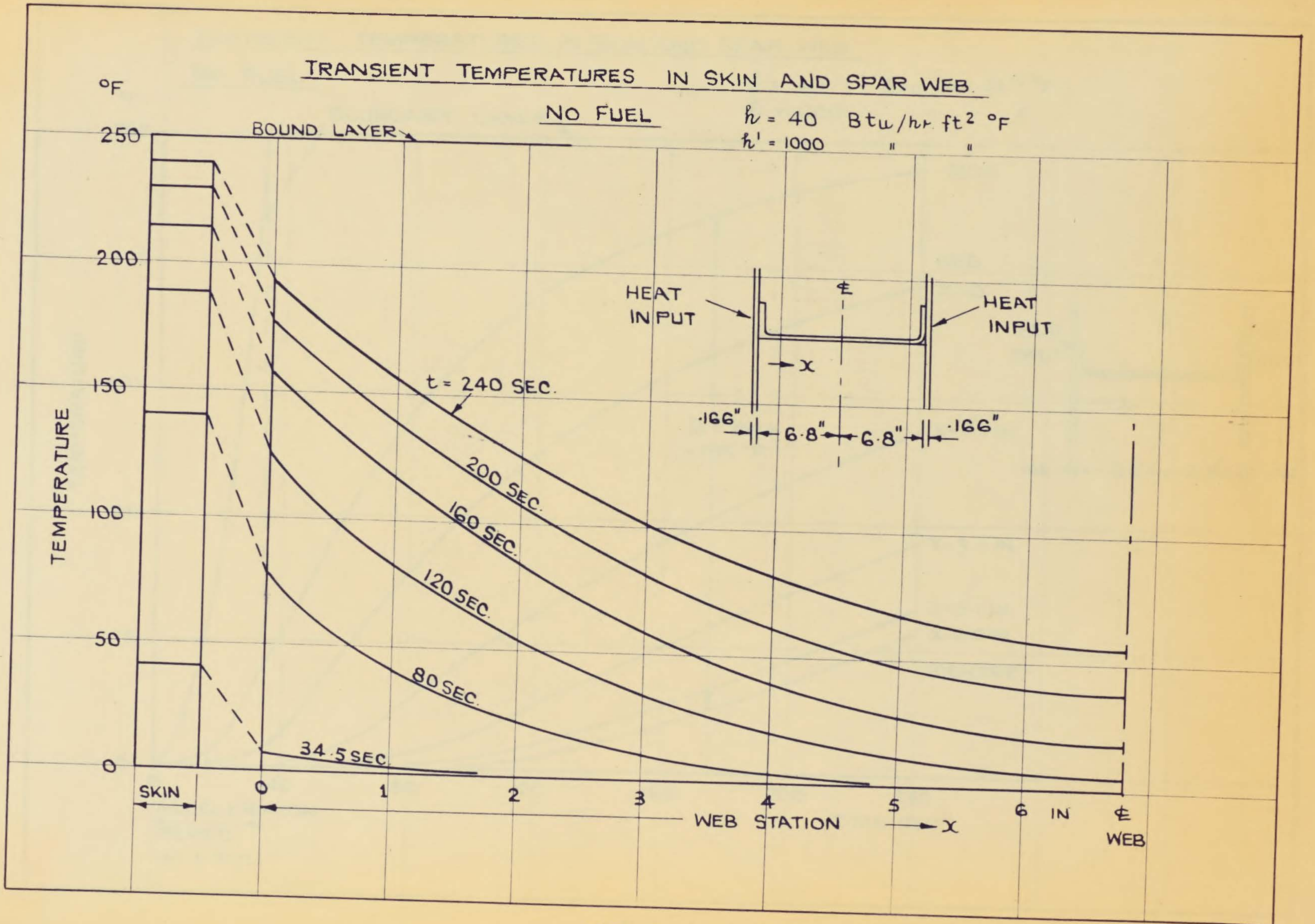
TRANSIENT TEMPERATURES IN SKIN AND SPAR WEB.
NO FUEL
h = 40 Btu/hr ft² °F
h' = 1000 " "
°F
TEMPERATURE
250
200
150
100
50
0
BOUND LAYER
t = 240 SEC.
200 SEC.
160 SEC.
120 SEC.
80 SEC.
34.5 SEC.
HEAT INPUT
HEAT INPUT
℄
x
·166"
6·8"
6·8"
·166"
SKIN
0
1
2
3
4
5
6 IN
℄ WEB
WEB STATION
x

FIG. 13.

Fig. 15 shows the result of this investigation. Type "A" is a normal lap joint. Type "B" uses a high heat treat steel strap just below the skin. This method did not improve the offset but did put the steel in the high stressed area. Since steel has a much higher endurance limit than aluminum there was a very significant improvement in the fatigue life of the joint.

Type "C" is however the optimum arrangement. A very thin steel strap is used externally by machining a step in the skin. This arrangement reduces the offset to a minimum and provides very satisfactory fatigue characteristics.

From these S.N. curves, design allowables were obtained which when compared to the actual loads, dictated the type of joint required. In fact all three types of joints are used in the wing design.

The decision to do component fatigue testing will be made following static testing. If the load distribution is found to be satisfactory and if the specimen test results are satisfactory, expensive component fatigue testing may not be justified. However, if high concentration of stresses do occur in certain areas, it may be necessary to fatigue test that particular component. (Load distribution will be determined from the static test by means of a multitude of strain gauges).

At the present, good fatigue life characteristics are dependent on good load analysis and specimen fatigue testing.

New Materials

New alloys are being used extensively to improve performance and safety of the aircraft. The following lists four relatively new alloys being used. New and better materials will be vigorously pursued.

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New Materials

1. Magnesium

The large area fuselage and relatively low loads allowed the advantageous use of a low density high strength material. Magnesium was the obvious choice. High allowables at elevated temperatures have been obtained and small production orders have been placed. Very small rivets are used to eliminate the need for dimpling. This combination produces an excellent aerodynamic surface and with a much higher point of initial buckling. Buckling occurs in the magnesium sheets at 2.75 g. Fig. 15 is a photograph of a magnesium panel under high shear loads producing tension field effects.

Proper precautions are taken against electrolytic action.

Fig. 16 shows the general use of magnesium on the C-105 aircraft.

2. Aluminum Forgings

The necessary use of high strength aluminum forgings has produced a serious problem. High strength aluminum forgings have particularly poor elongation properties in the short transverse grain as well as high residual stresses. The large scale use of X7079 ST has improved this situation considerably. This material has a guaranteed minimum elongation in the short transverse grain direction of 4% with the average elongation between 6% and 7%. This is considerably better than the 1% guaranteed for the 7075ST6. However, careful control of stresses in the short transverse grain is still required.

3. High Heat Treat Steels

Steels in the heat treat range between 240,000 psi and 260,000 psi are being used. Space requirements and weight requirements have

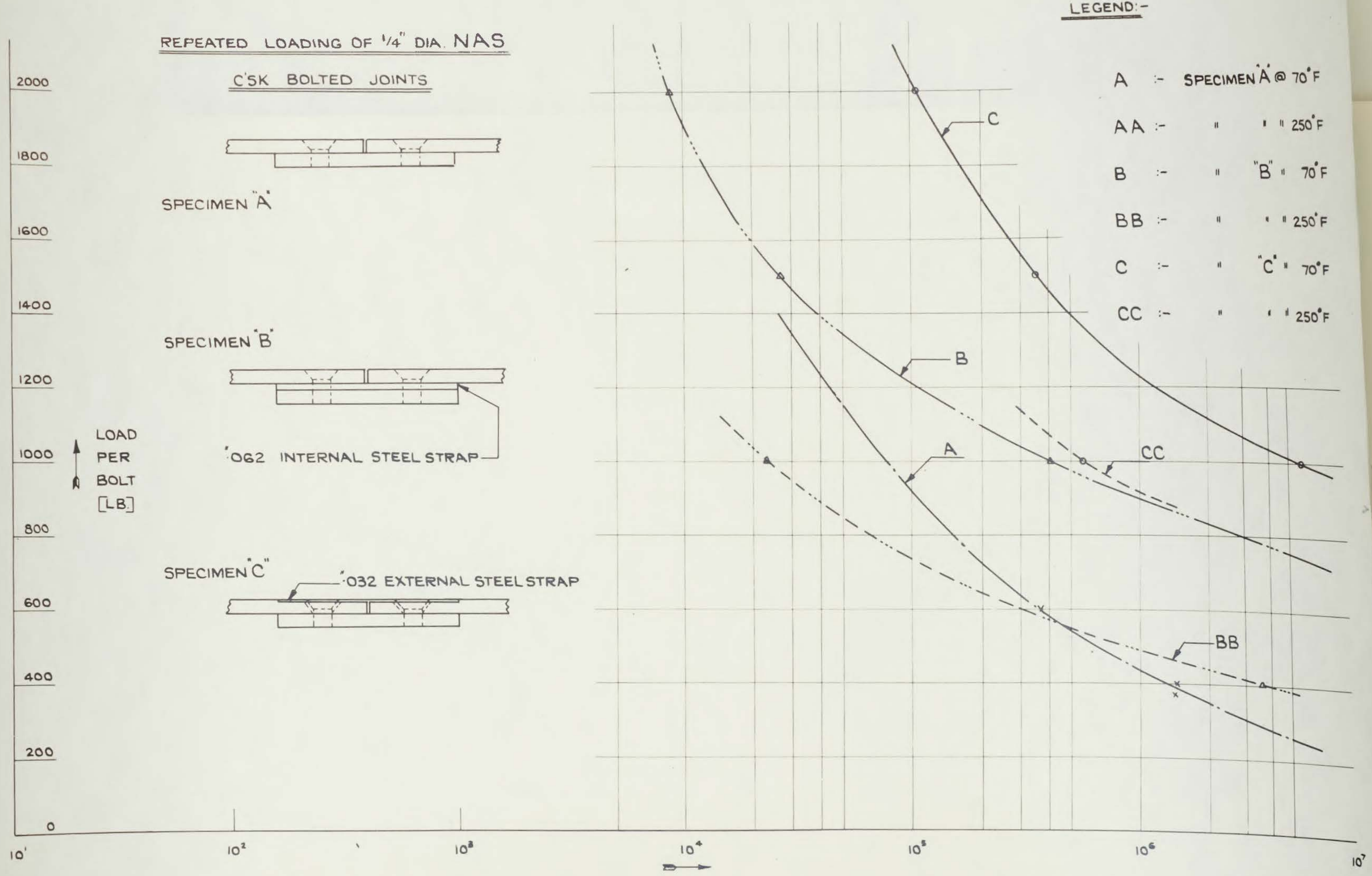
/continued ....

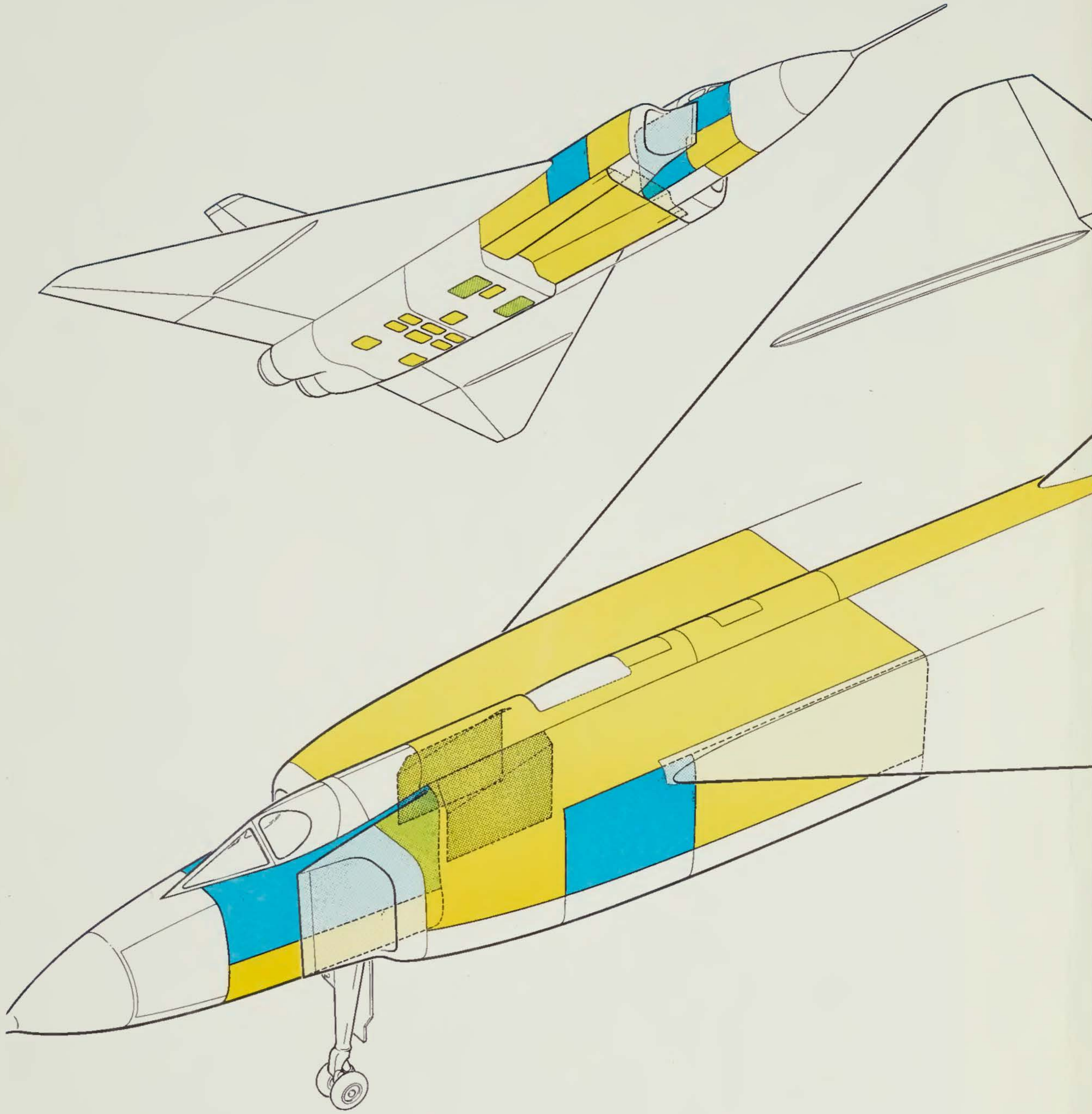
New Materials (contd.)

dictated the use of this material in some areas. Extensive research testing is under way to develop satisfactory manufacturing processes.

4. Titanium

Titanium is used in engine compartments where temperatures require something better than aluminum. In these cases titanium is considerably lighter than steel.

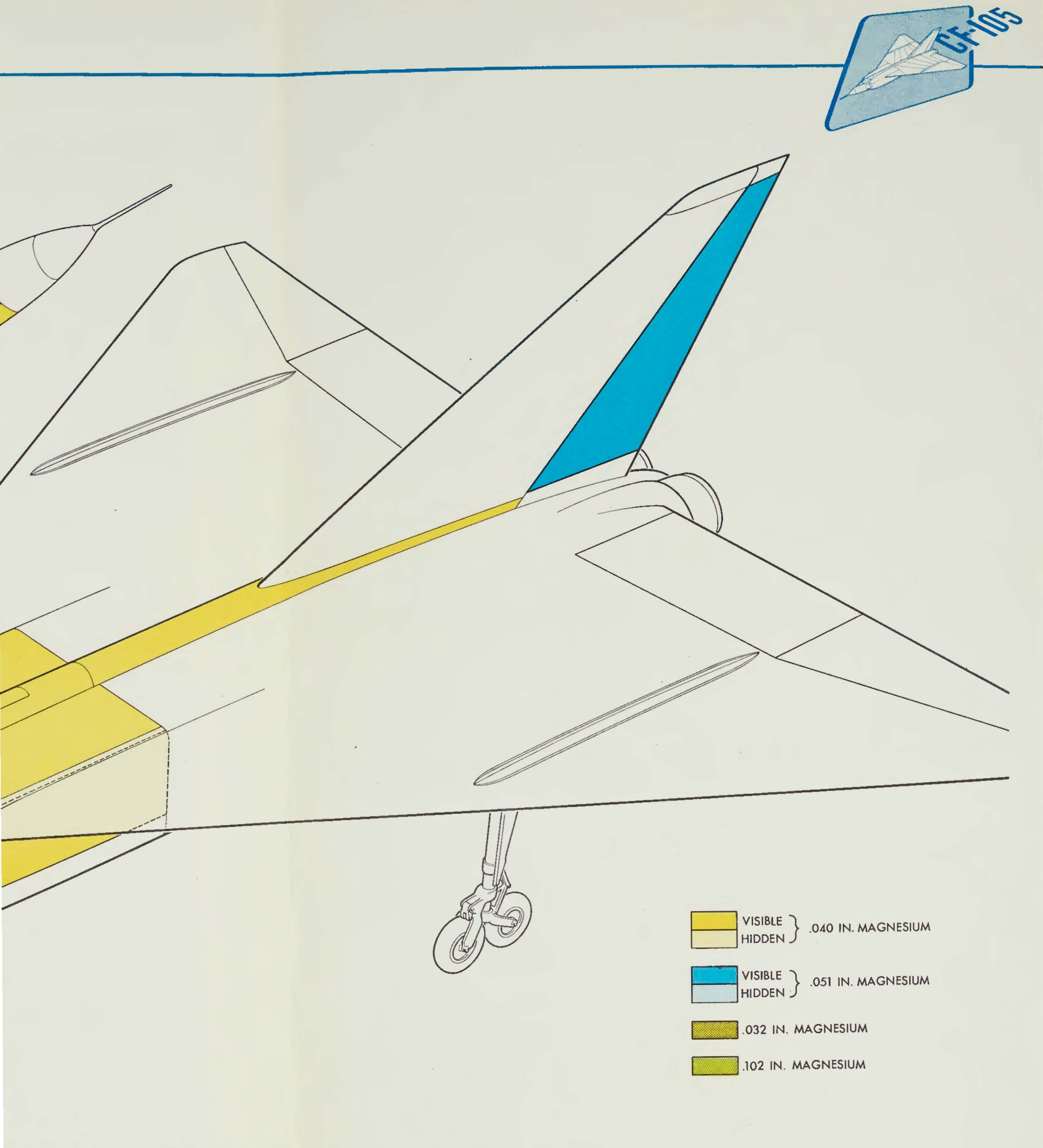

REPEATED LOADING OF 1/4" DIA. NAS
C'SK BOLTED JOINTS
SPECIMEN "A"
SPECIMEN "B"
.062 INTERNAL STEEL STRAP
SPECIMEN "C"
.032 EXTERNAL STEEL STRAP
LOAD PER BOLT [LB.]
2000
1800
1600
1400
1200
1000
800
600
400
200
0
10^1
10^2
10^3
10^4
10^5
10^6
10^7
LEGEND:-
A :- SPECIMEN "A" @ 70°F
AA :- " " " 250°F
B :- " "B" " 70°F
BB :- " " " 250°F
C :- " "C" " 70°F
CC :- " " " 250°F
A
B
C
BB
CC
NUMBER OF LOADING CYCLES
FIG. 15



CF-105 USE OF MAGNESIUM S

AVRO AIRCRAFT SPECIFICATION M-7-6 PROTECTIVE TREATMENT -

~~SECRET~~



F-105 USE OF MAGNESIUM SHEET ZH-62

CIFICATION M-7-6 PROTECTIVE TREATMENT - DOW 17 AND HEAT RESISTING ENAMEL

Fig. 15

Classification cancelled / changed to Unclassified
By authority of AVRO Arrow Declassif. Board
Date 28 Jul 87
Signature Baubrey, Co-Chairperson
Unit / Rank / Appointment DSIS 3

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