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GERMAN HIGH SPEED AIRPLANES AND DESIGN DEVELOPMENT

COMBINED INTELLIGENCE OBJECTIVES
SUB-COMMITTEE

RESTRICTED

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PREFACE

This report is not a result of any one investigation (CIOS) trip to Germany, but is a summary of information obtained on several trips and includes data obtained from interrogations of German personnel detained in London, documents from the Air Document Research Center, and reports by other agencies and investigators. It has been made as broad and comprehensive as possible at this time, but, of course, cannot be considered as a complete story of German high speed airplane design.

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GERMAN HIGH SPEED AIRPLANES AND DESIGN DEVELOPMENTS

Reported by:

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On behalf of

U.S. Technical Industrial Intelligence Committee

CIOS Items 4, 5, 6, 25, 26 & 27
Rockets & Rocket Fuels
Jet Propulsion
Directed Missiles
Aircraft
Aircraft Engines
Instruments & Equipment

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COMBINED INTELLIGENCE OBJECTIVES SUB-COMMITTEE
G-2 Division, SHAEF (Rear) APO 413

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I. HIGH SPEED AIRPLANES

1. Introduction

In the following pages, data and scale sketches for 25 of Germany's outstanding airplanes and airplane designs are given. The list cannot be considered a complete index of German aircraft, but it is quite representative of German thought on high speed airplane configurations. No attempt has been made to cover any but fast airplanes, and it should be noted that 16 of the 25 are in the 600 m.p.h. class. Only 2 (the Do-335 and BV-155) are propeller driven and these are the only two with top speeds less than 500 m.p.h. However, the BV-155 attains its maximum speed above 50,000 ft. and the Do-335 is one of the fastest propeller driven airplanes in the world.

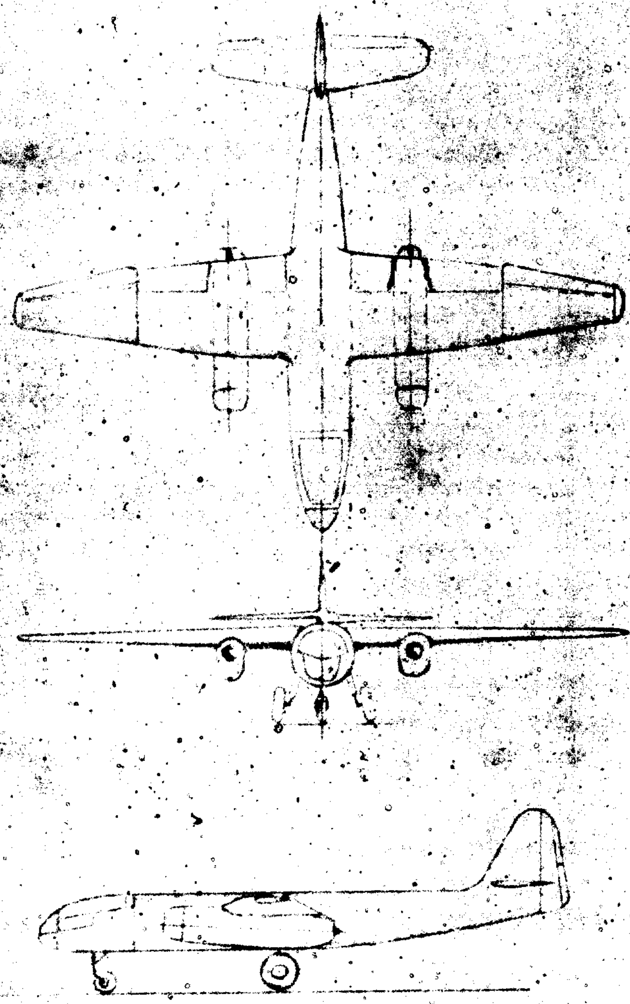
The airplanes are divided into two classes, Production Airplanes and Experimental and Proposed Airplanes. The Production Section includes the airplanes that were either in full scale production or had passed the prototype stage and were designed for full scale production, even if it had not yet been attained. Included in this latter class are the Ju-263, Ju-287, and Do-335.

The first ships covered in the Experimental Section are the eight designs submitted in the last fighter competition in Germany. Of these, the Me-1101 was being built, but only as a research airplane. (It was designed so that sweepback and dihedral could be varied). Of the remaining airplanes in the Section, only the Horten IX and the "Natter" have been flown. Lippisch's LP-11 and LP-13 were being built in glider versions and the BV-155 prototype was almost complete, but none of the designs, other than those already mentioned, had reached any sort of construction stage.

As a means of comparing German airplanes with others, an additional section has been added giving the specification list for the standard equipment required in military airplanes.

2. P R O D U C T I O N

A I R P L A N E S



AR-234 B2 JET PROPELLED BOMBER

AR-234 B-2 JET PROPELLED BOMBER

The AR-234 B2, like the earlier models is a single seat light bomber with two Junkers 004B turbo-jet units. Later models of the 234 are reported to have four jet units. It is of the high wing type with a tricycle landing gear and was of all metal construction.

This was one of the production models and was designed and fabricated by Arado Flugzeugwerke G.m.b.H. Potsdam.

Dimensions:

Span - overall	14.4 m
Length - overall	12.64 m
Height - from ground to tip of fin	4.3 m
Wing - root chord	2.55 m
Wing - tip chord	.85 m
Span - tail surfaces	5.00 m
Stabilizer - root chord	1.15 m
Tread - main landing gear	2.05 m
Wing area	26.4 m^2

Weights	Normal	With Bombs 1-500kg.	With Bombs 3-500 kg.
Max.take-off weight (tons-metric)	8.41	8.85	9.35
Limit weight-with bombs	-	7.35	8.1
" " without bombs	6.91	--	--
Landing Weight	5.71	5.65	5.6
Performance			
Speed -with bombs at 8km.alt.	-	700 km/hr.	570 km/hr.
" " " " 7km.alt.	-	705 "	600 "
" without " at 10 km.alt.	600km/hr.	--	--
" " " " 8 km.alt.	707 "	--	--
" " " " 6 km.alt.	720		
Time to climb to 8km.	18 min.	21.6 min.	34.1 min.
" " " " 6 km.	11 "	12.8 "	17.5 "
Range	1630 km.	1560 km.	1100 km.
Take-off distance without rockets.	1300 m.	1550 m.	1780 m.
Takeoff distance with rocket assist.	690 m.	965 m.	860 m.

Power Plant

2 Junkers 004B turbo-jet units were slung beneath the wing on each side of the fuselage. Provision is made for attachment of rocket units for assisted take-off.

The fuel system included two large fuselage tanks. The front one is located between the pilot's cabin and the wing and has a capacity of 1800 liters, and the rear tank located aft of the wing has a capacity of 2000 liters. Both tanks are of the flexible self-sealing type.

Provision is made under each jet unit for installation of droppable fuel tanks.

Push-pull rods are used for operation of the engine controls.

Wings

The all metal wing, including the tips, is constructed in one unit. It is of the conventional two spar stressed skin type, with nose ribs, inter spar ribs and trailing edge ribs.

The wing is attached to the fuselage at four points; two at the extreme leading edge and two at the rear face of the rear spar. Forgings and machined parts are used for this attachment. A special expansion shear bolt passes through the eye fittings on both the fuselage and wings.

The ailerons and flaps are also of the conventional type of all-metal construction. The ailerons are mass-balanced. The trailing edge of the aileron extends aft of trailing edge of the flap. High aspect ratio tabs are fitted to both the R.H. and L.H. ailerons. The tab controls are linked to the flap operating mechanism.

Tail Surfaces

Conventional type all-metal construction is used for all tail surfaces. The horizontal stabilizer is constructed in one piece. It has two sheet metal spars and pressed ribs and stressed skin. High aspect ratio tabs are fitted to both R. and L. elevators. It has been reported that in earlier models an adjustable stabilizer was used in place of elevator tabs.

A single fin and rudder also of all-metal construction is located on the centerline of the fuselage. Two tabs extend the full height of the rudder; one is used for trimming and the other is used as a Flettner control.

Fuselage

The fuselage is of the semi-monocoque construction. Formers and six stringers are of "Z" sections, while the two longerons to which the wing attachment fittings are riveted are of "hat" sections.

The nose section forms the pilot's cabin and is detachable from the main fuselage section.

Immediately behind the cabin section is the forward fuel tank and aft of this is the main attachment point. The rear fuel tank is aft of the wing fittings. A strong bulkhead separates this tank compartment from the rear fuselage. Access can be gained to the interior of the rear fuselage through two hatches in the top fuselage skin.

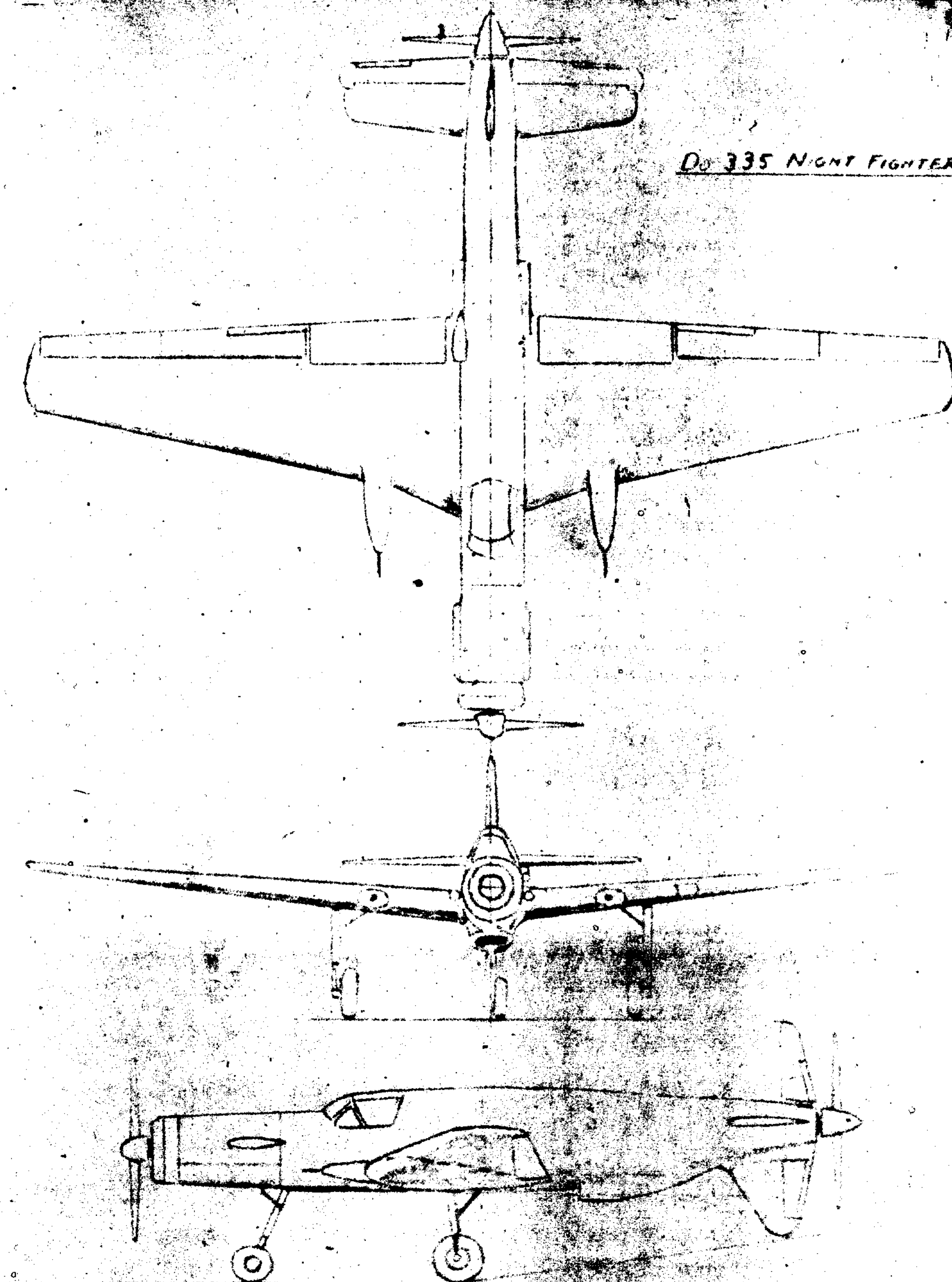
Landing Gear

The landig gear is of the tricycle type. The complete gear retracts into wells provided in the fuselage. The nose gear retracts rearward while the main wheels fold forward and inward. Both nose and main gear is retracted hydraulically.

The shock struts on the main gear have a stroke of approximately 12 inches. The large wheels have dual hydraulic brakes and are fitted with 935 x 345 mm tires.

The nose wheel tire is 630 x 220 mm.

Do 335 NIGHT FIGHTER



DO-335 2-MOTORED NIGHT FIGHTER-BOMBER

General

The Do-335 is a two engine, multi-purpose, all metal, low wing monoplane. This model has two Daimler Benz 603 liquid cooled engines in tandem. There is one tractor propeller in the front and one pursher propeller at the rear. The aircraft is equipped with a retractable tricycle landing gear.

Engine

The Do-335 is powered with two Daimler Benz 603 engines, 1800 HP each. Each engine drives a three bladed controllable pitch, metal propeller 10.5 ft. in diameter. The chord of each blade is 12.5 in. The rear propeller is driven by a hollow steel shaft 4.8" in diameter and 13.3' in length.

Fuselage

(a) The fuselage is 45'2" in length with a maximum depth of approximately 6'3". The fuselage is skin stressed construction using flush riveting. It starts off by having a circular cross section at the front gradually enlarging and elongating into an oval cross-section with a maximum cross-section at approximately the center of the fuselage. The cross section then decreases gradually keeping its oval shape up to its tailsection where it tapers off to a sharp "v". The fuselage contains the two engines, cockpit, fuel tank, radiator cooling system, a possible dinghy compartment, a compartment for the front nose wheel and a compartment whihh may be used as a bomb-bay. The forward motor has its radiator in the form of a radial engine nacelle, in the front of the fuselage directly behind the propeller. The air passes out of adjustable cowl flaps fastened to the nacelle. The rear engine has its radiator in the fuselage. A large air scoop, similar to that of the P-51 and in the same relative position, directs the air past the radiator through the fuselage and out the controllable vents in the tail. The forward section immediately behind the radiator contains the engine mounted on a special frame. This frame is rubber mounted and fastened to the fuselage by four bolts. Immediately behind the fire wall are two compartments. The upper compartment is accessible from the left side. It is 2'3" long by 3' wide by 1' deep. It may be used as a dinghy compartment. Teh lower section is for the forward nose landing gear. This gear is retracted into this compartment during flight. The compartment is 2'5" wide, 7'7" long, and 2'6" deep. This compartment has hinged doors. The cockpit contained standard engine instruments and navigational aids in addition to its accessories. The outstanding feature is the pilot seat. It has an ejection system which catapults thepilot free of the aircraft in emergencies. The cockpit is located at approximately the leading edge of the wing, allowing for good visibility.

(b) Immediately behind the cockpit there is a large fuel tank. Below this tank is another compartment which opens up from below by means of two hinged doors. This compartment is 8'11" long, 3'8.5" wide, and 2'5" deep. It provides a convenient aperture by which the rear engine can be served or serviced. This compartment may possibly be used as a bomb bay.

(c) Above the compartment and continuing beyond is the rear engine. This engine is also in the fuselage and mounted on the frame. This engine is accessible for service by two large cowlings which are fastened to a common hinge on the top side of the fuselage. Behind and below the engine in the fuselage is located the radiator for the rear engine. The air comes in the air scoop attached to the bottom of the fuselage. This air passes out again by means of an adjustable opening in the tail.

<u>Tail Surfaces</u>

(a) <u>Stabilizers</u>: The stabilizer is all metal construction with fabric on its leading edge: it is flush riveted and passes through the fuselage above the air vents at the tail. The stabilizer is fixed. It has a span of 18'2". The chord at the base is 3'4". The stabilizer tapers off to a 10" chord with rounded end.

(b) <u>Fin</u>: This fin is similar in construction and dimension to the stabilizer. The fin passes through the fuselage at right angles to the stabilizer. The fin extends an equal distance above and below the fuselage. The tip on the lower portion of the fin is of cast metal so as to protect it from damage on accidental contact with the ground.

(c) Elevator: The elevator is aerodynamically balanced. It is of all metal construction and is equipped with movable tabs. It has a span of 19' with rounded tips. Its chord at the base is 1' 9.5". The chord near the tip is 1' 8;5". The tab is in two sections with a 3" spacing in between.

(d) Rudder: The rudder is of all metal construction. It is equipped with movable tabs. The bottom rudder is several inches shorter than the top one because of metal casting on the fin.

Wings

(a) The wing is of all metal construction. It has a span of 45'1". The wing has a symmetrical taper with a square tip and rounded corners. The chord of the wing at the root is 12'; the chord at the tip is 6'8". The length of the wing from root to tip is 20'7". The depth of the wing at the root is 1'8.5".

(b) Attached to the wings are hinged flaps and ailerons. The ailerons are 9'11.5" long with chord of 1'4.5". Attached to the ailerons are movable tabs 5'3.5" long and 3.25" wide. The hinged flaps are 8'5.5" long and 2'4.5" wide.

(c) The wing has two built up metal spars. The metal covering is flush riveted. There is a compartment for the wheel fitted into the wing. The wing also contains two gasoline tanks. In the leading edge, half-way out on the wing, there is a built-in landing light.

Landing Gear

(a) The landing gear is the tricycle type. The main wheels each fold inward into the wings when retracted. The fairing is attached to the outside of the landing gear so that when the wheels are retracted the fairing fits flush with the wing. The nose gear folds backward into the fuselage and the compartment doors close when the wheels are fully retracted. The main gear is attached to the wings and located at approximately the center of the chord. The wheels have a tread of 16'6". The nose gear is 13'2" forward of the main gear.

(b) The landing gear is also oleo type and is hydraulically retracted. The nose wheel tire is 685 by 250 mm. and the main gear tires are 1015 by 380 mm.

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DO-335

NIGHT FIGHTER

	A-6 Adapted as night fighter		B-6 Wing area 414 sq.ft.		B-6 Wing area 441 sq.ft.		B-7	
Wing area	414		414		441		441	
Engine	DB 603 E		DB 603 E		DB 603 E		DB 603 LA	
Take-off h.p. (with MW50)	2150 (2400)		2150 (2400)		2150 (2400)		2300	
Combat rating at full power height with ram, h.p.	1470		1470		1470		1430	
Max. continuous at full power height with ram, h.p.	1280		1280		1280		1180	
Total petrol capacity, gall.	510		471		471		471	
Power boosting (MW 50) gall.	33		33		33		44	
Take-off weight, lb.	22230		22710		22800		23160	
Armament in nose	2 x MG 151/20 (200 r.p.g.) 1 x MK 103 (70 rounds)		2 x MG 151/20 (200 r.p.g.) 1 x MK 103 (70 rounds)		2 x MG 151/20 (200 r.p.g.) 1 x MK 103 (70 rounds)		2 x MG 151/20 (200 r.p.g.) 1 x MK 103 (70 rounds)	
	FuG 220	FuG 218	FuG 218	FuG 218	FuG 218	FuG 218	FuG 218	FuG 218
	Aerial Ext.	Aerial Ext.	Aerial Ext.	Aerial Int.	Aerial Ext.	Aerial Int.	Aerial Ext.	Aerial Int.
Max. speed at emergency without MW 50, m.p.h.	416	426	427	448	426	447	--	--
with MW 50, ft.	419	430	430	451	429	451	453	476
Full power height with ram without MW 50, ft.	21000	21000	21000	21000	21000	21000	--	--
with MW 50, ft.	17700	17700	17700	17700	17700	17700	32000	32000

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DO-335

	A-0/A1 Adapted as heavy fighter and bomber	HEAVY FIGHTER B-2	HEAVY FIGHTER B-3	TRAINER A-10 A-11	TRAINER A-12
Max. speed at emergency					
without MW50, m.p.h.	477	460	--	472	478
with MW 50, m.p.h.	--	465	490	--	--
Full power height with ram					
without MW 50, ft.	21,000	21,000	--	21000	21000
with MW 50, ft.	--	17,700	32,000	--	--
Cruising at max.cont.mph.	428	416	416	427	427
At ft.	23300	23300	30500	23300	23300
Range at max.cont., miles	868	994	931	508	725
Endurance @ max. cont. hr:min.	2:0	2:30	2:14	1:11	1:40
Econ. cruising @ 19700 ft.					
speed, m.p.h.	295	303	303	270	280
range, miles	1280	1340	1170	776	1090
endurance, hr:min.	4:22	4:11	3:54	2:53	3:52
Time to climb to 26250 ft. @ take-off wt., min.	13.3	17.8	13.7	9.5	11.4
Service ceiling @ mean weight, ft.	37400	34500	39700	40000	38400
Landing speed, m.p.h.	109	114	114	106	109
Landing run, yds.	689	832	832	602	689
Landing run with reversible pitch prop., yds.	514	624	624	448	514

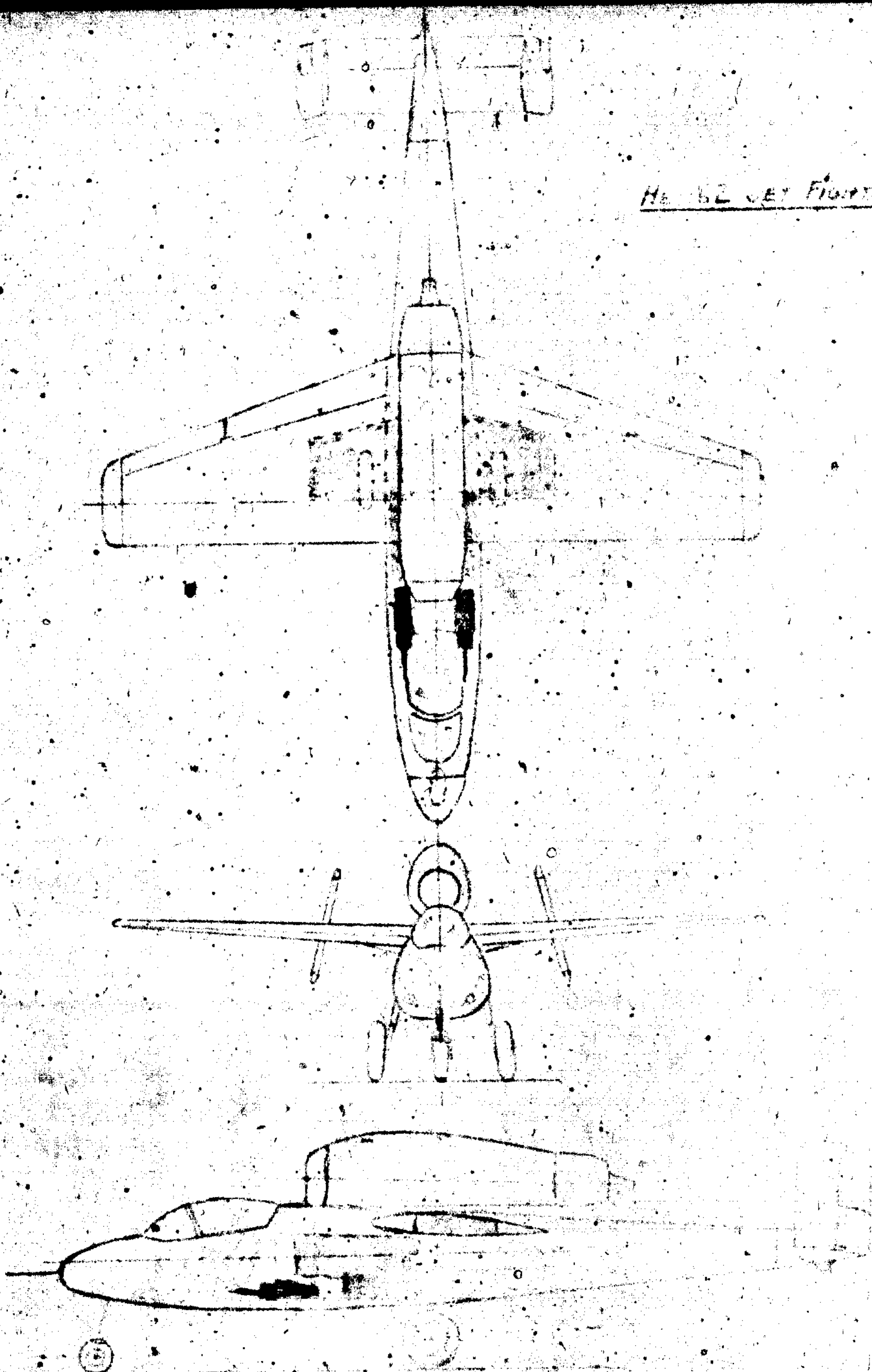
DO-335

	HEAVY FIGHTER			TRAINER	
	A-0/A1 Adapted as heavy fighter and bomber	B-2	B-3	A-10 A-11	A-12
Wing area, sq.ft.	414	414	414	414	414
Engine	DB 603 E	DB 603 E	DB603 LA	DB603 E	DB 603 E
Take-off h.p. (with MW 50-Methanol injection)	2,150	2,150 (2,400)	2,300	2,150	2,150
Combat rating at full power height with ram, h.p.	1,470	1,470	1,430	1,470	1,470
Max. continuous at full power height with ram, h.p.	1,280	1,280	1,180	1,280	1,280
Total fuel capacity, gall. (British)	407	473	473	253	335
Power boosting (MW 50) gall. (British)	--	33	44	--	--
Take-off weight, lb.	21,160	23,380	23,700	17,850	19,600
Armament in nose	2 x MG 151/20 (200 r.p.g.) 1 x MK 103 (70 rounds)	2 x MG 151/20 (200 r.p.g.) 1 x MK 103 (70 rounds)	2xMG151/20 (200 rpg) 1x MK 103 (70 rounds)	--	--
Armament in wing	--	2 x MK 103 (140 r.p.g.)	2 x MK 103 (140 r.p.g.)		

DO-335
NIGHT FIGHTER

	A-6 Adapted as night fighter		B-6 Wing area 414 sq.ft.		B-6 Wing area 441 sq.ft		B-7	
Cruising @ max.cont. speed, m.p.h. @ ft.	369 23300	379 23300	379 23300	400 23300	378 23300	399 23300	382 30500	403 30500
Range @ max.cont., miles.	886	911	863	886	863	886	825	886
Endurance @ max.cont., hr:min.	2:26	2:26	2:17	2:17	2:17	2:17	2:14	2:14
Econ.cruising @ 19700ft. speed, m.p.h. range, miles endurance, hr:min.	275 1280 4:46	279 1330 4:51	283 1190 4:12	292 1280 4:25	268 1290 4:50	275 1380 5:02	268 1130 4:12	276 1220 4:23
Time to climb to 26250ft @ take-off wt., min.	18.2	18.2	17.2	17.0	16.1	15.9	12.3	12.1
Service ceiling @ mean weight, ft.	33400	34700	34000	34350	35650	36000	41250	41600
Landing speed, m.p.h.	112	112	114	114	110	110	111	111
Landing run, yds.	766	766	810	832	788	810	832	854
Landing run with reversible pitch prop., yds.	580	580	602	624	580	602	613	635

He 162 Jet Fighter



HE-162 JET FIGHTER

General Description

The 162, as shown in the attached three-view drawing, is a shoulder-wing monoplane of mixed construction, employing as little as possible of material which is in short supply. The most remarkable features of the design are the low span/length ratio, the mounting of the jet unit above the fuselage, the tricycle undercarriage which retracts into the fuselage, and the sharp dihedral of the tailplane. Wing tips bent down 55° were added to the latest airplanes because of stability troubles.

Designation

Official German documents generally describe the aircraft as "Type 162" or as "8-162" although references to the He-162 have been found. The name "Volksjaeger" is popularly used.

Dimensions and Weights

Wing span	23' $7\frac{3}{4}$"
Root chord	6' $8\frac{1}{2}$"
Wing area (gross)	120 sq.ft.
Aspect ratio	4.65
Overall length	29' $8\frac{1}{2}$"
Tailplane span	7' $5\frac{1}{2}$"
Tailplane chord	2' $8\frac{1}{2}$"
Tailplane dihedral angle	14 degrees
Undercarriage track	4' 11"
Normal all-up weight	5,480 lb.
All-up weight with maximum fuel	5,940 lb.
Landing weight with 20 per cent fuel	4,820 lb.
Wing loading at landing	40 lb.per sq.ft.

Performance:

The following performance figures have been taken from the makers' specification and are presumably subject to the usual tolerance of ± 3 per cent.

Maximum speed, at sea level	490 m.p.h.
at 19,700 ft.	522 m.p.h. (One document gives a max.

	speed of 515 mph at an unspecified altitude.)
at 36,000 ft.	485 m.p.h.
Rate of climb, at sea level (at mean weight)	4,230 ft/min.
at 19,700 ft.	2,460 ft/min.
at 36,000 ft.	690 ft/min
Time for climb to 19700ft. (at mean weight)	6.6 min.
to 36,000 ft.	20.0 min.
Ceiling at mean weight	39,400 ft.
Full throttle range with normal fuel, at sea level	136 miles
Full throttle range with normal fuel,	
at 19,700 ft.	267 miles
at 36,000 ft.	410 "
at 38,400 ft.	434 "
Full throttle range with maximum fuel,	
at sea level	242 miles
at 36,000 ft.	620 "
Full throttle endurance (normal fuel) at sea level	20 min.
at 19,700 ft.	33 "
at 36,000 ft.	67 "
Full throttle endurance (maximum fuel) at sea level	30 min.
at 36,000 ft.	85 "
Take-off run (normal fuel)	
without A.T.O.	710 yd.
with A.T.O.(2,200 lb. thrust)	350 yd.
Take-off run (maximum fuel)	
without A.T.O.	875 yd.
with A.T.O.(2,200 lb. thrust)	415 yd.
Landing speed	102 m.p.h.

Wing

The materials used in the construction of the cantilever wing are:

Spar boom	TBu 20 (presumed wood)
Webs and skin	Beech plywood
Ribs	Pine booms with plywood webs.

Fittings and connecting bolts	Steel
End caps	Aluminum alloy

The wing has a straight leading-edge, blunt tips and a pronounced sweep-forward on the trailing-edge. The dihedral is about 3 degrees.

The main and auxiliary spars are of T-section and the skin is made up of 4 mm. plywood which is thickened locally on the upper surfaces between the main and auxiliary spars to 5 mm. The skin is also stiffened by longitudinal stringers. A space between the main and auxiliary spar in each mainplane is used as a fuel tank. Servicing is done "by means of special equipment and a mirror after the removal of the end caps."

The wing is connected to the fuselage by four vertical bolts; there are three additional connections on the upper surface for the power unit. The auxiliary spar carries two fittings for each aileron and for the landing flaps.

Ailerons

Wooden construction is used for the ailerons which are dynamically and statically balanced and have a range of movement of 18 degrees upwards and 18 degrees downwards.

Landing Flaps

The two landing flaps are connected by a shaft and are lowered hydraulically. Structurally they are similar to the ailerons. A mechanical stop prevents their being lowered more than 45 degrees.

Fuselage

Duralumin formers and skin are used in the construction of the fuselage. The fittings are partly of steel and the inspection covers of dural or wood.

In section the fuselage is pear-shaped and is constructed in the following assemblies: (1) Nose cap; (2) Forward fuselage, port section; (3) Forward fuselage, starboard section; (4) Fuselage bottom section; (5) Fuselage, central section; (6) Fuselage, rear section; (7) Cockpit enclosure.

The main functional compartments aft of the fuselage cap are the cockpit, armament compartment, tank compartment and undercarriage compartment.

The tail cone, which carries the empennage, is attached to the fuselage in such a way that it may be adjusted to vary tailplane incidence.

Tail Unit

The cantilever tailplane has a dihedral of 14 degrees and is of duralumin construction with steel fittings. The elevator is likewise of metal construction. A range of tailplane adjustment of +3 degrees to -2 degrees is possible.

The wooden fins are rectangular and are attached to the ends of the tailplane by three bolts. Each rudder is carried on three bearings, the center one being fixed and the two outer ones adjustable. The rudders are fully mass-balanced and their movement is limited by fixed stops to 25 degrees on either side.

Controls

The following materials are used in the control system:

Tubes	Dural and steel
Shafts	Dural and steel
Levers	Elektron, Hydromalium and steel.

It is stated that despite the use of plain bearings the controls are very light. The elevator and ailerons are actuated by push/pull rods and the rudder by cable and rods. Tailplane trim is mechanical by means of a wheel on the port side of the cockpit. Twenty turns give a five degree movement.

Undercarriage

The tricycle undercarriage retracts rearwards into the fuselage by hydraulic pressure and is lowered by means of a spring which is compressed during

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retraction. It is locked in the "Up" position by a bolt which is released by a cable, and in the "Down" position by a toggle which passes over dead-center. Undercarriage position is shown by a mechanical indicator in the cockpit and the retracted nose wheel may be seen through a special window.

The oleo legs are of the Me-109 type and the wheel dimensions are 660 x 190. The brakes are foot-operated from the rudder pedals.

When stowed in the fuselage the undercarriage is covered by doors which open and shut automatically. The size of the nose wheel is given as 380 x 150, but it is stated that there is a space for a larger wheel if necessary.

Hydraulic Installation

The hydraulic system is used to retract the undercarriage, lower the flaps, and operate the wheel brakes.

The undercarriage retracting system comprises hydraulic header tank, oil filter, hydraulic pump, control switch and pressure relief valve. Oil is circulated by an engine-driven pump with a capacity of 2.6 gallons per minute at 3,500 r.p.m.

The brake installation is not connected to the main hydraulic system; brake fluid is stored in small tanks on the pedal-operated cylinders.

Power Plant

Although variants of the 162 may be fitted with the Jumo 004 or Heinkel-Hirth 011 turbo-jet unit, the version described in the Heinkel publication is fitted with a BMW 003 E-1 or E-2. Another document mentions a BMW 003 A-1.

The BMW 003 E-1 runs on J-2 fuel and has a sea-level static thrust of 1,760 lb. at 9,500 r.p.m. At 36,000 ft. and a speed of 500 m.p.h. the thrust is 585 lb. The specific fuel consumption is high, 1.61 lb. per hour per lb. thrust. The weight of the unit complete is probably about 1480 lb. and the overall dimensions, scaled from a drawing, are:- length 11ft.10in.; height 2ft.10in.; width 2ft.4in., Ignition is by means of two sparking plugs.

The jet unit is fitted above the fuselage on the upper surface of the wing. At the forward end it is attached by two vertical bolts and at the rear by one horizontal bolt. The front and rear cowlings are fixed to the power unit and are delivered with it; the center cowling consists of two large flaps which may be opened sideways and are normally held shut by a quick-release fastening. If necessary, these flaps may be removed entirely. There is a detachable fillet between the jet unit and the wing.

With special lifting tackle a quick power plant change is possible by virtue of the fact that the engine can be suspended from two points and that all joins in the pipelines and leads are close together. It is inferred that there is a special method of separating the jet unit control rods.

Power Plant Operation

The following controls are fitted:-

Throttle lever with switch for ignition)
 and fuel injection)On port
Clamp for throttle lever)fuselage
Lever for fuel cock)wall
Main electrical cut-out
Starter switch
Main switch for jet unit
Switch for tank pump

Starting and Ignition

The jet unit is started by a Riedel two-stroke starter engine which is in turn, started electrically by a switch on the starboard side of the cockpit. The ignition is switched on when a button on the throttle lever is pressed.

Fuel Tankage.

The normal fuel supply is from a single flexible fuselage tank mounted well forward. This tank has a capacity of 125 gallons for normal flight, with an additional 28 gallons for warming up, take-off and initial acceleration. For the maximum fuel condition, the capacity of the fuselage tank is increased to 168 gallons (including the 28 gallon allowance) and an additional 40 gallons can be carried in the fuel-tight wing compartment.

Both the main tank and the specially prepared wing compartment are filled from a single point on the upper surface of the wing. Fuel is delivered to the jet unit from the fuselage tank, into which the wing tank feeds, by an electric immersion pump via a fuel cock and filter.

Cockpit

Entry and exit are through the roof which opens rearwards and can be jettisoned. The pilot's seat can be catapulted out of the cockpit in an emergency by means of an explosive cartridge. A parachute is attached to the seat, which is adjustable on the ground for the pilot's height.

On the port side of the cockpit are controls for the power unit and on the starboard side are switches for the electrical installation and radio.

The following engine instruments are fitted: revolution counter; fuel pressure gage; oil pressure gage; exhaust temperature indicator; thrust indicator; fuel contents gage. Flight instruments comprise a fine-coarse altimeter; ASI; turn indicator, and pitot head heating indicator. A type FK 38 magnetic compass is fitted. A signal pistol with shortened barrel fires through an opening in the starboard fuselage wall.

Armor Plate

The pilot is protected by an armor plate forward of the cockpit and above the instrument panel. "Movable shutters" are also mentioned.

Harness and Oxygen Equipment

The pilot's seat is provided with Sutton harness and the special seat-type parachute has emergency oxygen equipment in the seat cavity. The main oxygen equipment is carried on the port side of the fuselage, with the indicator and pressure gage in front of the pilot. The oxygen is supplied from a two-liter bottle. The shut-off valve may be operated in flight.

Armament

Alternative armament schemes are 2 x MK 108 or 2 x MG 151/20. The MK 108 is a low-velocity 30 mm. gun and the MG 151/20, a high-velocity 20 mm. weapon. The guns are mounted low in the forward part of the fuselage, one on each side of the cockpit, and the mountings are sufficiently far back to accommodate the barrels of the MG 151/20's in the fuselage. If MK 108's are fitted special blast tubes are attached to the muzzles.

An ammunition box above the guns holds 2 x 120 rounds for the MG 151/20's or 2 x 50 rounds for the MK 108's. The empty cases are ejected into the airstream.

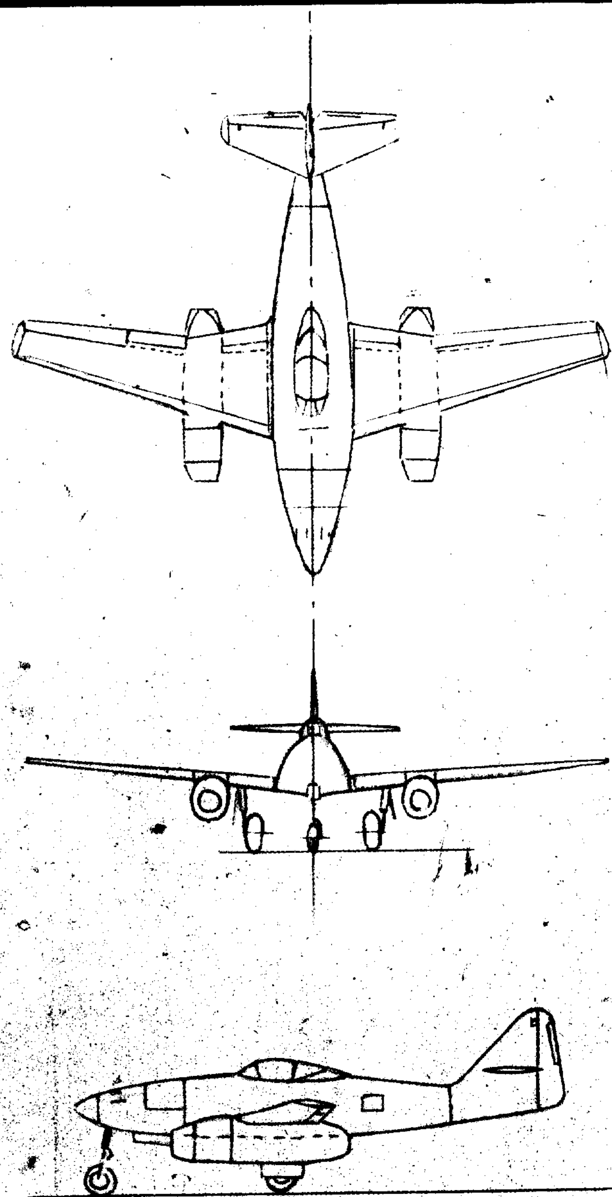
The gun mountings are accessible for installation and removal of the guns through large doors in the sides of the fuselage.

The standard gunsight to be fitted is the Revi 16 G but a Revi 16 B may be used if this is not available. The gunsight is mounted behind the windscreen, immediately in front of the pilot.

Radio Equipment

The radio equipment consists of the FuG 24 (R/T and homing) and the FuG 25 A (I.F.F.). The receiver is mounted on the right of the cabin and the transmitter in the rear part of the fuselage. Two separate aerial matching units are provided for transmitter and receiver, these being mounted in the port and starboard fins respectively. The homing loop is fitted on the cowling of the power unit.

The entire FuG 25 A installation is fitted in the rear fuselage, with the exception of the control box which is near the FuG 24 receiver. The FuG 25 A aerial is built into the port fin.



ME 262

ME-262 JET PROPELLED FIGHTER/BOMBER.

Dimensions and Weight:

Gross Weight	
Fighter	14,740 lbs.
Bomber	15,620 lbs.
Length	34.8 ft.
Height	9.0 ft.
Span	41.5 ft.
Wing Area	233.5 $ft.^2$

Performance (approx.)

V_{max} level	530 MPH
Landing Speed	100 MPH
Rate of Climb, Max.	4200 ft/min.

Fuselage

The fuselage has an overall length of 34 ft.9in. and for a single-seater aircraft is exceptionally roomy; the large stowage capacity has been dictated by the heavy fuel requirements of the two jet-propulsion units.

Construction is semi-monocoque and steel is employed in the fabrication of the pointed nose portion including the thin outer skin. The rear portion of the fuselage is made from duralumin.

The fuselage is of substantially triangular cross section, blending to circular at the nose.

The armament, which is described in another section of this report, is grouped in the nose. The installation is very clean aerodynamically.

The space between the armament and the pilot's cockpit is occupied by fuel tanks. The cockpit has a "tear-drop" enclosure, the center portion being hinged at one side and secured in the closed position by lever-operated bolts.

In emergency, the whole of the cockpit cover can be jettisoned mechanically. Beneath the pilot and immediately aft of the main spar are the wells which receive the main wheels. An auxiliary fuel tank can be fitted under the pilot's feet.

Behind the cockpit there are additional fuel tanks, followed by the radio compartment. Another auxiliary fuel tank can be mounted in the rear portion of the fuselage

The tail end of the fuselage is very slim, the maximum width at a point 18 inches forward of the tailplane leading-edge being only 2 ft. 1 in.

Wing

The thin single-spar wing, has a span of 41 ft. There is pronounced sweepback on the leading-edges and the trailing-edges are also swept back slightly outboard of the jet-propulsion units. Inboard of the propulsion units, however, the trailing-edges are swept forward.

The single spar is of composite I-section, the top and bottom caps being of steel with a built-up duralumin web. At the root end the spar is 1ft.2in. deep and there is a bolted joint on the center line of the fuselage.

Automatic slots extend along the full length of the leading-edges both inboard and outboard of the propulsion units. They are of steel construction throughout and are mounted on built-up steel brackets.

Each aileron is made in two sections, each 3ft.6in. long, and extending out to the wing tip unit. The two sections are connected to a common lever centrally disposed between them. A trimming tab is fitted to the inboard section.

Flaps of modified Handley-Page type are fitted inboard and outboard of the propulsion units. Rollers at both ends of each flap section operate in curved guides. The flaps are actuated by toggle levers which cause them to move bodily rearwards and also tilt downwards. The upper surface of the wing extends over part of the flap chord so that even when the flap is fully extended the leading-edge is still shrouded.

The maximum rearward movement of the flap is about 5½ inches and when it is extended a slot effect is produced. Graduations at 0, 10, 20, 30, 40 and 50 degrees are marked on the upper surface of the flap, the 20-degree position, which is used for take-off, being indicated in red.

Tail Unit

For ease of construction the fin is made in two halves, being divided on the vertical plane of the fuselage axis. The halves are bolted together. Holes are provided in the outer skin to afford access for this purpose and are afterwards covered by doped fabric discs. The joint along the leading edge of the fin is covered by a plywood fairing.

The narrow-chord rudder has an overall height of 6ft. 11 in. It extends from the top of the fin to the bottom of the fuselage. A mass balance is provided near the top of the rudder and there is a large trimming tab which is also mass-balanced.

The 12 ft.4 in. span tailplane is set fairly high and has pronounced sweep-back on the leading edge. Tailplane incidence can be varied by an electric jack which is connected to the mid-point of the leading edge, and causes the whole unit to tilt about the axis of the single spar.

Like the fin, the tailplane is made in two halves which are bolted together, but no attempt to mask the joint along the leading edge is made.

Narrow-chord elevators are shrouded into the trailing-edge of the tailplane and have large mass balances towards the outer ends. Controllable trimming tabs, also with mass balances are fitted.

Undercarriage

The undercarriage is of the tricycle type, the main wheels being retracted inwards into the bottom of the fuselage just behind the main spar. The oleo-pneumatic legs are hinged to the spar at points in line with the outer ends of the inner flap sections. The nose wheel when retracted is accommodated in the

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vertical position in the space beneath the armament installation. Hydraulic retraction is employed for the nose wheel in addition to main wheels. Hydraulic brakes are fitted to all wheels, the nose wheel brake being hand-operated and the main wheel brakes foot-operated.

The tire sizes are as follows:-

Main wheels 840 x 300
Nose wheel 660 x 160

Jet-Propulsion Units

The Me-262 is powered by two Jumo 004 turbo-jet propulsion units. The sea-level static thrust of this unit is about 1800 lb. It is of the axial-flow type with an 8-stage compressor at the leading end. With this design it has been possible to keep the maximum overall diameter of the unit down to 32 in. The length measured from the intake to the exhaust outlet is 11 ft. 8 in. and the extreme length over the cowlings appears to be about 12 ft. 6 in. The estimated dry weight of each unit is 1500 lb.

The units are slung low beneath the wing so that there is no interruption of the main spar. On either side of each unit there is a built-up steel rib or bulkhead, the whole being enclosed by the cowling.

Some of the auxiliaries are mounted on top of the propulsion unit at the forward end and the cowling enclosing them produces a slight bulge above the leading edge.

At the rear end the full circle of the cowling is below the trailing edge and blending is effected by a shaped portion which sweeps upwards to join the wing surface.

For starting, the turbines are run up by means of small 2-stroke engines mounted at the front of the rotor assemblies. A small quantity of gasoline is provided for the initial firing of the main units and as soon as this has been expended the main supply of diesel fuel oil is employed.

Fuel Tankage

There are two main self-sealing tanks, each of

900 liters (198 gallons) capacity, one mounted in front of the cockpit and one behind.

A rigid light-alloy tank in the after part of the fuselage accommodates a further 600 liters (132 gallons) but is normally filled to only two-thirds of its capacity. A 200-liter (44 gallon) auxiliary tank can be provided under the pilot's feet at the expense of part of the armament.

Armament

The full designed armament is 4 x MK 108 guns of 30 mm. calibre. These guns are grouped in the nose of the fuselage and as their overall length is only 3ft. 6 in. a very compact installation has been achieved with no external projections.

There is a large spherical support round the barrel near the rear end which facilitates adjustment when harmonizing the installation.

The full armament is formidable. The MK 108 gun, which weighs 134 lb., has a rate of fire of 575 to 600 r.p.m. with a muzzle velocity of 1570 feet per second.

The capacity of the ammunition containers is at least 75 r.p.g. Possibly the number of rounds per gun is increased when only two guns are fitted.

Compressed air for cocking and sear release is contained in eight bottles which are housed in the lower portion of the fuselage under the forward petrol tanks.

Armor

The only armor comprises two pieces of 15 mm. These together form a bulkhead 2ft.10in. wide by 1ft.7in. deep, which is fitted behind the pilot in the upper part of the fuselage The windscreen is of bullet-resisting glass, 10 cm. thick.

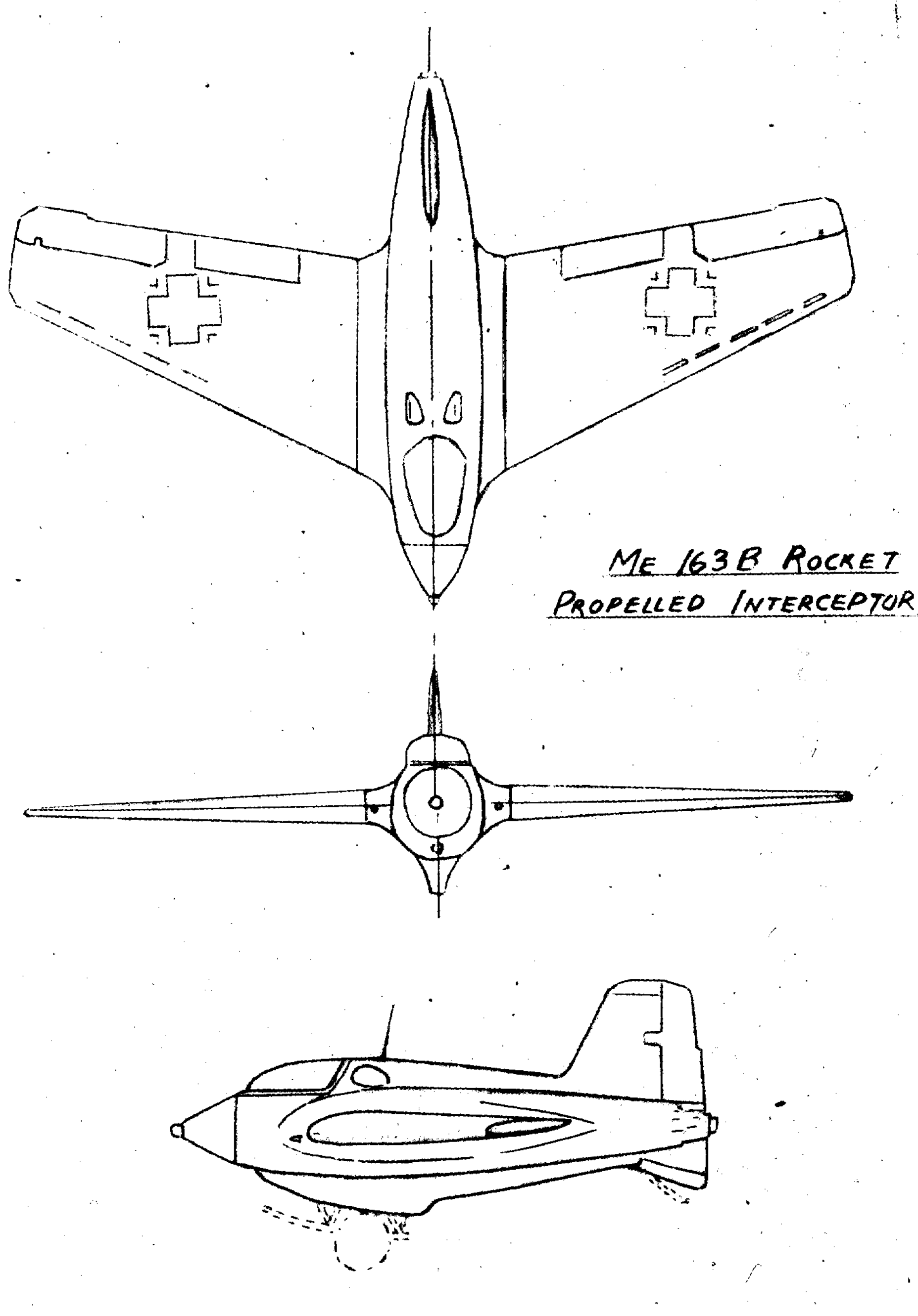
Bomb Carriers

Bomb carriers may be fitted on one or both sides of the fuselage forward of the wheel wells.

If there is only one carrier it is mounted on the port side. The carrier measures 2ft.9in. by 2⅞in. by 4½in. overall and will accommodate one size of bomb only, namely 250 kg. The two fuzing plugs are spaced 1ft.4in. apart from center to center.

Radio

The radio fitted to the Me-262 conforms to the usual equipment for German aircraft fulfilling similar functions. It comprises FuG 16 Z (VHF R/T and D/F) or FuG 16 ZY (VHF R/T, D/F and retransmission facilities for ground control by fighter Benito stations). In certain cases FuG 25 A (I.F.F.) is fitted either for identification or for ground positioning in conjunction with Freya stations.



ME 163B ROCKET PROPELLED INTERCEPTOR

ME-163B - ROCKET POWERED INTERCEPTOR

Dimensions:

Wing Area	186 sq.ft.
Wing Span	30.5 ft.

Weights:

Weight Empty	4190 lbs
Fuel	4410
Ammunition	220
Crew	220
Gross Weight	9040 lbs
Landing Weight	4630 lbs

Performance:

Maximum Level Speed - limited to	590 mph
Endurance at 495 mph	8 min.
Time to climb to 32,800 ft.	3 min.
Rate of Climb - at sea level	11810 ft/min.
at 32800 ft.	33470 ft/min.
Landing Speed	90 + mph.

Description:

This airplane is a tailless interceptor powered by the Walter rocket power plant of 3750 lbs. thrust. The two fuels carried are H_2O_2, the oxygen carrier, and an alcohol base hydrocarbon. The total weight of fuel is 2/3 H_2O_2 and 1/3 hydrocarbon. Combustion is automatic when the two fuels are mixed. Part of the H_2O_2 is used to drive the turbine that powers the centrifugal pumps necessary to drive the fuel into the combustion chamber under the proper pressure. The H_2O_2 goes through a catalyst of MnO_2 crystals which breaks it down into steam and oxygen. The steam then drives the pump turbine.

The airplane, designed by Dr. Lippisch, is the first tailless ship to get into combat, and is also the first airplane to be rocket powered..

The wings have approximately 30° sweepback of the leading edge, fixed leading edge slots outboard of 60% semi-span, and carry elevons, trimming flaps and landing flaps. The trimming flaps are at the trailing edge of the inboard section of the wing and are similar in design and location to ordinary plain flaps. The landing flaps are at a point approximately 50% of chord of the wing and are located forward of the trimming flaps. The airplane is a mid-wing with a root-section of 14% t/c and a tip section of 9% t/c. The root section has reverse camber while the tip is a symmetrical section. There is structural wash-out of 5° and the root section has an angle of incidence of 3°. The wings are wooden of quite conventional construction and tie to short stubs on the fuselage. The hydrocarbon fuel tanks are in the inboard section of the wing panel.

The fuselage is all metal with a basic circular cross-section. The profile is of the equation $y = x\sqrt{1 - x}$. Above this basic section, the cabin and vertical tail are faired in, and below the basic section the landing skid and tail wheel housing are also faired in. A small propeller in the nose drives the generator, aft of this is the pilot's cockpit, then the fuel tank and Walter unit. An MK-108 30 mm rapid fire cannon is located in each wing stub.

The airplane takes off on a pair of droppable wheels of very narrow tread and lands on a skid that is retractable. Directional control on the ground is very poor.

Because of this airplane's very advanced design some of its history of development may be of interest. The following is from an interview with Dr. Lippisch.

In 1932 Lippisch designed the Delta IV, which was modified as the Delta IVa in 1935, and led to the design of DFS 39 in 1936. These airplanes were the forerunners aerodynamically of the Me 163. The DFS 39 had downturned wing-tips to improve the lateral stability. There were rudders on each wing-tip which moved only outward, singly for rudder action and together for use as aerodynamic brakes. Elevons were used for pitch and roll control, together for pitch and differentially for roll. When asked why he did not use spoilers for lateral control, Lippisch said that spoilers on swept-back wings give trim difficulties unless the spoilers are on the center of gravity. Trailing edge flaps also give trim difficulties. Therefore Lippisch in some designs had used flaps ahead of the trailing edge. These are less effective than trailing edge flaps but give less moment change.

The Me-163 originated in a project proposed by Antz of the Air Ministry to put the Walter rocket engine (cold system) in an airplane. It was requested that the engine be used with an airplane whose development was most advanced as regards flying qualities, and the aerodynamic design of the DFS 39 was selected. The triangular fuselage cross section (on Me-163 A) was intended to give the same lift distribution across the fuselage as on the wing.

The order was placed at the end of 1937 with a speed requirement of 300 to 350 km/hr. Wind tunnel tests were made in the large wind tunnel at AVA Göttingen . A central rudder was used. The end plates on the wings or turned-down tips were discarded for fear of wing flutter, since the rudders were back of the elastic axis of the wing. The central rudder gave improved control characteristics, for the aerodynamic coefficients of the end plate rudders changed with lift distribution. The rudder could not be omitted because of the destabilizing effect of the fuselage nose extending outside the wing contour.

Lippisch stated that it was very easy to obtain longitudinal stability in tailless aircraft, but the first and most difficult requirement was to secure directional stability about a vertical axis. To avoid reversal of the stick force at the stall, the wing must be designed to stall at the center first. To achieve this fixed slots were used at the wing tips, first adjustable, and later fixed. A special form of fixed slot was developed by Lippisch which had low drag in the normal flight condition. It was designed to have equal pressures at the two ends of the slot and hence no flow through the slot in the normal flight attitude. This section with slot gave a smooth polar curve. For aspect ratio 5, the lift was maintained to an angle of attack of 30° and the maximum lift coefficient was 2.2. The airplane with slot length equal to 40 percent of the semispan could not be spun. With crossed controls, the airplane merely sideslipped. The outer half of the slot was then closed, leaving the slot open from the 60 per cent semispan to the 80 percent semispan position measured from the center. The airplane could then be spun in a perfectly controlled manner.

At first the fuselage was to be built by Heinkel since they had already built the He-176 with the Walter engine, and the wings were to be built at Lippisch's own factory. The He-176 was unsuccessful and could not take off from the ground with the available thrust.

The Me-163A weighed 2300 Kg, had a wing area of 17.5 m^2 and an available thrust of 750 Kg.

Lippisch contends that high speed aircraft must be designed for a low lift coefficient, and hence must use a low wing loading; otherwise the drag is excessive.

The divided responsibility between Heinkel and Lippisch was unsatisfactory and Lippisch decided to make the whole machine. The lofting was ready when Lippisch moved with 20 collaborators from the DFS to Messerschmitt, becoming section L of that firm.

The rocket motor was tested in the DFS 194 in 1940. The motor then agailable ("Cold" Walter motor) gave a thrust of 300 Kg. Flights were made at speeds up to 550 Km/hr.

The Me-163A was ready in the spring of 1941 and was tested by towing at Augsburg to measure flying qualities. Flights were made at 4000 and 8000 m both at normal speeds and in dives to speeds of 850 km/hr. There was at first some trouble with rudder flutter because the rudder was not quite mass balanced. The airplane was taken to Peenemünde in September 1941 to have the motor installed. The motor turned out to be somewhat unreliable. The jets became clogged from particles of the calcium permanganate catalyzer. Safety devices have to be provided in the combustion chamber because the motor will explode in the absence of catalyzer. Many serious accidents occurred in the development. At Kummersdorf a whole building and all its occupants were destroyed. Another fatal accident occurred at Busemann's rocket station at Trauen Flugzeugprüfstelle. If an insufficient amount of catalyst is supplied, the pressure and thrust fluctuate violently. The motor could be throttled from full thrust to 10 percent of full thrust. The same type of turbine system and centrifugal pumps as was later used on V-2 was used in this early rocket motor. The weight of fuel was 1200 Kg, and at the full thrust of 750 Kg, the fuel was consumed at the rate of 9 Kg/sec. Flight tests were made in September at speeds up to 810 Km/Hr., then 930 Km/hr, and in October 1941 to 1003 km/hr. The altitude was 3600 m and the speeds were determined from theodolite observations at six stations.

The thickness of the wing was 14 percent at the center, and 9 percent at the tip. The thickness distribution was an NACA one but the center line shape was Lippisch's own development to give stable wing sections, the trailing edge being turned up. The sections used were of relatively small camber. In the earlier design (300 Km/hr) a washout of 6° had been used but in the Me-163A stability was secured by the use of stable airfoil sections.

At 1003 Km/hr at the test altitude the Mach number was 0.85 and compression shock occurred at the wing tip. The lift actually became negative at the tip and caused a sudden tendency to dive. The tendency was less at high altitude.

Lippisch then got an order to design an interceptor fighter using the "hot" Walter motor and the same wing as the Me-163A. The fuselage was to be larger to carry more fuel and equipment. A 15 minute duration was desired. The "cold" Walter engine uses hydrogen peroxide and calcium permanganate as catalyser. The products of combustion are steam and oxygen and the temperature is low, 500 to 600°C. Byadding gasoline to combine with the oxygen, the thrust and also the temperature are greatly increased. Three substances must be fed into the combustion chamber, hydrogen peroxide called T-stoff, catalyzer called Z-stoff, and a hydrocarbon. Another development was a BMW fuel system, using nitric acid as the oxydyzer, called Salbei. Still another development by Lutz was a catalyzer, hydrazine hydrate, which could be mixed with the hydrocarbon in the "hot" Walter engine to reduce the number of components from three to two. One combustion chamber would stand about 20 runs. Fuel was used for cooling. The total life was about one hour for intermittent operation, with full load operation for 5 or 6 minutes at a time.

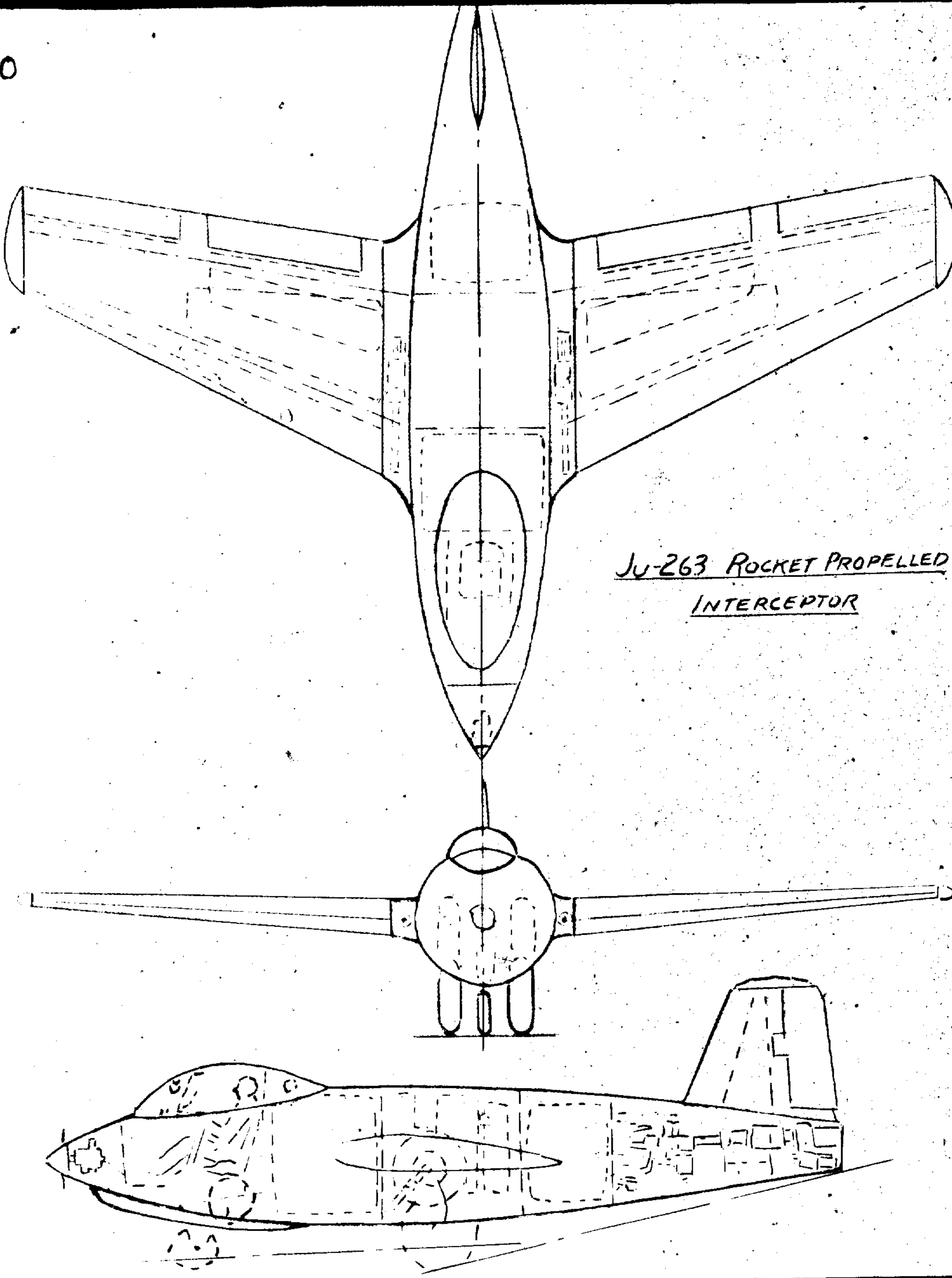
In 1942 the airframe of Me-163B was tested in towed flight without the engine. The engine was due in the spring of 1942 but did not arrive until the summer of 1943. The method of throttling was not perfected by Walter until then. A Dr. Schmidt (neither the Schmidt of Braunschweig or Munich) developed the motor for Walter.

Lippisch described the early development of hydrogen peroxide as a rocket fuel by Walter. Walter was an engineer in the Reichsmarineamt. His first use of hydrogen peroxide was in life-saving apparatus, i.e., guns firing life lines to ships. He later applied it to the propulsion of torpedoes, this use being attractive because the chief product of decomposition is steam which is absorbed by water and hence the wake is invisible. The original invention was about or even earlier than 1934. The process was applied at the DVL to improve the rate of climb of a small biplane and later for assisted takeoff. The Me-163 was the first airplane using the Walter motor to fly.

The weight of Me-163B was intended to be 3300 Kg but it actually came out 4100 Kg. It was therefore necessary to use an auxiliary rocket for assisted take-off. A solid-propellant rocket was used, first of 500 Kg thrust, later two with a total of 1000 Kg thrust. To provide for control if one launching rocket failed, rudders were used working in the boundary layer of the exhaust stream. The rudders were of steel. The carbon rudder within the jet was developed at Peenemünde specifically for V-2. Several flights were made, first in Bremen, then in Augsburg, then at Brandes near Leipzig. The landing speed was about 160 Km/hr.

Only 10 machines of the A type were built. They were found to be very useful as trainers. The flying qualities were good, the response being quick because of the small moment of inertia. The interceptor (B-type) with engine was flown first in the summer of 1943, and production began in 1944. Some 60 of the V-Muster series were built at Regensberg. Then Klemm had the contract and ran into much trouble with fairing of the wing and fuselage because of their inexperience. Then all development was transferred to Junkers.

From the end of 1944 to the end of the war Junkers produced about 300 type Me-163B airplanes. They also developed a new model (Me-263) with larger fuselage and with retractable landing gear. The Me-163B had jettisonable wheels for take off and landed on skids.



JU-263 ROCKET PROPELLED INTERCEPTOR

JU-263 - ROCKET POWERED INTERCEPTOR.

Dimensions:

Wing Area	192.6 ft^2
Wing Span	31.2 ft^2

Performance:

Max. Level Speed -limited to	590 mph
Endurance at 495 mph	15 min.
Time to climb to 32,800 ft	3 min.
Landing Speed	90+ mph

Weights:

Wing	1100 lbs
Fuselage	992
Landing Gear	463
Controls	88
TOTAL Structure Weight	2643 lbs
Equipment (Guns, Electr., Oxyg., Hydr.)	882
Power Plant Install	440
Fuel Tanks	440
Weight Empty	4405 lbs.
Fuel	
H_2O_2	4410
Hydrocarbon	2205
Ammunition	445
Pilot & Chute	220
GROSS WEIGHT	11685 lbs.

Description:

This airplane is an outgrowth of the Me-163B and incorporates improvements found necessary to make a rocket powered interceptor effective as a weapon. The major effort was directed toward increasing the range and improving the ground handling characteristics of the original Me-163B.

The wing panels were structurally and aerodynamically the same as the 163, the only change being additional tank space for the added hydrocarbon fuel. The control surfaces and operating system are also the same as the 163.

The fuselage is entirely redesigned and is approximately 3 feet longer than the 163. It is slightly bigger in diameter than the 163 and has slightly larger wing stubs, thus increasing the wing span approximately 8 inches. A tear-drop canopy is used for improved vision. The cabin is pressurized by air taken in at a point just below the nose and the air is compressed by the nose propeller driving a compressor. The cabin maintains 8 km. altitude pressure up to 15 km., the maximum pressurization being .3 Kg/cm^2 (4.3 lbs/in.^{2}). There is also a connection from the cabin to the fuselage fuel tanks (H_2O_2) so that the tanks are also pressurized. The nose air intake is faired to a point far back on the fuselage and this fairing helps cover the nose wheel when it retracts into the fuselage. (Retracted position is underneath the pilot's seat).

The main wheels also tie to the fuselage and retract into it. The tread is very small, being less than the diameter of the fuselage. Although the incorporation of a retractable tricycle gear was expected to greatly improve the ground handling of the airplane compared to that of the 163, the narrow tread was considered unsatisfactory enough so that the wing tip skids of the 163 were still left on the 263. The landing gear is retracted hydraulically.

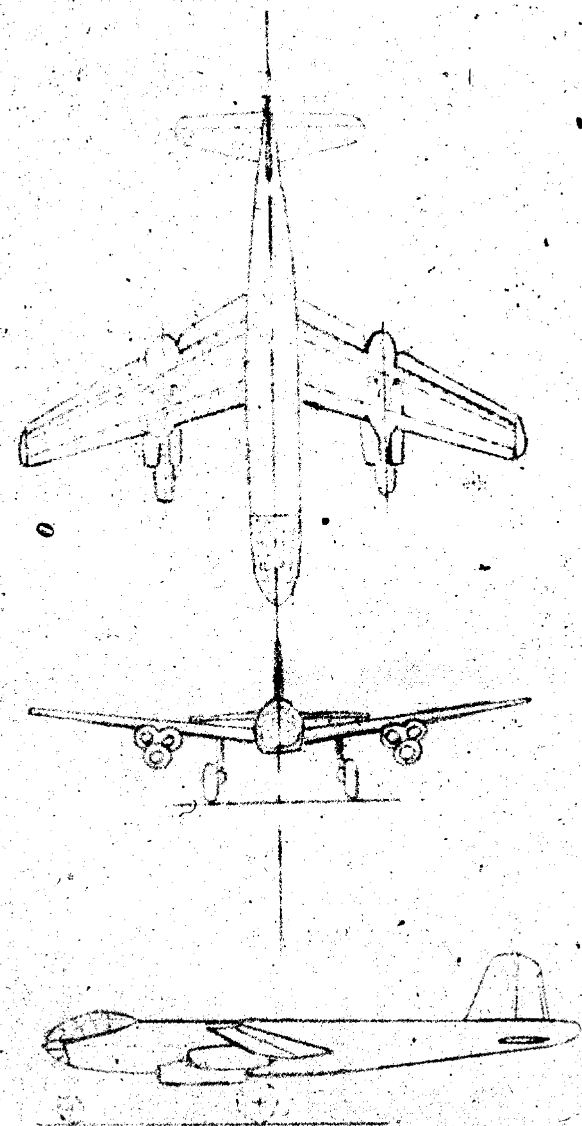
A mesh type parachute of 11.5 ft. diameter is kept in a compartment in the aft fuselage and is released after the airplane lands to shorten its run.

Fifty percent more fuel is carried in the 263 than in the 163. The hydrocarbon is carried in the wings and the H_2O_2 is carried in three tanks in the fuselage - one large tank just behind the pilot, a small tank over the landing gear well, and another large tank aft of the well. All three tanks were interconnected, but the large fore and aft tanks had a regulator between them that kept equal volumes of fuel in these two tanks. This was necessary because otherwise the high angle of climb of the 263 (40° approx.) would cause the forward tank to empty and the rear tank to be full with a consequent large aft movement of the airplane center of gravity.

The Walter rocket unit used in the 263 incorporated a separate "cruising" rocket chamber that produced 885 lbs. thrust. In addition to this, the output of the main rocket chamber was increased to 4410 lbs. thrust. compared to 3750 lbs. for the one used on the 163. The auxiliary cruising jet was used because it attained a higher efficiency than the large unit could when operating at partial capacity, as was done on the 163 for cruising. (The fuel consumption when operating at full capacity was 5.5 gm/kg.sec. while at part capacity was 10 gm/kg.sec.)

This cruising jet arrangement, with the help of 50% more fuel, raised the flight duration of the 263 to 15 minutes compared to 8 minutes for the 163.

44.



JU 287 JET BOMBER.

JU-287 HIGH SPEED JET BOMBER

The Ju-287 is of special interest because of its swept forward wings and the novel arrangement of its six jet motors. The airplane is an indication of Germany's thoughts on bombers - high speed, comparatively short range. A flying mock-up of this ship has flown, but the first actual airplane was not yet complete when the war ended. The mock-up had a Do-217 fuselage, the correct wings, a non-retractable tricycle landing gear and tried many locations of the six power plants.

The Ju-287 was designed as a production airplane.

Dimensions:

Wing Area	602.5 ft^2
Span	63.7 ft.
Length	64.3 ft.
Wheelbase	24.98 ft.
Tread	18.37 ft.

Weights:

Empty Weight	29,500 lbs.
Fuel	15,850
Bombs	4,400
Ammunition	220
Crew	660
Total Gross Weight	50,630 lbs.
Landing Weight	34,200 lbs.

Performance:

High Speed - Sea Level	518 mph.
at 19,700 ft.	541 "
Rate of Climb - Sea Level	4730 ft/min.
at 19,700 ft.	2950 ft/min.
Service Ceiling	36,000 ft.
Calculated Range	1100 miles
Cruising at	496 mph.

Take-off Run (With rocket assist. of 3 x 2200 lbs. x 12 secs.)	3940 ft.
Landing Speed	109 mph.

Description:

The Ju-287 is an all-metal low wing fast bomber. The wings are swept forward approximately 20°, have the Junkers reverse camber high speed airfoil of 12% maximum thickness, and have fixed leading edge slots between the nacelles and fuselage. Three BMW 003 A-1 jet units are carried in a triangular cluster under each wing, the lower power plant extending considerably forward of the upper two in each cluster.

The wing is of conventional two-spar construction with fairly close former spacing and only occasional full ribs. There are no spanwise stringers. The outer panel spar caps are each one piece forgings and include the splice fittings, which are of the German ball and socket type. Only the upper fittings transfer shear.

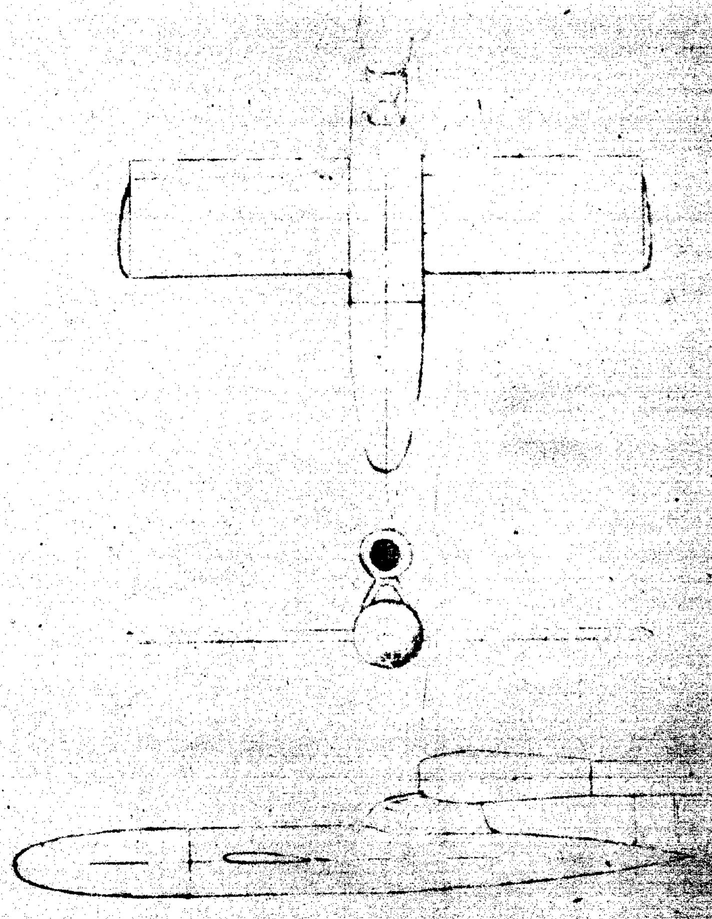
The fuselage includes the pressure cabin, which houses the 3 man crew, the bomb bay, fuel tanks, and the remotely controlled rear gun turret. The bomb bay is forward of the wing structure, but is still on the center of gravity because of the swept forward wing. The fuel tanks start aft of the cabin and run to a point aft of the wing. The tail surfaces are conventional.

The plane has a tricycle landing gear with the nose wheel retracting aft. The main wheel struts are mounted in the wing and fold inboard. The wheel axles rotate 90° with respect to the oleos (when viewed from the front) and the wheels then fit into the fuselage in a vertical position.

The armament is the Heckstand 131Z and the ship reputedly has a bomb capacity of 8800 lbs.

The ship was intended to mount four HeS-011 engines, but Junkers was forced to use 6 BMW 003-A-1's because of the slow development of the 011.

The plane was designed for a "safe load factor" of 4g for normal load and 3.2g for overload. (These are multiplied by 1.8 for failure load). The design diving speed was 590 mph. or a Mach number of .8.



V-1 PILOTED BOMB

PILOTED V-I "BOMBER"

Dimensional Data:

Overall length	26' 3"
Wing Span	18' 9"
Length of fuselage	24' 3"
Maximum diameter of fuselage	2' 9"
Length of propulsion unit	12'
Maximum diameter of propulsion unit	1'10"
Diameter of rear of propulsion unit	1' 3"
Length of Wing	8' ½"
Width of wing (without aileron)	3' 6"
Length of aileron	7' 8"
Width of aileron	9 ¾"
Horizontal stabilizer w/control surface	6' 9" by 2'1"
Horizontal stabilizer control surface (each)	2'10" by 9"
Vertical stabilizer w/control surface	3'2" by 1'9"
Vertical stabilizer control surface	1'4" by 1'9"
Cockpit opening length	1'9"
Cockpit opening width	1'5"
Seat to top of canopy	3'1"
Windshield length	11"
Windshield width at top	6"
Windshield width at bottom	9"
Length from nose to headrest	14'
Length of fuel cell	4' 6"
Length of war head	6'
Length from nose to wing	7' 1"
Length of wing spar	14' 4"
Diameter of wing spar	4 3/8"

Technical Data

Piloted V-1 bombs were intended to be launched from bomber type aircraft. No information was obtained which indicated that piloted V-1 bombs were to be launched from ramps.

The piloted V-1 bomb was to be flown to and directed into the target by the pilot, and a parachute jump was to be made just before collision with the target.

Piloted V-1 bombs were constructed of steel except for the wings and nose cover, which were plywood. Structural parts of the piloted V-1 bombs are identical to the late models of the V-1 bomb except for the following:

(1) Ailerons added to standard wings.
(2) Cockpit, controls, and flight instruments have been installed just forward of propulsion unit.
(3) Tapered war head with two fuze cells and plywood cover have been substituted for standard ogive shaped war head.
(4) Only one compressed air container is used, and this is located in about the position of the automatic pilot of the standard V-1 bomb.

Instrument panel (arranged left to right).

(1) Arming switch operated by key wired to instrument panel.
(2) Clock.
(3) Airspeed indicator
(4) Altimeter
(5) Combined inclinometer and turn indicator.

A gyro compass was mounted in a shock mounted bracket with a small 24-volt wet battery and 3-phase inverter. This assembly was mounted on the floor between the pilot's knees, so that the gyro compass was just below the instrument panel.

The flight controls are the conventional stick and pivoted crossbar rudder control.

Personal equipment consisted of parachute, life-preserver, helmet with headphones and throat microphones, sunglasses, safety belt, shoulder straps, crashpad above instrument panel, plywood bucketseat, and padded head-rest at the top of seat back.

Performance: Performance of piloted V-1 bombs is expected to be the same as that of the late models of the standard V-1 bombs. The following data were obtained:

(a) Speed at end of launching ramp is 400 km/hr.
(b) Speed of late model V-1 bombs was 900 km/hr. Old models had speed of 800 km/hr.
(c) Time of flight is 40 minutes.

Pilot Survival: It was stated that although the pilots were equipped with parachutes it was expected that 99% would not survive.

The canopy is mounted by two curved prongs, which hinge into sockets on the right side of the cockpit. The left side of the canopy is held by two eyelets and sliding pins. To release the canopy it is necessary to operate the lever in the left side of the cockpit, disengaging the sliding pins from the eyelets. The canopy must swing about 45°, before the front prong on the right side will release. The back edge of the canopy interferes with the cowling of the propulsion unit. It is believed that it would be difficult, if not impossible, to jettison the canopy while in flight. Lines on the windshield, are used to estimate diving angles.

Communication between launching aircraft and the pilot in the V-1 bomb was maintained through a four channel connector at the top of the fuselage in front of the canopy.

Quantity of Piloted V-1 Bombs Produced: Approximately 175 piloted V-1 bombs had been built.

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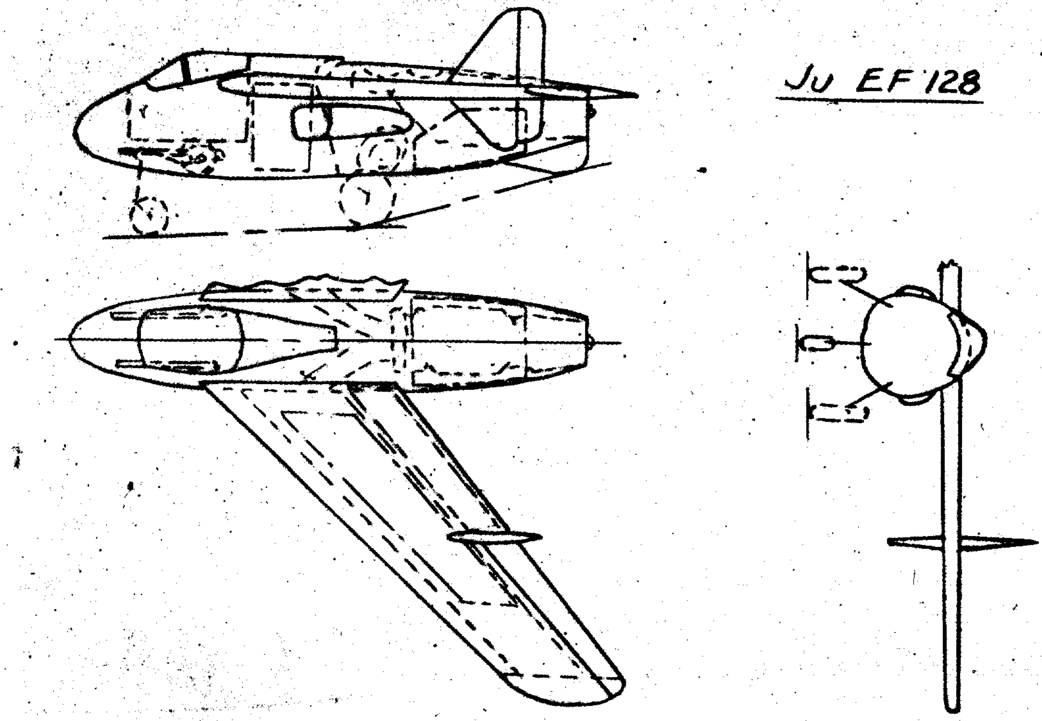
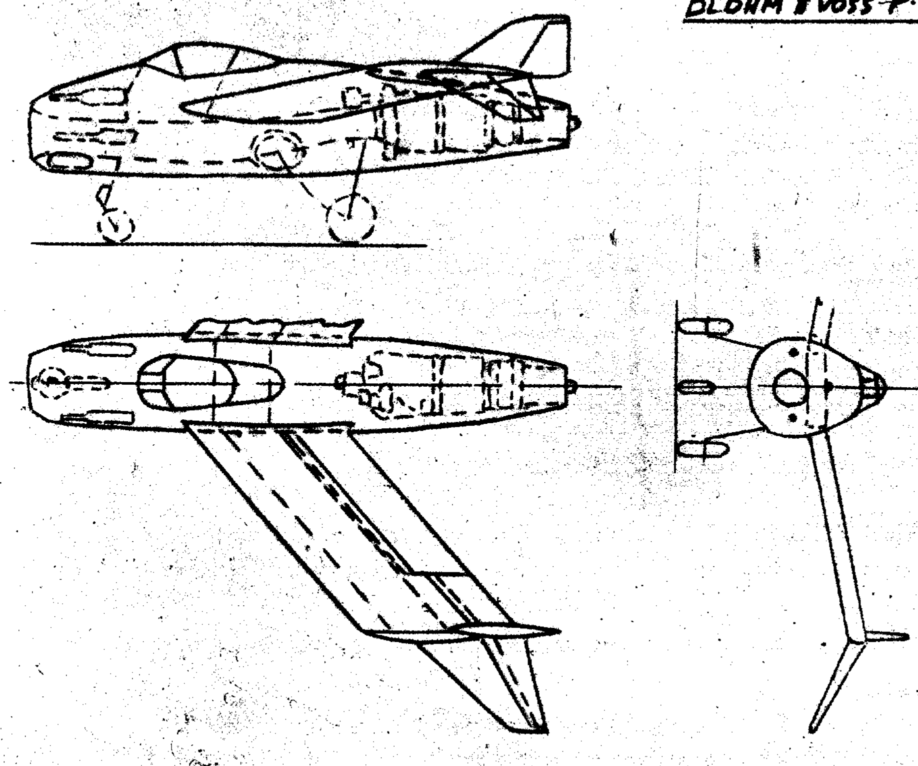
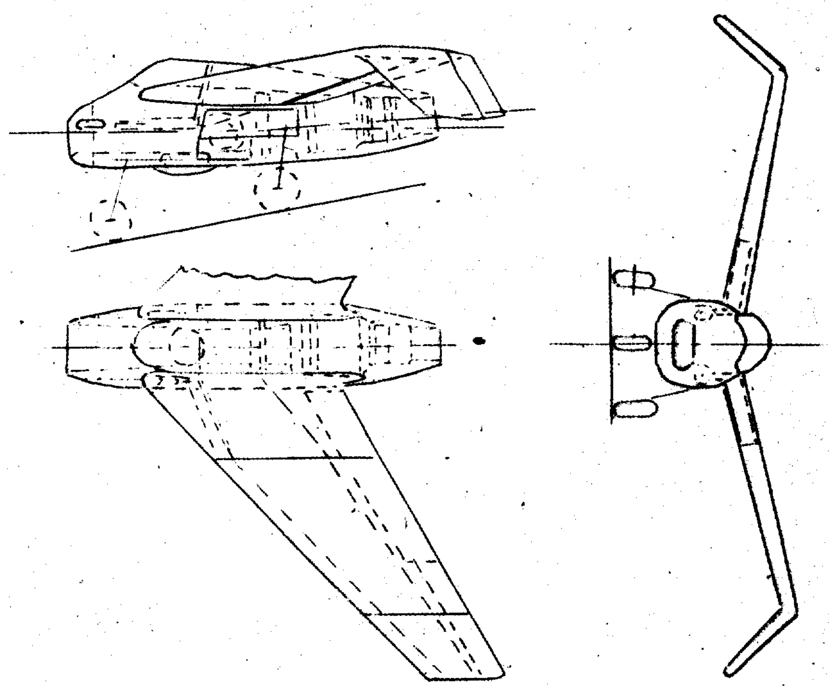
3. EXPERIMENTAL AND PROPOSED AIRPLANE

Ju EF 128



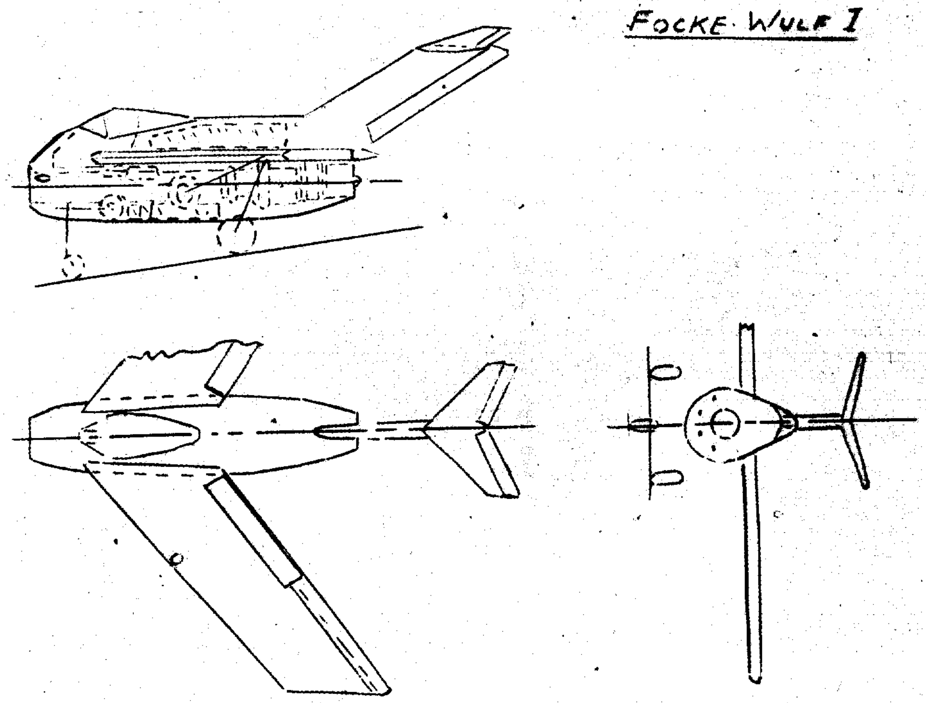
BLOHM & VOSS-P.212

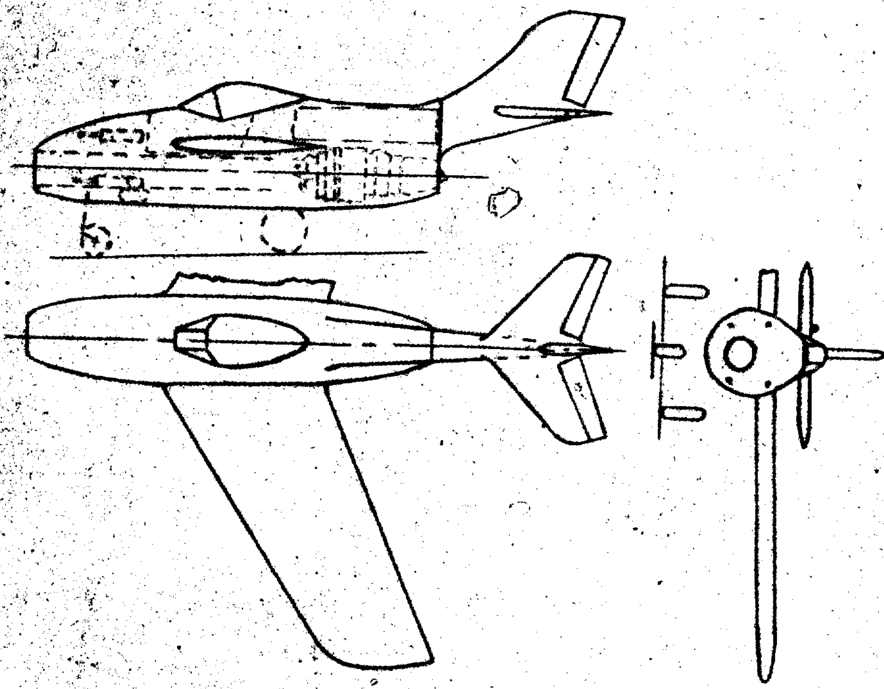




HEINKEL P1078

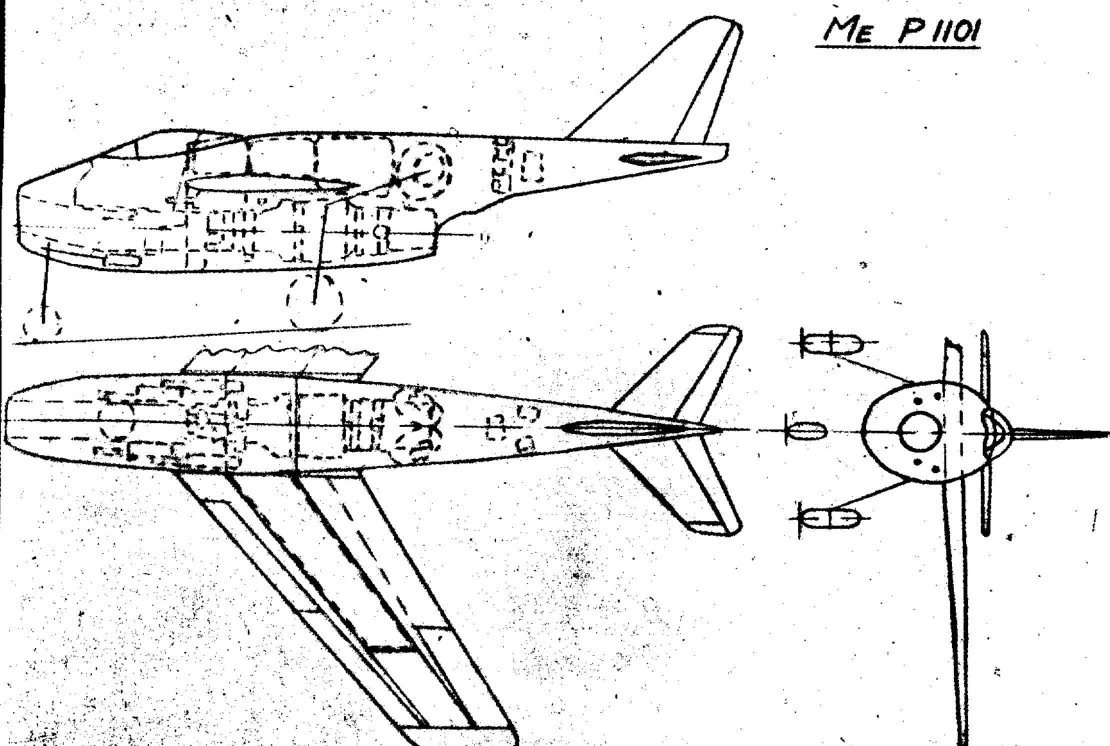
FOCKE-WULF I

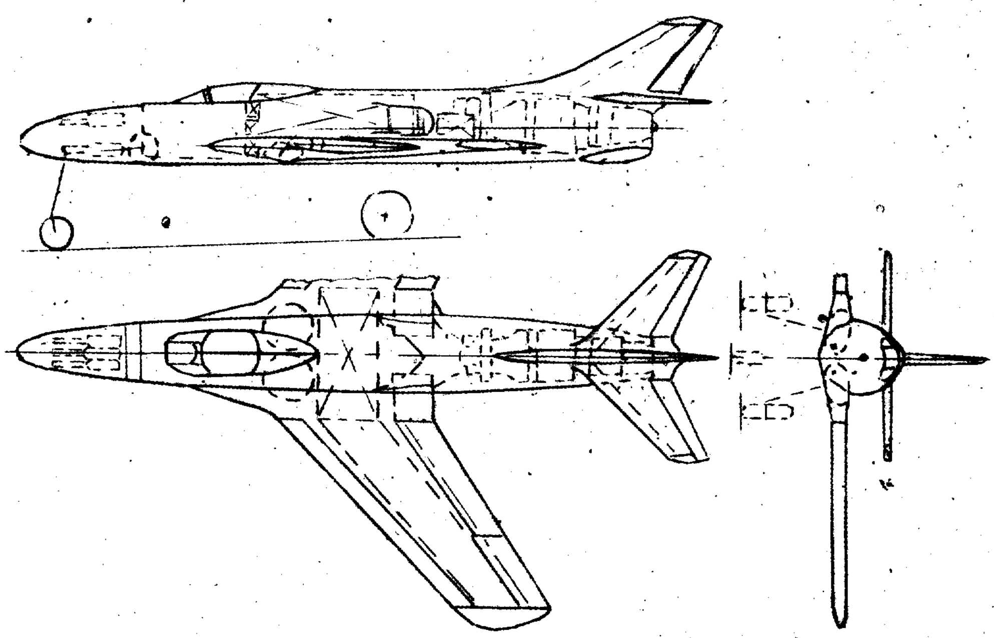




FOCKE-WULF II

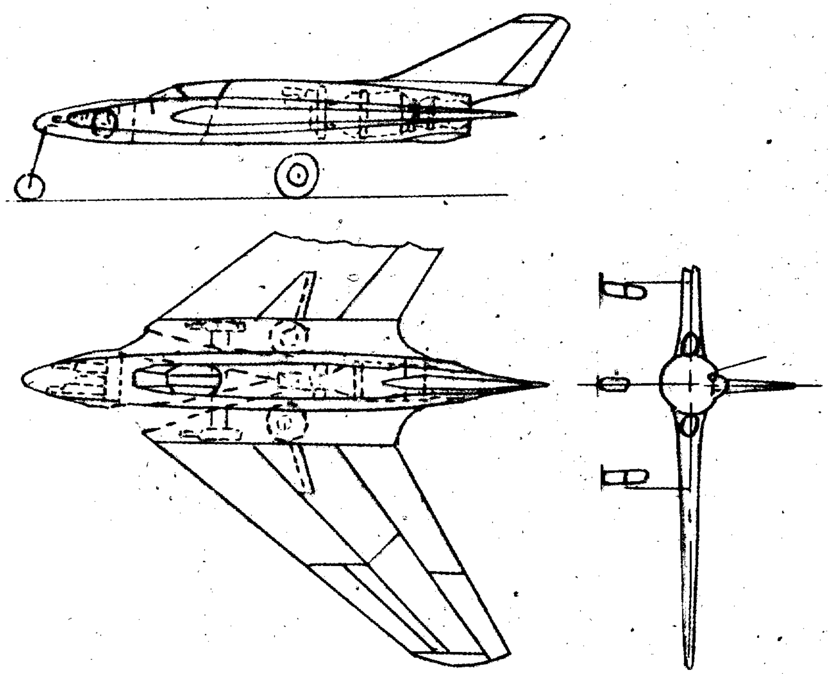
ME P1101





ME P.1110

ME P.1111



SINGLE-JET DAY FIGHTER COMPETITION

The designs shown and described in this section were submitted to the Chief of the Development Commission and head of the Technical Air Equipment Section in a design competition. No decision had been made as to the "winner", but the following companies had entered proposals:

Blohm and Voss
Focke-Wulf
Ernst Heinkel
Junkers
Messerschmitt

The performances are comparative figures for the same method of calculation applied to all the proposals and are not to be considered as absolute estimates of velocities.

1. SHORT DESCRIPTION OF JUNKERS EF-128

General: Tailless, high wing, swept-back wing.

Wings: Two spar all-wood wing, portion of fuel carried in wing.

Fuselage: All metal fuselage. Nose portion contained room for nose wheel, pressure cabin, fuel tank, space for main wheels and air duct above the gear. In the rear of the fuselage, the engine and another fuel tank are housed.

Controls: Elevons; vertical surfaces above and below the wing at the inboard ends of the elevons.

Undercarriage: Nose and main undercarriage are hinged, link type with bending free air oleo legs. Pneumatic retraction main wheels 710 x 185 (28" x 7.3"), nose wheel 465 x 165 (18.3"x6.5").

Engine: HeS 109-011 enclosed in rear of fuselage. Accessible through removable parts of outer shell of fuselage. Intake openings for air on the side of the fuselage under the wings. Boundary layer duct provided. Suction of boundary layer air through vent at rear of pilot's hatch. Fuel installation - 142 gals. (American) unprotected in the wings, 272 gals. in two protected tanks in the fuselage.

Equipment: Standard fighter equipment plus catapult seat and fire extinguishing apparatus.

Armament: 2 x MK 108 30 mm. rapid fire cannon with 100 rounds each in the forward part of the fuselage under the pilot's cabin - additional 2 x MK 108 with 100 rounds each provided for.

Armor: Pilot protected from 12.7 mm. from the front and 20 mm. from behind.

2. SHORT DESCRIPTION OF BLOHM AND VOSS P212

General: Tailless, mid-wing with sweptback wings and controls at the tips of the wings.

Wings: Steel skin wings of the Blohm and Voss type of construction. Part of the fuel carried in the wings. Very deep landing flaps at the trailing edge and nose flaps which slide into the wing leading edge.

Fuselage: The curved steel intake duct is the inside support of the fuselage. Just in front of the turbine inlet, the support pipe widens out to attach to a longitudinal beam each side of which is attached to the wing in front and the turbine behind. The guns are placed in front of the "pressure proof" pilot's cabin, which is situated above the inlet duct. The engine is in the stern.

Stabilizers: At the end of each wing are two surfaces of which one, slanting downwards, fulfils the function of the elevator and partly the functions of the vertical stabilizer and ailerons. The second fin is an additional small vertical stabilizer. In addition, there is a small normal aileron.

Undercarriage: Main undercarriage ties to the fuselage beams and moves forward into the fuselage. Nose wheel also moves forward. Main and nose wheels same as on Ju EF-128.

Engine: The HeS 011 is located in the aft portion of the fuselage. Air is ducted through the nose of the fuselage.

Fuel Installation: 216 gals. unprotected behind the pilot and 40 gals. protected in the center panel beneath this. 126 gals. in protected tanks in front of the engine. Additional tanks in the wing total 105 gals.

Armament: 2 x MK 108 with 100 rounds each next to the air duct in the nose of the fuselage. An additional 108 possible above the duct. Also, additional 2 x MK 108 with 60 rounds each can be installed behind the pilot in the underpart of the fuselage. Armament may be varied - either MG 103 or MK 112. Possible to build a place in fuselage for 1100 lb.bomb.

Armor: Protection from front and rear shots.

3. SHORT DESCRIPTION OF HEINKEL P1078

General: Tailless, mid-wing with swept back wings and controls at the end of the wings.

Wings: Contain full fuel.

Fuselage: The pilot's cabin is situated in the nose of the fuselage above the curved and elongated inlet duct. The guns, nose wheel, main wheels, equipment, and engine are all housed in the fuselage.

Stabilizers: The function of the controls (elevators, rudder controls, and ailerons) is performed by the downward sloping wing tips.

Undercarriage: The main wheels retract forward and sideways into the fuselage while rotating through an angle of 180° about the oleo-axis. The nosewheel pivots towards the rear and rotates through an angle of 90° about the oleo axis. Main wheels are 660 x 190 (26" x 7.5"), nosewheel 560 x 200 (22" x7.9").

Engine: The engine is installed in the fuselage and inlet air is ducted from the fuselage nose.

Fuel Installations: The total of 385 gallons is located in the wings and is unprotected.

Armament: 2 x MK 108 with 100 rounds each installed beside the pilot.

4. SHORT DESCRIPTION OF THE FOCKE-WULF I

General: Swept back mid-wing with sharply swept back vertical tail and highly placed, swept back horizontal tail.

Wings: Two piece wings with steel spars, wooden ribs, and wooden covering. Fuel carried in the wing. No means of increasing $C_{L_{MAX}}$ are used (such as slots in the leading edge or other means of preventing separation), but only landing flaps at the trailing edge.

Fuselage: The upper part forms the main structural part of the fuselage. It contains the pilot's cabin, equipment and fuel space. The intake duct, the armament and landing gear, are enclosed in the lower part of the fuselage. The vertical tail is tied to the fuselage above the engine.

Stabilizers: The sharply swept back vertical tail supports the horizontal tail and is a single spar wood torsion tube. The rudder is the same as the wing aileron and is interchangeable with it. The swept back horizontal tail is wooden.

Undercarriage: The main gear is of the single leg type and pivots forward into the fuselage. The oleo leg is from the FW 190. The nose-wheel pivots aft by a hydraulic strut. Main wheels are 700 x 175 (27.5" x 6.9"), nose wheel is 465 x 165 (18.2" x 6.5")

Engine: The HeS 011 engine is built into the aft fuselage; gets its air through a cylindrical inlet duct from the nose of the fuselage. The engine is accessible through the undercarriage opening and removable cowling.

Fuel Installation: A protected tank of 45 gallons is located in the fuselage. The rest of the 380 gals. is in the wings and is unprotected.

Armament: Normal complement 2 x MK 108 with 100 rounds each located in lower fuselage beneath the pilot. Additional armament of 2 x MK 108 with 60 rounds each.

Armor: Aircraft protected against 12.7 mm. from the front.

5. SHORT DESCRIPTION OF FOCKE-WULF II

General: Swept back mid-wing with turbo intake in nose of fuselage.

Wings: Two piece wing with steel spar, wooden ribs and wooden covering. Wings can hold 344 gallons of fuel. No lift increasing devices in the leading edge - only plain flaps at trailing edge.

Fuselage: Top part of fuselage includes pressure cabin and space for 2 protected fuel tanks and the equipment. Lower part of fuselage carries the tricycle gear and the guns. The engine is installed in the aft part of the fuselage. The fuselage is dural.

Stabilizers: Wooden tail, swept back, normal arrangement.

Undercarriage: The main struts are air-oleo, using hydraulic retraction. The nose wheel rotates through an angle of 90° when retracted. Main wheels 700 x 175 (27.5" x 6.9") interchangeable with 740 x 210 (29" x 8.3"). The nose wheel is 465 x 165 (18.3" x 6.5").

Engine: The engine is under the fuselage behind the pilot. Air is taken in through a cylindrical duct running from the fuselage nose.

RESTRICTED

Fuel Installation: Two protected fuselage fuel tanks hold 79 gals. and 132 gals. The wing tanks are unprotected with a total capacity of 344 gals. Each panel has 6 fuel compartments and the fuel is pumped from the wing into a fuselage tank.

Equipment: In addition to standard fighter equipment, space is provided for an automatic pilot.

Armament: The regular armament is 2 x MK 108, with 100 rounds for each, arranged left and right of the intake duct. A third MK 108 is possible at the expense of the forward fuel tank.

Armor: Front protection for the pilot against 12.7 mm. fire.

6. SHORT DESCRIPTION OF MESSERSCHMITT P.1101

General: Swept back, mid-wing with engine in stern of fuselage and normal arrangement of swept back stabilizers.

Wings: 2 piece wooden covered wings with 40° sweepback. Slot with varying percentage depth at leading edge. Plain flaps.

Fuselage: The forward fuselage contains the air duct for the engine, the nose wheel and the "pressure proof" cabin. The center fuselage holds the fuel tanks and main gear at the top and the engine below. The aft fuselage is cone shaped and contains the equipment and supports the tail.

Stabilizers: Normal wooden swept back horizontal and vertical surfaces.

Undercarriage: The nose wheel rotates through an angle of 90° on retraction. The main gear is cantilever using air oleo struts and hydraulic retraction. Main wheels are 740 x 210 (29" x 8.3") and the nose wheel is 500 x 180 (19.6" x 7.1").

RESTRICTED

Engine: The He 011 engine is located in the underside of the fuselage with the air intake at the nose of the fuselage.

Fuel Installation: 317 gallons in protected fuselage tanks above the engine. 73 gallons unprotected in the wings.

Armament: 2 x MK 108 with 100 rounds each located left and right of the intake duct under the pilot Additional 2 x MK 108 with 100 rounds each provided for.

Armor: Protection for the pilot against forward fire of 12.7 mm. and rear fire of 20 mm.

7. SHORT DESCRIPTION OF MESSERSCHMITT P-1110

General: Swept back, low wing with turbo in aft fuselage and normal swept back tail arrangement.

Wings: Same as P-1101.

Fuselage: Circular, all-metal fuselage. Forward fuselage contains the guns in a nose compartment, nose wheel, and pressure proof pilot's cabin, with radio equipment. The center fuselage takes the wings, main gear, protected fuel tanks, and air intake ducts with boundary layer removal slots. The turbo and tail surfaces are located in the aft fuselage.

Stabilizers: Normal tail arrangement with sweepback of 40°. V-tail can be substituted.

Undercarriage: Nose gear is single strut with bent wheel forks retracted by hydraulic struts into the gun compartment. Main undercarriage retracted obliquely forward into the wing roots by self-latching hydraulic struts.

Engine: The HeS-011 engine is located in the aft fuselage. Air is ducted to the engine through inlets in the side of the fuselage. The boundary layer running against the long forward fuselage is taken in through a supercharger, coupled to the turbo, through 2 slots in the curved duct.

Fuel Installation: 395 gals. protected fuel tanks in the center fuselage between the pilot's cabin and the turbo.

Armament: 3 x MK 108's with 2 x 70 and 1 x 100 rounds are located in the gun compartment. The addition of two more MK 108's is possible.

Armor: Protection for the pilot against fire of 12.7 mm. from the front and 20 mm. from the rear.

8. SHORT DESCRIPTION OF MESSERSCHMITT P-1111

General: Tailless, swept back mid-wing.

Wings: Wings designed to hold 395 gallons of unprotected fuel. Big sweepback with small aspect ratio and large taper. Slots are installed in the region of the ailerons on the wing leading edge.

Fuselage: All metal fuselage. Forward fuselage contains gun compartment, nose wheel, pressure proof pilot's cabin, with the radio equipment behind the pilot. Center fuselage comprises wing supports, undercarriage, equipment and air ducts. The aft fuselage supports the turbo and vertical tail.

Stabilizers: Elevons are used, and the vertical tail on the fuselage is highly swept back.

Undercarriage: The nose wheel retracts aft into the gun compartment. The wide-tread main gear retracts inboard into the wing roots. The main wheels are 740 x 210 (29" x 8.3") and the nose wheel is 465 x 165 (18.3" x 6.5 ").

Engine: The air is ducted through two slightly curved ducts from the leading edge of the wing root to the HeS 011 located in the aft fuselage.

Fuel Installation: Total fuel of 395 gallons is carried unprotected in the wings.

<u>Armament</u>: 2 x MK 108 with 100 rounds each in the nose gun compartment. 2 x MK 108 with 100 rounds each located in the wing roots.

<u>Armor</u>: Protection for pilot against forward fire of 12.7 mm. and rear fire of 20 mm.

	Units	JU EF 128	BV P-212	He P-1078	FW.I	FW.II	Me P-1101	Me P 1110	Me P-1111
WINGS:									
Wing Area	FT^2	189.5	150.5	191.5	242	215	170.5	170.5	301
Span	FT	29.2	23.	29.5	32.8	31.1	27.	27	30
Aspect ratio		4.5	3.5	4.55	4.45	4.5	4.29	4.29	3.0
Taper ratio		.57	1.0	--	.8	.64	.524	.524	.3
Sweepback angle $\cdot\frac{1}{4}C$		45°	40°	40°	40°	32°	40°	40°	45°
Root Profile		0010-45	0012-50	0012	0010-40	0010-40	0008-40	0008-40	0008-40
Tip Profile		0010-45	0012-50	0012	0010-40	0010-40	0012-40	0012-40	0008-40
Landing Device on L.E.		Nose split flaps	Nose split flaps	---	---	---	Slats	Slats	Slat in front of ailerons
Landing Device on T.E.		Split flaps	Split flaps		Plain Flaps	Plain Flaps	Plain Flaps	Plain Flaps	Split flaps
FUSELAGE: (without cabin)									
Height of Fuselage	FT.	4.6	5.25	5.09	5.9	5.09	5.7	3.67	3.54
Width of Fuselage	FT.	4.16	4.44	3.67	4.6	4.26	3.94	3.28	3.12
Slenderness of Fuselage	D/L	.178	.171	.208	.205	.177	.155	.108	.12

SURFACE AREAS

	JU EF 128	BV P-212	HE P-1078	F.W.I	F.W.II	ME P-1101	ME P-1110	ME P-1111
Stabilizer Surfaces(Ft^2)								
Horizontal Tail	---	---	---	28.7	32.1	28.1	34.5	
Vertical Tail	36.3	---	---	35.6	18.4	16.1	20.6	22.9
Total	36.3	42.5	24.8	64.3	50.5	44.2	55.1	22.9
Surfaces - Total Wetted Area								
Wing	359	294	345	460	400	305	307	565
Fuselage	247	322	196	334	295	334	300	196
Stabilizers	73	83	50	128	101	88	110	46
Total Surface	679	699	591	922	796	727	717	807

RESTRICTED

WEIGHTS

	JU EF 128	BV P-212	HE P-1078	F.W.I	F.W.II	ME P-1101	ME P-1110	ME P-1111
Wings	1060 lbs	1070 lbs	lbs	1160 lbs	961 lbs	992 lbs	992 lbs	1259 lbs
Fuselage	793	928		916	810	661	771	441
Tail Surf.	110	210		286	220	199	199	110
Controls	88	88		77	77	84	99	88
Undercarr.	375	610		555	464	509	509	509
Airframe	2426	2906	2180	2994	2532	2445	2570	2407
Turbo Install.	2275	2040	2205	2190	2180	2170	2240	2080
Equipment	418	353	670	503	405	420	420	420
Armament & Bullet Proof.	630	680	353	729	680	682	972	1078
Weight Empty	5749	5979	5408	6416	5797	5717	6202	5985
Crew	220	220	220	220	220	220	220	220
Fuel	2750	2750	2640	2750	2750	2750	2750	2750
Ammo.	264	264	255	264	264	264	264	424
Useful Load	3234	3234	3115	3234	3234	3234	3234	3394
Take-Off Weight	8983	9213	8523	9650	9031	8951	9436	9379
Landing Weight (less 1760 lbs.fuel)	7223	7453	6763	7890	7271	7191	7676	7619

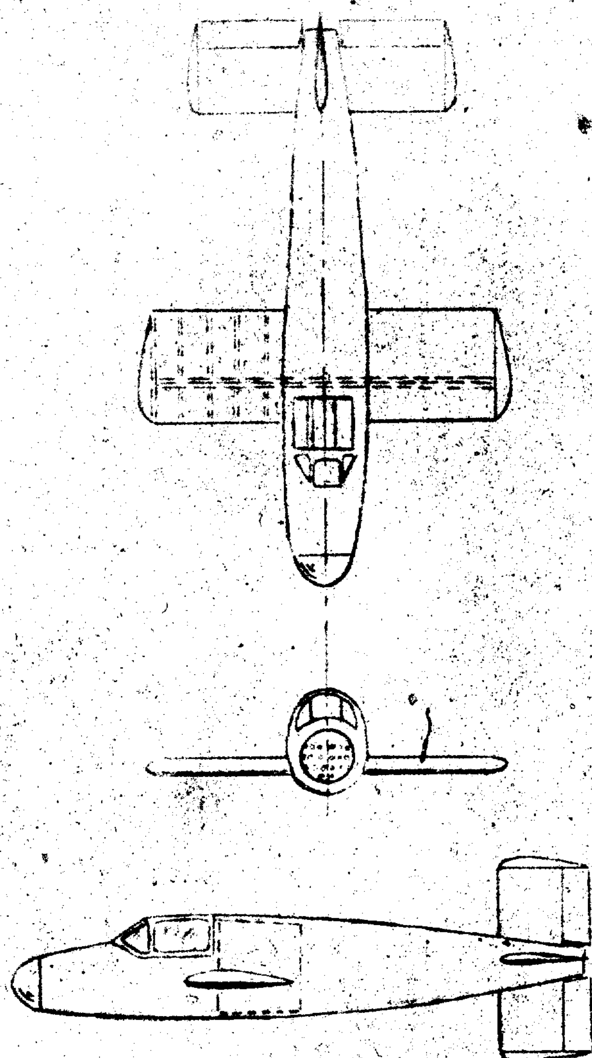
RESTRICTED

PERFORMANCE (IN ACCORDANCE WITH DVL PROCEDURE)

	JU EF-128	BV P-212	HE P-1078	F.W.I	F.W.II	ME P-1101	ME P-1110	ME P-1111
Horizontal Speed								
At Sea Level (MPH)	a.562 b.551	568		542	562	551	a.562 b.550	a.560
At 23000 ft. (MPH)	a.618 b.610	600		595	600	610	a.622 b.619	a.620
Rate of Climb								
At Sea Level (ft/min.)	4510	4200		4040	4550	4380	4240	4670
Take-Off Run(Ft)								
(Unassisted)	2300	2750	2300	2130	2180	2330	2590	1970
Landing Speed (MPH)	108	105	106	101	102	107	105	96
Landing Run (ft)	2180	1990	2100	1640	1700	1870	2000	1480

a. 96% Thrust

b. 92% Thrust



BP-20B "NATTER"

BP-20B "NATTER" (VIPER) ROCKET PROPELLED AIRCRAFT

This airplane was designed as a defence weapon. In operation it is in between a directed missile and an interceptor fighter. Originally it was intended to be completely expendable and to have a pilot ejecting device, but the R.L.M. subsequently decided to salvage the rear half of the fuselage, together with the expensive rocket unit. A parachute was built in the fuselage for the purpose of salvage and also as a brake to catapult the pilot from his seat after the nose section is released. This was not too successful as there was sufficient fuel left in the unit to cause an explosion when it hit the ground.

Dimensions:

Wing Area	39 sq.ft.
Span overall	13.1 ft.
Length overall	20.6 ft.
Horizontal tail area	27 sq.ft.
Horizontal tail span	8.2 ft.

Weights:

Take-off weight inc. ATO units	4925 lbs.
Weight empty	1940 lbs.
Weight of ATO units	1000 lbs.
Wing loading at take-off	95.5 lbs/sq.ft.
Wing loading at end of flight	37.6 lbs/sq.ft

Load Factor — 6 ("safe")

Performance

Speed max.	620 mph at 16400 ft.
Rate of Climb	37400 ft/min.
Max. Range after climb	36 miles at 500mph at 9800 ft.
Max. endurance	4.36 min. at 500 mph at 9800 ft.

Power Plant

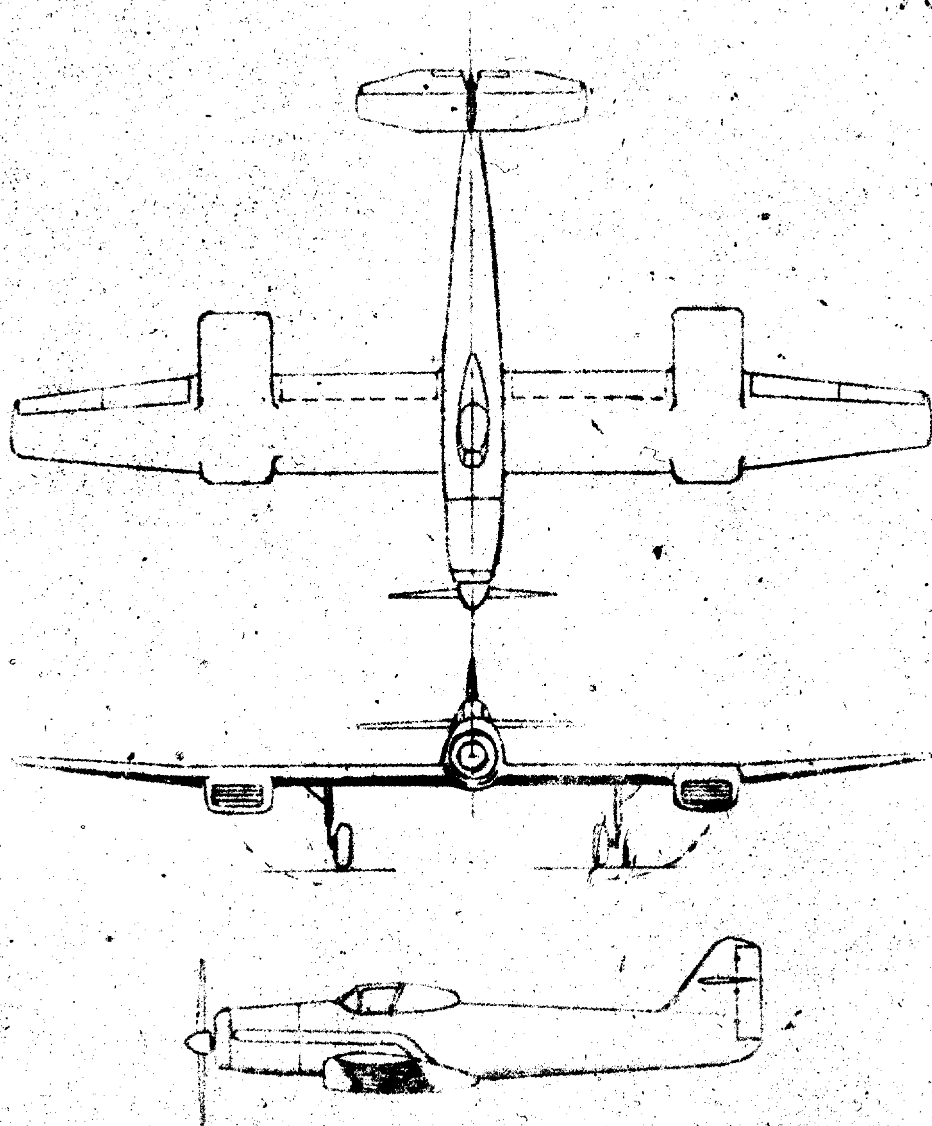
Motor 1 x HWK 109-509 bi-fuel rocket unit.
2 fuel tanks are located behind the pilot; T = 96 gal. C = 42 gal., total, 138 gal.

Construction

The fuselage is of semi-monocoque construction with laminated skin, stringers and formers, and is made in two sections: the joint being designed tobreak or be forced apart as the pilot is ejected.

The wings and tail surfaces are also constructed of wood. The wing is of the single spar type. The spar is laminated wood and is continuous from one wing tip to the other tip. The same type spar is used for both vertical and horizontal tail surfaces. All surfaces have built up ribs and plywood covering. There are no ailerons on the wing, rolling control being obtained by differential operation of the elevators. The wing and tail are of ractangular planform without dihedral, sweepback, or taper. The airfoil used is 12% of chord thick and the point of maximum thickness is located at 50% of chord. The section is symmetrical. The tail setting is -1° to the wing. The vertical surface is approximately 2/3 above the fuselage and 1/3 below. The structure was stressed for 6g acceleration at 1100 km/hr. at 3 km altitude (684 miles/hr. at 9840 ft.)

From wind tunnel tests at Mach numbers close to 1, no bad effects of compressibility on stability or control were reported. At $M = .4$, $C_{DR} = .08$.



BV-155B HIGH ALTITUDE FIGHTER

BV-155 B and BV-155 C HIGH ALTITUDE FIGHTER

	BV-155 B	BV-155 C
Weight (lb.)	12,320	13,200
Wing Area	420	384
Fuel Cap.(Imp.Gal.)	110	264
Type of Fuel	not quoted	B4
Engine	DB-603A + TK 15 Supercharger	DB-603U + TK 15 Supercharger
Armament:	2 x MK 151/20 + 1 x MK 108 or 2 x MG 151/20+ 1 MK 103 or 3 x MK 108 or 3 x MK 103	1 x MK 108 with 60 rounds + 2 x MG 151/20 with 200 rounds

Performance:

BV-155 B MPH	at Ft.	BV-155 C MPH	at Ft.
295	10,000	306	10,000
325	20,000	332	20,000
362	30,000	363	30,000
404	40,000	398	40,000
429	50,000	428	50,000

	BV-155 B	BV-155 C
Max. Speed	431 MPH between 51000 & 54500 ft.	428 MPH at 50000 ft.
Ceiling	56000 ft.	55200 ft.

It is noteworthy that the top speed of the BV-155B is attained at 51000 ft. and maintained up to 54500 ft., after which it falls off very rapidly. A graph shows that at the ceiling of 56000 ft. the speed is only 320 mph. The sub type "C" attains its max. speed of 428 mph at 50000 ft., but the fall off is more gradual, the speed at the ceiling of 55200 ft. being 384 mph.

The wing structure is interesting in that all loads are carried in a steel box beam. The landing gear, intercoolers, etc. are bolted into this beam. The pressure cabin is just a welded steel "bathtub"

The BV-155C had the intercoolers in the fuselage, rather than on the wings, and the landing gear folded inboard, instead of outboard.

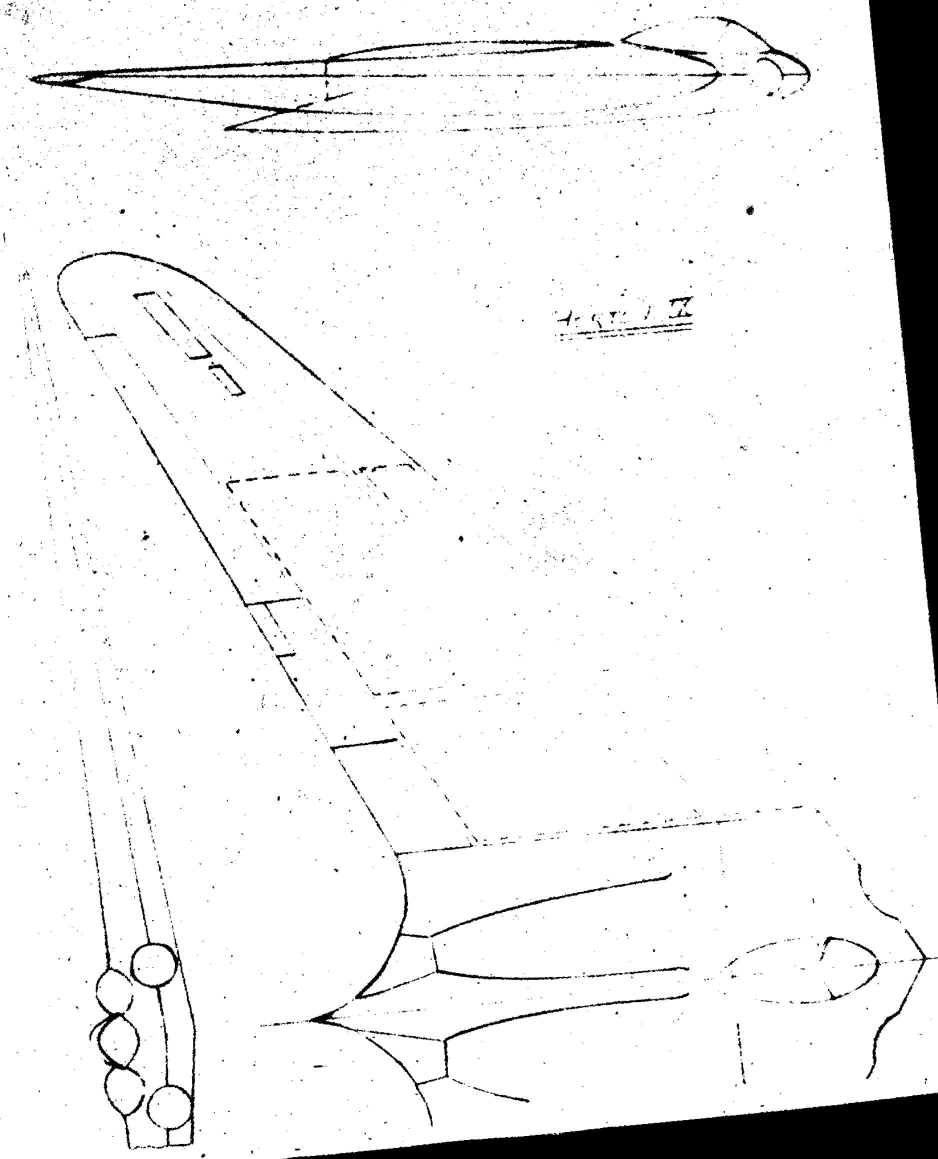


Figure IX

THE HORTEN IX - (Ho 229)

The Ho-229 is a flying wing fighter with large sweep back. There are no vertical surfaces, directional control being obtained by spoilers on the wing. Elevons are also used, as well as plain flaps. There are no leading edge slots. Tip stall has been eliminated by washing out both root and tip sections with respect to the middle third of the semi-span.

The ship is powered by two Jumo 004B jet engines mounted close inboard and passing through the main wing spar. The jets exhaust over the wing trailing edge which is suitably protected by steel skin.

The cantilever wing has one main and one auxiliary spar. The center section is of welded steel tube construction and the outer panels are wood with plywood covering. The wing tips are dural.

The landing gear is of the tricycle type with the main wheels retracting inboard and the nose wheel retracting aft. The nose wheel is self-centered by a spring and cam.

The control system is described under the High Speed Control section of this report.

It was proposed to carry a maximum armament of 4 MK 108 guns and a bomb load of 4400 lbs.

A spring catapult seat is provided for pilot safety in emergency.

The wing section is 14% t/c at the root and 8% at the tip. The airfoil is symmetrical at the tip and has 1.8% camber at the 30% chord of the root (similar to RAF 34). The jet entry in the leading edge of the root effectively reduces the thickness of the root section. $C_{L_{MAX}}$ is 1.3 with flaps and 1.15 without. For the airplane C_{D_O} = .0110 and f = 6.18 ft.^{2}.

The Ho-229 was designed for 7 g "safe load factor" with a safety factor of 1.8 making ultimate load factor equalling 12.6.

The gross weight with guns, etc. is 18,750 lbs., wing area is 560 sq.ft. and span is 55 ft. Top speed is approximately 590 mph and rate of climb at 16,550 lbs. is 4330 ft./min. Service ceiling is claimed to be 52,500 ft. and duration is 3 hours with 635 gallons of fuel.

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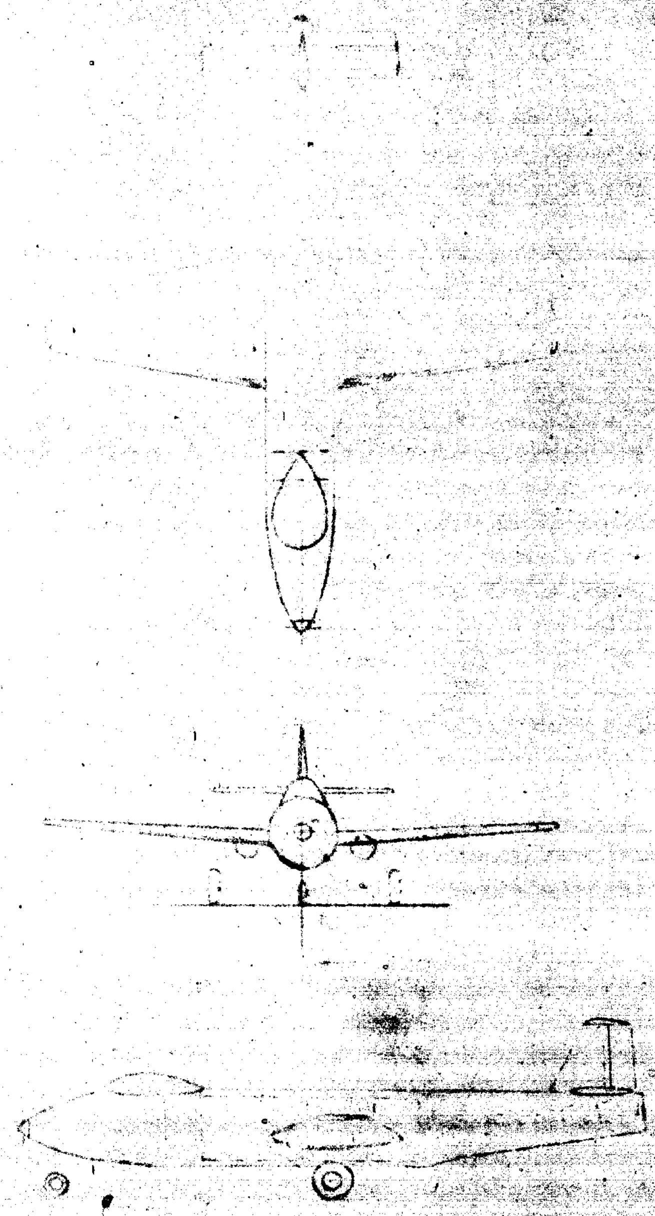


EF 127 • WALLI

EF 127 - "WALLI"

The EF 127 is a rocket propelled target Defence Interceptor. It is a Junkers design. It is not known if an airplane of this design has ever been built.

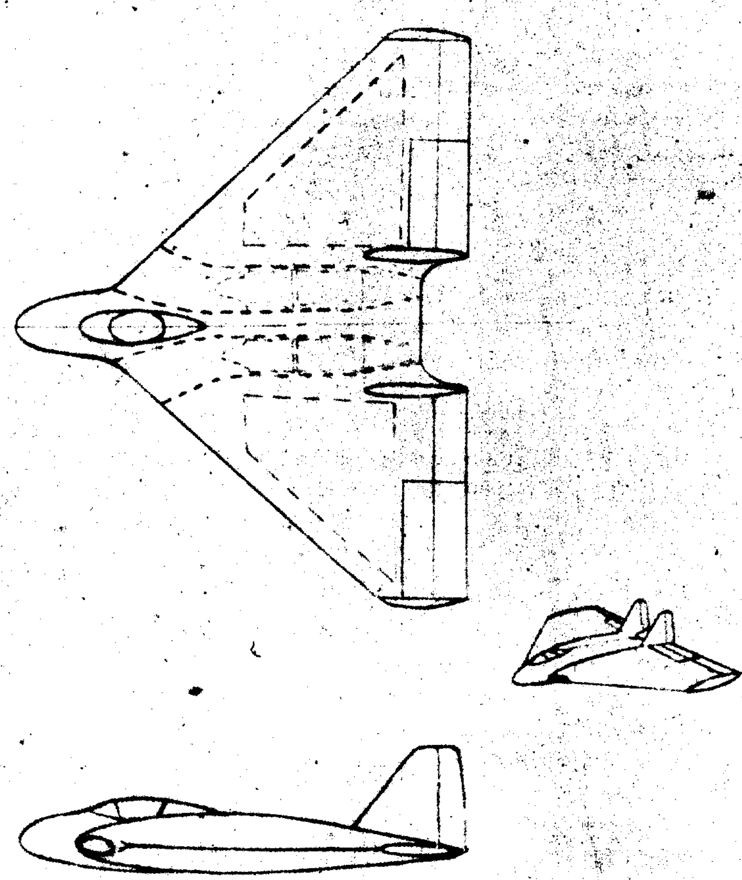
Power Plant	1 x HWK 109-509
Thrust range	440-3750 lbs.
Cruising unit	330-660 lbs.
Take-off	Horizontal
Landing	on skids
Armament	2x MK 108 (60 r.p.g.)
Wing Area	95.7 sq.ft.
Fuel - T-Stoff	2400 lbs.
Fuel - C-Stoff	1100 lbs.

Weights

Assisted take-off units	398 lbs.
Take-off weight	6540 lbs.
Flying weight, at take-off	6140 lbs.
Weight - Fuel expended	2720 lbs.
Wing loading at take-off	68 lbs/sq.ft
Wing loading-fuel expended	28.8 lbs/sq.ft.

Performance

Level Speed	630/S.L., 560/36,000 ft.
Rate of climb at S.L.	26,200 ft./min.
Range after climb	66.5 m at 435 mph at 16400ft.
	and 61.5 m at 435 mph at 32800ft.
Endurance	9.6 min. at 435 mph at 16400ft.
	and 9.17min. at 435 mph at 32800ft.



LP-11

THE LIPPISCH LP - 11

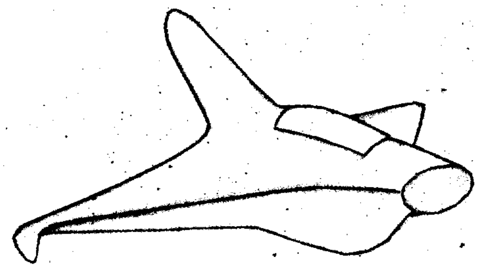
The LP-11 is a low aspect ratio flying wing intended for use as a fighter. A glider version was in process of construction at the close of the war, but was not completed.

The ship was to be powered with two Junkers 004 gas turbine jet engines and has a gross weight of 16,000 lbs. The total wing area is 539 sq.ft. and has a span of 35.5 ft. which gives an aspect ratio of 2.34. The wing has root thickness of 17%, tip thickness of 9%, and has no twist. The wing profile is the Lippisch configuration described in the airfoil section of this report. The maximum occurs at 37% of chord for the root section and 33% for the tip. The airfoil is a symmetrical section. Without flaps, the maximum lift coefficient is 1.0, and with slots and flaps is 1.2

The wing panels are wood, have no shear webs, and are tied all around the periphery to the fuselage by a series of bolts. Two beams then carry the load through the fuselage. The shear webs in this carry-through are not continuously attached to the caps, being connected only by a series of blocks glued to web and cap with 8 inches or so between blocks. Between blocks the wooden web has no fore and aft restraint being free-edged top and bottom.

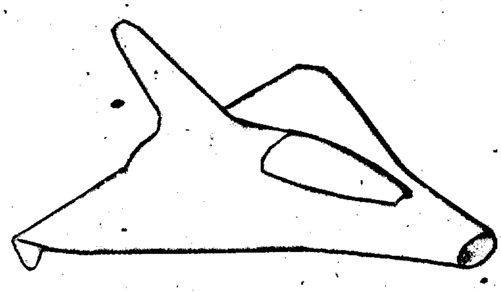
The over-all length of the ship is 23 ft. A very rugged tricycle landing gear is used to allow a high rate of descent on landing. The armament was to consist of 2 MK 103, which is the long 30 mm. cannon, in each wing. An alternate installation is a 7.65 cm. tank gun in each wing. The fuel is all carried in the wings, 475 gallons in each panel.

High speed in level flight is estimated to be 650 miles per hour at 19,700 ft. altitude. Cruising speed is 530 mph and a range of 1870 miles is expected.



LF-12

LP-12



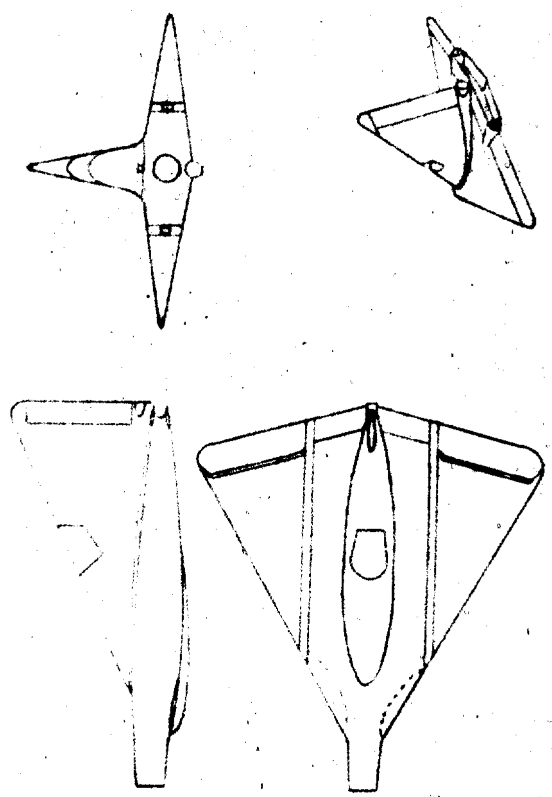
THE LIPPISCH LP-12

This ship was proposed by Dr. Lippisch, but was never built and only a small amount of engineering work was done on it. It was to be powered in flight by a liequid fuel athodyd built into the plane. It was expected to be a supersonic airplane.

The total wing area was approximately 395 sq.ft. and the aspect ratio was 1.33. The wing section was Dr.Lippisch's special elliptical profile with maximum thickness at 45% chord. The wing had no twist.

The undercarriage consisted of a single main wheel and wing tip skids . Junkers 004 gas turbine fuel pumps were used to transmit the fuel from the tanks to the burners.

The ratio of duct, entrance area to the maximum cross-sectional area of the combustion chamber was approximately .4. The power was to be controlled by adjustable flaps at the jet exhaust.



LP-13

THE LIPPISCH LP-13

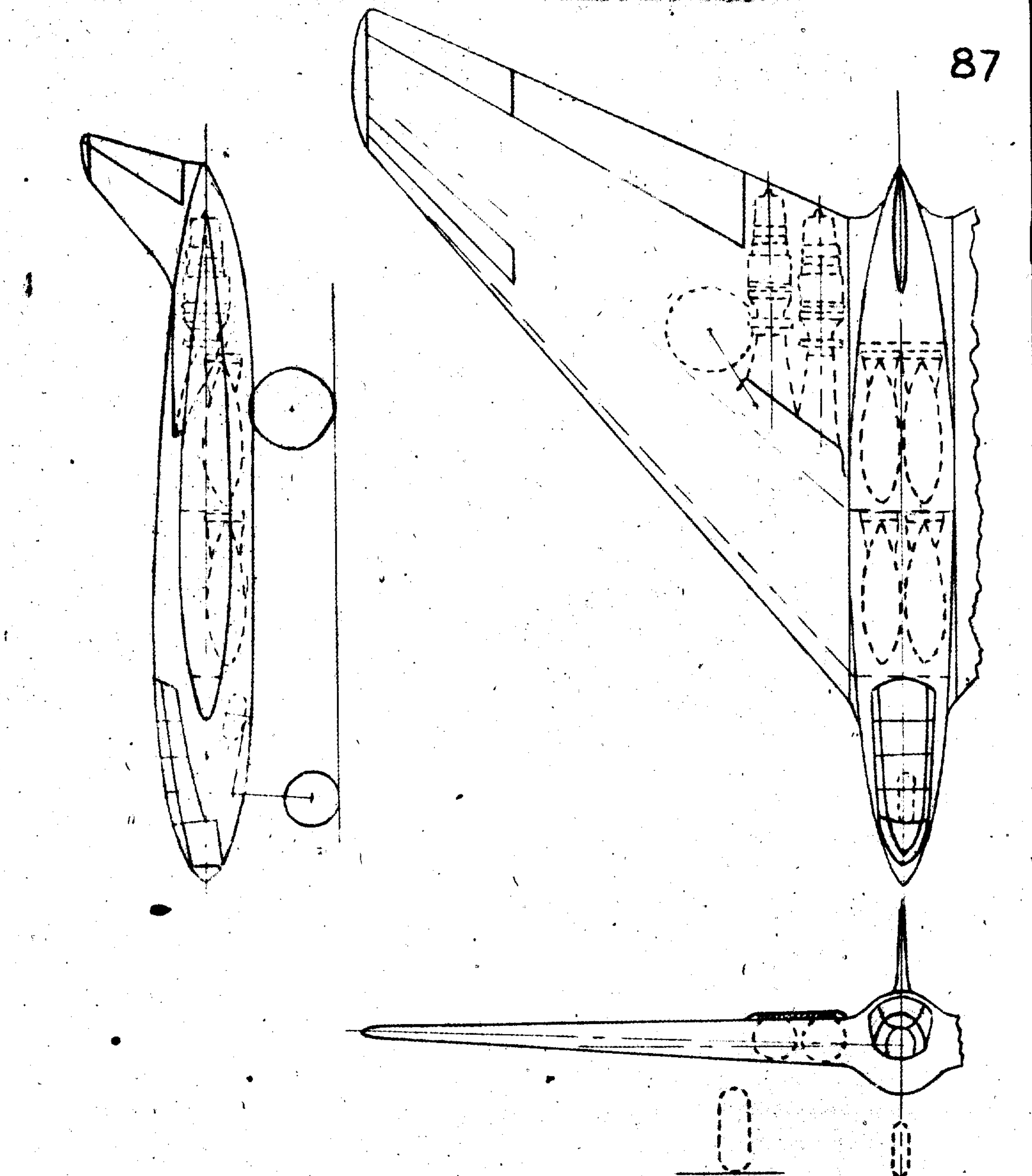
This ship was a low aspect ratio flying wing powered with a coal burning athodyd. A glider version of this airplane was almost completed at the end of the war and is known as the DM-1.

The wing area was 215 sq.ft., gross weight was 5080 lbs., and the maximum level flight speed was intended to be 1020 mph. The wing span was 19.7 ft. and height from bottom of skid to top of fin was 10.65 ft. The wing aspect ratio was 1.8. The airfoil section is Lippisch's elliptical profile with maximum thickness at 45% chord - the same as on the LP-12. Sweepback of the wing leading edge was 60°. Total length from leading edge of the duct to the wing trailing edge at centerline of ship was 22 ft. The fin and two outer panels were separated from the center panel, which is the athodyd chamber, by cooling air ducts. The wing had no twist or dihedral.

Because of the high angle of attack on landing (as high as 35°), a windshield was provided in the bottom of the wing for pilot vision. Although a skid for landing is shown in the accompanying sketch, the glider version has a tricycle landing gear. This is possible because there is no power plant in the center panel of the glider.

According to wind tunnel tests, the LP-13 is just barely stable directionally, probably due to the comparative inefficiency of the vertical tail. It has a low aspect ratio and its center of pressure has a poor lever arm to the center of gravity because of its long, narrow shape. This large fin also produced an excessive amount of drag and was not considered satisfactory for a supersonic airplane.

The power plant is of special interest. Very simply, it is just a basket of lumps of coal centrally located in the burning chamber of the duct. This type of power plant has not been fully developed and its performance is not too definite, although extensive tests were being run in wind tunnels by Dr. Lippisch.



MESSERSCHMITT BOMBER

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THE MESSERSCHMITT TAILLESS JET BOMBER.

In the Fall of 1944, Messerschmitt began the investigation of designs for a jet bomber to carry 8800 lbs. of bombs for 4350 miles at 500 mph. The design study was based on using four HeS 011 jet engines of 2860 lbs. static thrust each.

A conventional arrangement with a tail was found to be less promising than a tailless arrangement for combined range and high speed. The following figures are for the latest arrangement studied, but are for a range of 3100 miles. Take-off weight would be increased to 77000 lbs. to get the 4350 mile range.

Wing Area	1290 sq.ft
Span	65.6 ft.
Aspect Ratio	3.33
Taper Ratio	.25
Tip chord	7.9 ft.
Root chord	31.5 ft.
Sweepback	45° at 25%C.
Thickness ratio-root and tip	10%
Max.thickness of wing at ..	55% chord
Take-off weight	67,600 lbs.
Landing Weight	30,100 lbs.
Wing loading at take-off ..	52.4 lbs/sq.ft.
Wing loading at landing ...	23.3 "

Take-off would be by rockets or catapult on a trolley, the landing gear being used only for landing. The ship was to be cruised at all times at its operational ceiling - which varied from 16500 ft. after take-off to 33000 ft. near the end of flight. It was proposed to cruise at a speed corresponding to a point on the drag curve just below the compressibility drag rise. The cruising drag coefficient was expected to be between .010 and .013.

Slotted flaps and leading edge slots were to be used which gave a $C_{L_{MAX}}$ of 1.4. The wing was to be metal, although a wooden wing outboard of the engines was being investigated as a method of reducing the cost of manufacture.

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The bombs are in the fuselage and most of the 5300 gallons of fuel would be stowed in the wing outboard of the engine and wheels, the remainder being in the fuselage above the bomb bay.

The fuel tanks are unprotected, the fuel being heavy oil (melting point about 0°C) and relatively non-inflammable. It would be kept fluid by immersion heaters. This fuel was used because of its cheapness and the fact that it was available in the Vienna region.

The c.g. of the airplane was to be around 18-20% of M.A.C. with a maximum trimmable travel of 5-6% M.A.C.

Early in 1945 the Government decided that Messerschmitt did not have sufficient engineering manpower to handle the bomber design and turned it over to Junkers and the Horten Bros. The work done by Junkers is covered in the next pages.

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THE JUNKERS TAILLESS JET BOMBER.

The Junkers Design Group had just begun work on a jet bomber to fulfil the specifications already mentioned for the Messerschmitt bomber. Although no drawings or data were available, the following information was obtained from Mr. Gropler, Chief Designer of Junkers.

The ship was to be tailless with the vertical stabilizers located at the trailing edge of the wing at approximately the middle of the semi-span. This is similar to the Junkers EF 128 arrangement.

The wings have approximately 40^{o} sweepback, aspect ratio of 4.8, area of 1290 sq.ft., span of 78.7 ft. The gross weight is approximately 77000 lbs. and the maximum level flight speed approximately 620 mph. The range is 3720 miles. All fuel is kept in the wings in tanks which are insulated because of the low melting point of the fuel. The tanks are not bullet proof.

The fuselage consists only of a nacelle jutting forward of the wing at its centerline. This nacelle holds the crew, nosewheel and most of the equipment.

The bomb bay is aft of the cabin and the bombs fit into the root airfoil contour. One layer of eight bombs (two rows of four) is provided for.

The 4 He 011 jet units are mounted abreast above the trailing edge of the wing (about half of the power plant extends forward of the actual trailing edge of the wing and half aft of the trailing edge). All four motors are enclosed in one nacelle and a boundary layer duct runs between the nacelle and the wing surface. The center section of the wing is the same span as the width of this nacelle and the bomb bay runs the full span of the center panel.

The main landing gear retracts forward and inboard to a location just outboard of the bomb bay. The nose wheel retracts aft.

The plane carries no armament.

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4. STANDARD EQUIPMENT FOR MILITARY AIRPLANES

The following list of equipment is standard for all German design aircraft and enumerates in detail the equipment referred to in the section dealing with the aircraft submitted in the recent fighter design competition.

I. Equipment for Fighters

A. Fundamental Equipment:

1. Current supply equipment

   generator-1000, 2000, 3000 watts
   battery 24 volt 20 amp hr.

2. (a) Ignition equipment (Reciprocating engine)
   Bosch Magneto ignition with buzzer boost coil.

   (b) Ignition equipment (Jet engine)

   Starting - Bosch vibrator coil
   Running - Hot wire

3. (a) Starting equipment (Reciprocating engine)

   Bosch - Intertia starter motor (elect)

   (b) Starting equipment (Jet engines)

   A small electric starter started a small reciprocating gasoline motor which in turn drove the jet engine compressor shaft to starting speed.

4. (a) Power plant instruments (Reciprocating engines)

   R.P.M. indicator
   Manifold pressure
   Oil - pressure and temperature
   Fuel pressure
   Engine temperature indicator
   Fuel supply
   Switch for controlling fuel pump

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(b) Power plant indicators (Jet engines)

R.P.M. indicator
2 ranges 0-2000, 0-10,000 R.P.M.
Gas temperature 300-1000°C.
Fuel injection pressure
Fuel supply (elect)
Elect fuel pumps

5. Variable Pitch Propeller, manufactured by VDM, including Electric Prop.Pitch Motor. Constant speed control unit electrical, manufactured by A.E.G.

6. Navigation instruments in a separate group located on the panel.

Daughter compass
Air speed indicator (True air speed, altitude compensated)
Rate of climb indicator
Combination artificial horizon and turn and bank indicator
Altimeter
Radio blind landing indicator
Pitot head heater indicator

For aircraft with high Mach number, a gyro stabilized saturated magnetic core type of indicator similar to the Fluxgate compass manufactured by Askania is used.

7. Instrument Illumination, Ultra Violet Light, fluorescent figures.

8. Electrical Flare Shooting Equipment

Flares were discharged automatically by operating a switch. Each flare contained identifying colors.

9. Radio set, "type Fu G 16zy, Ultra Short Wave from plane to plane and from plane to ground with additional direction and range measuring facilities.

To find direction and range the pilot contacted the ground station. The ground station sent a pulse which was rebroadcast instantaneously

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by Fu G 16zy was again picked up by the ground station. From this, the ground station could calculate direction and range. Further development on Fu G 15 or 24 was planned.

Receiving antenna - Profiled pole
Sending antenna - Sling or Area type.

Fu G 25 operated as plane carried recognizing equipment for anti-aircraft batteries and as direction indicator for the ground station.

10. Elevator trim was accomplished in two ways: 1. The conventional trim tab on the movable surface controlled mechanically from the cockpit; 2. an electric motor which changed the angle of attack of the whole elevator surface. This motor controlled by a reversible switch in the cockpit. An electrical indicator showed the position of the surface in the cockpit.

11. Hydraulics: - Landing wheels and landing flaps were operated hydraulically. Position indicated electrically by lamps or magnetic indicators. A new electric-hydraulic wheel and flap operator was planned, using a ball-bearing screw jack. The screw was planned to be made by rollong up the thread form out of hard material on a mother form and casting this thread form in place. The object was an inexpensive way of manufacturing. The long thread took form of the nut and the short unit, around which the balls were fed, carried the external thread.

12. The instrumentation for the oxygen system consisted of a flow indicator similar to our blinker type, also a cabin altitude pressure gage, altitude warning indicator and oxygen indicator.

B. <u>Auxiliary Equipment</u>

1. Auto Pilots

KS 12 and PKs 11 by Patin

K12, K22 and K23 by Siemens
PKS 11 by Patin for stabilized flying at a target

2. Bomb release mechanism and fuse setters

Type ASK-R Bomb Selector Switch
Type SV 10 Intervalometer

Fuse setter: Old system operated on 240-volt DC. This system incorporated the automatic setting of the fuses for either instantaneous explosion upon contact with target or delay after penetration. This system also incorporated the releasing of the bomb for either diving or level flight.

The new system incorporated the additional feature of being able to set the fuse for a time delay after release of bomb. The new system operated on 500-volt DC and the timing of the fuse was proportional to the voltage applied to the fuse mechanism.

3. Electrical and Armament Equipment

Trigger mechanism MG 131 and Mg 151, electro-magnetic. Both electro-mechanical and electro-pneumatic trigger operators were used. For type numbers see report No.7.

Electrical Detonating System:

MG 151 - both mechanical and electrical primer detonation.
MG 131, 108, 109 and 101 - electrical only.

Charging Mechanisms:

All machine guns or cannons have electro-magnetic charging mechanisms with the exception of the following types which are mechanical:

Flak 18, Flak 43, MK 214 A.

4. Sighting Equipment:

   EZ 42 was the only complete calculating sight. Further development of sighting equipment for fixed guns included an apparatus designated as "Faun" which was a combination of EZ 42 and Fu G 217 or 218 which is a radio range computing mechanism which automatically gives range to the sight. "Oberon" incorporated the same principle with the addition of a calculating mechanism for rockets.

5. Receiver Fu G 29 with Push Button Operation for commander aircraft only.

6. Additional Navigation Equipment:

   A device was being worked on for the integration of course direction and distance. It operated from air speed, compass, and gyro position. It automatically integrated the flight path so that the plane's position could be read instantaneously at any time. Ground wind (direction and speed) could be turned in manually. This device was originally planned by Messerschmitt and given to L.G.W. Siemen for further design. It was in limited production, perhaps fifty (50) built.

II Bad Weather Fighter Additional Equipment

A. Radio Set, type Fu G 120 K equipment for navigation:

   1. It automatically records on a paper chart the direction to a ground station that is tuned in manually on the radio set. This equipment operated from a ground station with a rotating beam antenna.

   2. Fu G 125 Blind Landing Receivers: This could also be used with the rotating antenna ground station for navigation. The ground station could record a message on the paper chart of the Fu G 120 K equipment. A further combination was planned using "Eule" which was a system using 1½ centimeters in combination with "Faun" and "Oberon" for automatic firimg of guns or rockets.

III Additional Equipment Used in Night Fighters.

A. Night Fighter Search Equipment:

1. Fu G 218 1 to 2 meters Diapol antenna.

2. "Bremen 0" or "Berlin" both 3 to 12 Cm wavelength using parabolic reflector.

B. Fu G 350 ZC - Receiving Set for Night Fighters:

This is a warning system that picks up the enemy radar signal and warns the pilot that he is within striking range of the enemy night fighter.

IV Additional Equipment for Two-man Fighters

A. Radio set Fu G 10 P long distance sending and receiving communications transmitter receiver.

B. Fu BL 2 Equipment:

This is additional to the Fu G 125 blind landing equipment to give altitude information on approaching the field for complete blind landing. Defence Armament: Remote controlled turrets, type FA 3, FA 15, FA 10 or FA 6.

II. SELF PROPELLED MISSILES

Germany's development of self-propelled missiles, both unguided and guided, is of special interest as an extension of aeronautical science. The aerodynamic problems have been fairly well answered, but the control and direction of the missiles had not been developed sufficiently to make the missiles fully effective. The variations in direction methods do not fall within the scope of this report, and the aerodynamic developments are the same as those applicable to normal high speed airplanes, but the status of missile development and data on typical Flak rockets will be incorporated.

The missiles that had been considered are as follows:

A. Long Range Weapons

1. A-4 (the V 2)
2. 8-103 (the V 1)
3. HDP
4. Rheinbote

B. Anti-aircraft Weapons

1. Unguided Missiles
   a. Taifun
   b. Rym (Oxan)
   c. R 50 Bs
   d. R 100 Bs
   e. 21 cm RBs
   f. 21 cm Wgr 42Bs

2. Guided Missiles
   a. Schmetterling 8-117
   b. Enzian
   c. Rheintochter
   d. Wasserfall
   e. X 4
   f. 8-298
   g. 8-117 (H)
   h. Natter

At the end of February 1945, the status of the missiles was as follows:

A. The following had been completed and were in production:

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1. A-4 (the V2)
2. 8-103 (the V1)

B. The following had priority in development and pilot production:

1. Taifun
2. Rym (Oxan)
3. R 100 Bs
4. 21 cm RBs Spreng Granate Bord
5. 21 cm Wgr 42 Bs
6. Zieldarstellungs Gërat 8-246
7. Bienenkorp

(Note that these are all unguided missiles)

C. The following were to be closed out at the end of the development period:

1. 8-117 Schmetterling
2. 8-344 (X4)
3. Abwandlung Wasserfall
4. Natter

D. The following were to be closed out at once:

1. Enzian
2. Rheintochter
3. R 50 Bs
4. 8-117 (H)
5. 8-298
6. Flak R 42
7. HDP

The following data will give a general picture of the types of developments in Flak rockets (anti-aircraft weapons).

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NAME	Wasserfall	Rheintochter 3	Enzian	Schmetter-ling
TYPE	Supersonic	Transonic	Subsonic	Subsonic
DEVELOPED BY	Electromech-anische Werke	Rheinmetall Borsig	Holzbau Kissing	Heinkel
Length- ft.	25.7	16.4	12.0	14.05
Span - ft.	8.2	10.4	13.1	6.6
Diameter- in.	34.5	21.	35.4	13.8
Weight Empty (less explos.- lbs.)	2890.	1160.	860	330.
Wt.of Launcher	none	617	375	220
Chamber Powder	none	352	330	176
Weight of Main Propulsion Unit				
Fuel	794	194	232	29
Oxygen and Container	3300	740	1000	132
Compressed Air	143	40	77	7
Wt.of Explosive	670	352	660	55
Tot.Starting Wt.	7800	3450	3500	950
Wt.at Garget	3550	1500	1530	390
PERFORMANCE				
Launching direc.	Vert.	Incl.	Incl.	Incl.
Tot.Impulse (lb.sec.)	792,000	231,000	273,000	57,500
Total Launching Thrust	27,600	81,000	13,200	7,500
Launching Acceleration-ft/sec^2	72	540	120	252
Vel.at Target ft/sec	2520-1150	1310-655	820 app.	790 app.
Vel.at End of Burning ft/sec.	2520	1340	890	790
TYPE OF LAUNCHER	none	Powder	Powder	Powder
Impulse-lb.sec.	---	55,000	53,000	30,000
Duration -sec.	---	1	6	4
Thrust	---	55,000	8,800	7,500
Fuel Cons.lb/sec.	---	352	55	44

MAIN PROPULS.UNIT	Liquid Fuel. Compress.Air	Liquid Fuel Compr. Air	Liquid Fuel Pump	Liquid Fuel Compr. Air
Impulse- lb.sec.	792,000	176,000	205,000	27,500
Duration - sec.	About 45	About 45	About 62	About 57
Thrust	17,600	3750/ 5070	4400/ 220	835/130
Fuel Cons. lbs/sec.	About 90	About 22	About 20	About 2.9

The test records on these missiles are given below. The figures in parenthesis indicate the number that did not function properly:

Name	Max.Ht. (Ft.)	Range (miles)	Number Launched (totals)	Without Control	Preset	Radio Control
Schmetter-ling	35,000	13.85	From Ground 49 (59)	2 (1)	---	47 (27)
			From Air 20 (21)			20 (6)
			Total 69 (80)			
Enzian	45,300	15.7	23 (24)	23 (14)		
Rheintoch-ter	49,800	11.8	82 (88)	39 (14)	21 (8)	22 (4)
Wasserfall	60,000	16.5	25 (28)	1 (0)		24 (10)

The Taifun is also of interest. It is a supersonic rocket of 3.3 in. diameter, total weight of 44 lbs. fuel weight 22 lbs. and an impulse of 4400 lb.sec. corresponding to an exit velocity of 6400 ft/sec. Two designs differ slightly, Taifun F being 63.5 inches long, total weight 44.6 lbs. fuel weight 23.8 lbs., duration 2.5 seconds, chamber pressure 50 atmospheres and launching acceleration 1250 ft/sec.^{2}. Taifun P is 69 inches long, total weight 53 lbs., fuel weight 26.5 lbs., duration 1.5 seconds, chamber pressure 100 atmospheres, and launching acceleration of 1970 ft/sec^2. The explosive load of each is 1.45 lbs.

The drag coefficients used for computing the trajectories of the Taifun are as follows:

MACH NO.	.5	1.0	1.3	2.0	3.0	4.0	4.5
C_D	.195	.33	.52	.38	.27	.225	.22

A typical control method was used on the Schmetterling. The missile had both receiver and transmitter and the target was tracked by one device, the missile by another. A computer for lead and parallax furnished appropriate control signals.

Seven schemes consisting of combinations of existing tracking devices are covered under the code names Burgund, Franken, Hansa, Elsass, Brabant, Parsifal and Lohengrin. It is interesting to note that common practice for all directing methods was to direct the missile along the pursuit curve - i.e. always pointing at the target, rather than along an interceptor line.

Proximity fuses were to be used on all missiles. According to Dr. Osenburg of the Reich Research Council, German research had shown that the effective radius of 250 kg. (550 lbs.) of high explosive against a 4-engined bomber was less than 20 meters (65 ft.) Dr. Wurster, on his latest versions of the Enzian, was using a warhead of 2200 lbs. to increase the effectiveness range.

III. DESIGN DEVELOPMENTS AND PROCEDURES

1. Introduction

This division of the report is a survey of the latest information, techniques and theories that the Germans used for their most advanced airplane designs.

No one subject could be thoroughly covered in a report of this scope, but enough information has been given so that each item can be evaluated, and further information can then be obtained through the Air Document Research Center, C.I.O.S. Reports, and other sources. Where possible, specific references have been given.

It should be realized that German aviation was very highly developed. Through their airfoil design and the application of sweepback, German airplanes were attaining level flight speeds above 600 mph or a Mach Number in the neighborhood of .85. The use of sweepback to postpone the drag rise due to compressibility had led to a very definite trend toward flying wings (tailless airplanes). This trend, in turn, entailed much study and development in the field of preventing tip stall, increasing $C_{L_{MAX}}$, and providing good high speed control.

The development of high Mach Number Flak rockets, as well as that of supersonic airplanes such as Dr.Lippisch's, led to much work in the transonic and supersonic regions. Very high speed wind tunnels and Germany's excellent mathematicians had produced a great deal more progress in these regions than exists elsewhere in the world. Accompanying the progress into the design of high speed aircraft was a similar development of high speed aircraft propulsion units, especially the athodyd (propulsive duct or Loren unit).

Many approaches had been made to the aircraft problems, many tests had been run, and much information had been accumulated. Some of this is given in the following pages.

2. AERODYNAMICS

A. High Speed Airfoils

In general the attitude of the German aircraft industry toward laminar flow airfoils, as on the Mustang, has been quite indifferent. There were several reasons for this - the lower C_{Lmax} was not thought necessary or good; there was fear of aileron ineffectiveness at low deflection because of the cusp trailing edge; and the airfoil drag performance was not considered as good as for the German high speed airfoils.

According to comparative tests, the Mustang airfoil, although it had a lower drag coefficient below critical Mach number, reached its M_{CR} at a lower speed and increased in drag coefficient faster with Mach number past critical than the good German airfoils.

Generally, the German airfoils agreed in that most of them had elliptical nose sections, maximum thickness at 40 to 50% of the chord, and straight sided trailing edge section with a maximum included angle of 14° to 15°. Critical Mach numbers were in the neighborhood of .8, varying with thickness ratio - with the emphasis being placed on making the airfoil as thin as possible, of course. Critical Mach number throughout this report is defined as the Mach number at which the drag coefficient begins to rise rapidly.

For supersonic flight, such as for flak-rockets, the two-arc airfoil was most seriously considered (with slightly rounded leading edge). This was usable on flak rockets because the angle of attack could be kept very low and low speed behavior was of no importance. Dr. Lippisch claimed that his airfoil was a good compromise for use on an actual super-sonic airplane that had to fly at low speeds, also, because of its low C_{DMAX}/C_D Low speed. (See further discussion of Lippisch Section)

Below are given the opinions of several of the more prominent airplane designers in Germany on the design of high speed airfoil sections.

(1) Voigt - Chief Designer at Messerschmitt

A symmetrical section is favored for two reasons.

(a) Because there are no "disturbances of moment" at negative C_L. (The likelihood of large pitching moment in a dive is less.)

(b) Because at high C_L, or high g, although the shock wave forms at a lower Mach no. than it would on a cambered section designed for this C_L, it forms near the leading edge where the boundary layer is thin and thus the turbulent break away is less severe up to higher Mach number.

The nose radius is considered relatively unimportant, but the point of maximum t/c should be behind 40% chord. If the maximum t/c is behind 40% chord, then a cusp trailing edge should be used to give a satisfactory pressure recovery.

In no case should a trailing edge angle greater than 14° be used - otherwise it is difficut to get straight curves of C_L vs α , C_L vs β (flap), C_L vs ξ (aileron) and C_H vs ξ (aileron).

The most recent airfoils used at Messerschmitt (other than on flak rockets) are determined as shown below.

The airfoils are symmetrical with maximum thickness at 40% of chord. The thinnest section used was 8% of chord, thickest 12%. (The 8% section was at the root, the 12% section at the tip on the Me-110 high speed airplane - see discussion on sweepback).

The nose radius is given as aE, where E is the radius of an ellipse with the thickness as minor axis and 40% chord as semi-major axis.

$$a = \left(\frac{\text{actual nose radius}}{\text{nose radius of ellipse}} \right)^2$$

For the 8% section, a = 1.19, for the 12% section, a = 1.0. The larger nose radius on the thinner section was done for two reasons - a slight beneficial effect on $C_{L_{MAX}}$ and the holding of a constant nose section on the wing for manufacturing reasons.

Forward of the maximum thickness point the sections are very nearly elliptical, the equation being:

$$y = y_{max} \sqrt{1 - \frac{(m - x)^2}{m}} \times F$$

where
- y = ordinate
- m = position of max thickness
- x = abcissa from leading edge
- F = a factor swelling the ellipse
- $F \approx 1 - a \left(\frac{m - x}{x}\right)^2$

This form of nose shape is claimed to have advantages over the plain ellipse from the pressure distribution standpoint.

The equation of the section aft of the maximum thickness point is of the form

$$y = A_5 x^5 + A_4 x^4 + \ldots\ldots\ldots\ldots + A_o$$

The constants are determined by the conditions -

At the junction point.

$$y \text{ tail part} = y \text{ nose part}$$

$$\frac{dy}{dx} \text{ tail part} = \frac{dy}{dx} \text{ nose part}$$

$$\frac{d^2y}{dx^2} \text{ tail part} = \frac{d^2y}{dx^2} \text{ nose part}$$

And at the trailing edge,

$$y = 0$$

$$\frac{dy}{dx} = \tan(\tfrac{1}{2} \text{ trailing edge angle})$$

$$\frac{d^2y}{dx^2} = 0$$

(For the 12% tip section, another term $A_6 x^6$ was included in the above equation and the expression [illegible] $\frac{d^3y}{dx^3}$ [illegible] was used in the solution. This was done to maintain a constant trailing edge section over the span - again for manufacturing reasons).

(2) Gropler - Chief Designer at Junkers

For high speed, the airfoil should be as thin as possible, (12% chord maximum) and the point of maximum thickness should be between 40% and 45% of the chord.

It is felt that the trailing edge portion of such an airfoil has the greatest influence on critical Mach number, while the leading edge portion influences $C_{L_{MAX}}$. Therefore, it is possible to compromise and obtain good lift characteristics while maintaining minimum drag. The procedure to use is to design first a symmetrical airfoil for low drag and then "wrap" it about a curved chord line that will produce a desired $C_{L_{MAX}}$ while maintaining $C_M = 0$ at $\alpha = 0^o$. (Reverse camber)

This procedure was used at Junkers in the design of the Ju-287 and it is claimed that the cambered airfoil had a slightly higher M_{cr} than the basic symmetrical section.

The procedure recommended is as follows:

For the basic symmetrical airfoil, the leading edge portion is first laid out as half an ellipse with the minor axis equal to the thickness and the major axis equal to twice the distance from the leading edge to the point of maximum thickness. Then this nose is faired in the forward portion so that the leading edge radius is $\frac{1}{2}$ to $\frac{3}{4}$ that of the original ellipse radius at the leading edge.

From the point of maximum thickness, the curve tangent to the forward ellipse and terminating at the trailing edge is the same as the one used by Messerschmitt.

$$y = A_5x^5 + A_4x^4 + \ldots\ldots\ldots + A_o$$

The solution is made similarly to Messerschmitt's, except that Junkers used tan ($\frac{1}{2}$ trailing edge angle) = .1 and recommended less for sections less than 12% thick.

With the symmetrical airfoil thus determined about a straight chord line, its ordinates are then used on a curved chord line. This curved line can be found knowing the camber in the forward portion to obtain the desired $C_{L_{MAX}}$ and the reverse curve necessary to have $C_m = 0$ at $\alpha = 0^\circ$. Junkers figures for the Ju-287 were .015 C camber ordinate at 20% chord with the reverse curvature causing an intersection of the basic straight chord line at 90% chord.

(3) <u>Dr. Lippisch - Head of Aeronautical University of Vienna</u>

A symmetrical section is favored, largely because Lippisch was trying to design for supersonic speeds and he felt that the trans-sonic region could only be "pierced" in a zero lift dive with no wing moment change and he believed a symmetrical section was the only one that qualified.

The airfoil section recommended is of the equation

$$y = h\,(1 - x^2)\sqrt{x}$$

where h = constant that determines thickness.

This equation produces maximum thickness at 45% chord. Varying the exponent in the term $(1 - x^a)$ varies the point of maximum thickness. When a = 1, equation is that of the Munk profiles where maximum thickness is at approximately 33%C.

When this airfoil was used on Lippisch's triangular shaped wings, the center of pressure remained at 50% of root chord until M = .9 then moved to .45C and above M = 1.0 went to .60C and stayed.

The ratio of maximum drag coefficient to low speed drag coefficient reached a peak in the neighborhood of M = 1.2 to 1.4 and was 4.5 to 1, while for normal sections it is 14 - 20 to 1. C_{LMAX} is in the region of 1.0 depending on thickness ratio.

(4) Multhorp - Chief of Aerodynamics at Focke-Wulf.

In general agreement with Gropler and Voigt, Multhorp believed that the most important items in a high speed airfoil were that it should be kept as thin as possible, the point of maximum thickness should be between 40% and 50% of chord, and the trailing edge angle should not exceed 15°.

He did not believe that the leading edge radius was important, within limits, except for its effect on C_{LMAX}.

Multhorp thought the best approach to a high speed airfoil was to make pressure distribution calculations and vary the airfoil to maintain as near as possible constant pressure distribution. He admitted, however, that this had been tried at Goettingen with little success at actual prediction of M_{cr}. (The calculations predicted a lower M_{cr} than actually was experieneed in wind tunnel tests.) However, these Goettingen calculations formed the basis for all airfoils that have been mentioned in this section.

(5) Kaiser Wilhelm Institute für Stromungsforschung Göttingen.

The theoretical calculation of high speed airfoils was performed in the fluid mechanics institute at Göttingen under the direction of Dr.Riegals. Dr.Holstein conducted wind tunnel tests at Reyerhausen and Braunschweig up to Mach Numbers of 1.6, on airfoil sections having elliptical noses and flat surfaces over the rear 20% of chord.

Dr.Reigels had also developed an approximation for the rapid calculation of pressure distribution which was published in Deutsche Luftfahrtforschung in 1942. He had likewise solved the reverse problem of calculating the section required to produce a specified pressure distribution, by an extension of the same method, and had developed a series of aerofoils with gradually increasing favorable pressure gradient. Using the methods of Schlichting, Dr.Reigels derived the theoretical laminar flow instability point for each of these sections and confirmed Schlichting's earlier results that the laminar flow deteriorated rapidly at Reynold's Numbers above eight to ten millions.

Dr.Reigels was of the opinion that the critical region of actual transition to supersonic flow is between M = 0.8 and 0.9. Beyond Mach Numbers of 0.9, because nearly all the velocity over the airfoil is supersonic, he feels that the flow is well enough established so that no additional complications result.

Professor Betz has been successful in calculating the position of the shock wave on the airfoil, by equating the drag resulting from the pressure changes over the airfoil to the loss across the shock wave. The theoretical results agree, up to Mach Numbers of 1.4, with Schlieren photographs taken of the actual airfoil sections under consideration.

Professor Tollmien has performed extensive calculations to demonstrate that the shock wave is not necessarily a product of the boundary layer; but that it can, in fact, exist in the absence of a boundary layer on a smooth curved surface. Professor Tollmien did not consider the problem of locating the shock wave associated with a given surface, but instead considered the opposite problem of finding a surface which can be streamlined in a flow with a given shock wave. This has been accomplished by the method of characteristics combined with the application of a transformal confirmation.

The following reports were written by members of the Kaiser Wilhelm Institute für Stromungsforschung at Göttingen on the subject of high velocity fluid flow. Microfilm copies of these documents, in the original German; can be obtained from the Air Document Research Center.

Theoretical investigations on steady potential flows and boundary layers for high speeds. Lilienthal-Bericht S-13 by Kl. Oswatitch and K.Wieghardt.

Elaboration of method similar to Pohlhausen-Verfahren procedure for the calculation of laminar compressible boundary layers with a specified pressure distribution. Considers also the mutual influence between potential flow and boundary layer flow in the supersonic regime. Starting with an exact solution at a great distance from the body, the form of the body and its pressure distribution is calculated step by step.

Experimental investigation of the compressible flow on and near a curved surface (Parts I and II) Untersuchungen und Mitteilungen, Nr.6608 and 6611 by W.Frossel.

Results of the investigation of flow on and near a curved surface, to evaluate the influence of the exceeding velocity on the curvature. Examination by means of static pressure surveys were made and special consideration was paid to the region where the sound boundary is surpassed. The pressure measurements have been converted into velocity valves and represented in individual diagrams for comparison.

Transition from subsonic to supersonic flow Jahrbuch der Deutschen Luftfahrtforschung 1943 by Kl. Oswatitch and W. Rothstein.

Outlines the solution for a Laval nozzle and gives the application to any other condition of symmetrical flow in two dimensions about an axis.

The methods of characteristics in hydrodynamics. Zeitschrift für angewandte Mathmatik und Mechanik,1945, by Kl. Oswatitch.

Elucidates and simplifies the existing method for the examination and calculation of steadily axially symmetric supersonic flows and of unsteady one dimensional flows.

2. AERODYNAMICS

B. SWEEPBACK

The use of sweepback (or sweepforward) to delay the onset of the compressibility drag rise is probably the outstanding feature of experimental and proposed high speed airplanes in Germany. Although the Me-163 was the first combat airplane with sweepback, almost every German airplane designed since that time has incorporated some form of sweep as a means of increasing its speed.

Although no three dimensional theory for the airflow around a swept-back wing exists, the simple two-dimensional theory has been qualitatively proven in the wind tunnel and flight. Roughly, sweepback of the order of 35° on wings of aspect ratio in the neighborhood of 5 will increase the critical Mach number by .1, compared to a straight planform.

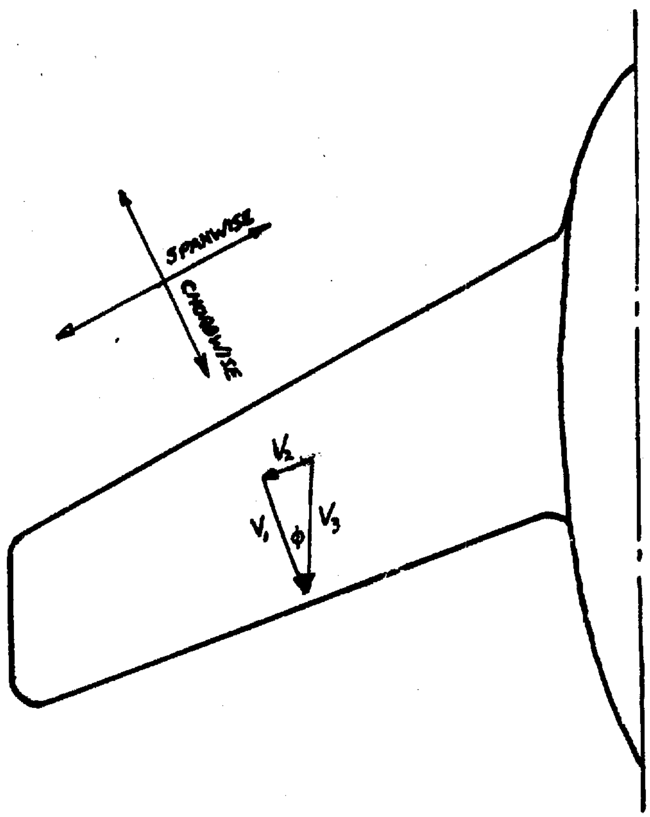
The following discourse attempts to demonstrate the theory and practice in the incorporation of sweepback in German airplanes.

The simple physical explanation of the sweepback effect is as follows:

Consider a wind tunnel with air velocity V_1 spanned by a wing of infinite span moving spanwise across the tunnel with velocity V_2. Then the resultant velocity is V_3 making an angle ϕ with respect to V_1 (or the chord of the wing section). Pressure distribution, C_L, C_m, and Mach No. effect would be dependent upon the chordwise velocity V_1 and independent of V_2, but the (low speed) frictional drag in the direction V_3 would be dependent upon V_3. While V_1 might be subsonic, V_3 would be greater and might even be supersonic.

The case of the sweptback wing is analogous, V_3 being the forward velocity and V_1 the chordwise component of this.

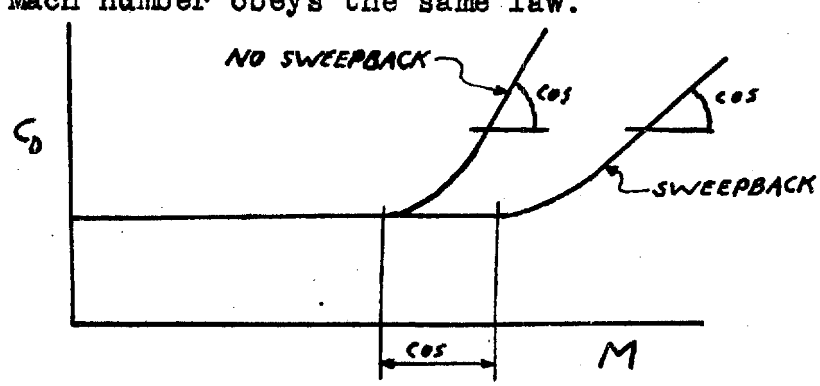
B. SWEEPBACK



From this, it appears that critical speeds are determined by the chordwise flow and therefore

$$\frac{M_{CRIT.}\text{ with sweepback}}{M_{CRIT.}\text{ without sweepback}} = \frac{1}{\cos\phi}$$

Similarly, the slope of the curve of drag rise with Mach number obeys the same law.



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B. SWEEPBACK

It should be pointed out that the beneficial effect of sweepback **is not** a function of the effective reduction in the t/c ratio in the line of flight due to the sweepback. The following data for a particular case examined are given as proof:

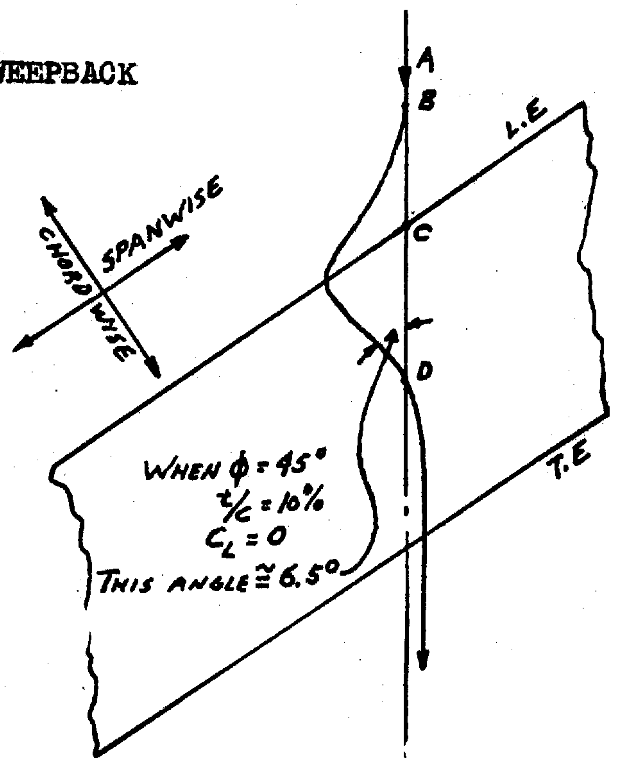
t/c chordwise	Sweepback	Critical Speed m.p.h.	M
12%	0°	562	.738
8%	0°	575	.755
8%	40°	612	.804

It is seen that sweepback gives a greater increase in M_{CR} at an already higher Mach number than does an effectively larger reduction of t/c.

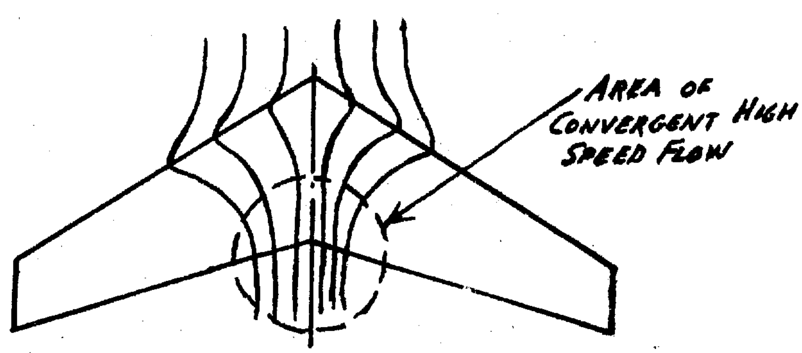
The sweepback theory neglects tip losses, center interference and boundary layer disturbances. Tip losses, by tunnel tests, appear very normal - if anything a little less than on straight wings. Boundary layer disturbances are being approached mathematically at LFA with no usable results as yet. Center interference is explained here.

The path taken by a particle of air flowing over a sweptback wing is as shown in the sketch. As the air particle approaches the stagnation point C, its chordwise component of velocity was falling towards zero at C and thereafter increasing again as the velocity increased over the profile, while the spanwise component remained constant. The reason for the cross over at point D is difficult to explain, unless as function of energy, but wind tunnel tests verify its occurrence.

B. SWEEPBACK

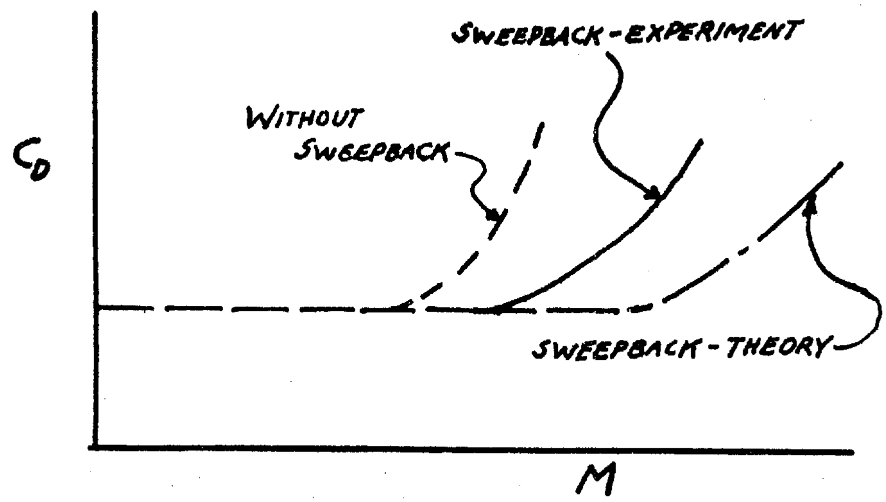


The practical significance of this flow pattern is that, in the central region of a sweptback wing, there develops an area of convergent high speed flow with consequent high local Mach number and liability to high drag and premature breakdown



This center flow pattern is claimed to be the reason that, to date, only about half the theoretical benefit of sweepback has been realized in wind tunnel tests of wings alone.

B. SWEEPBACK



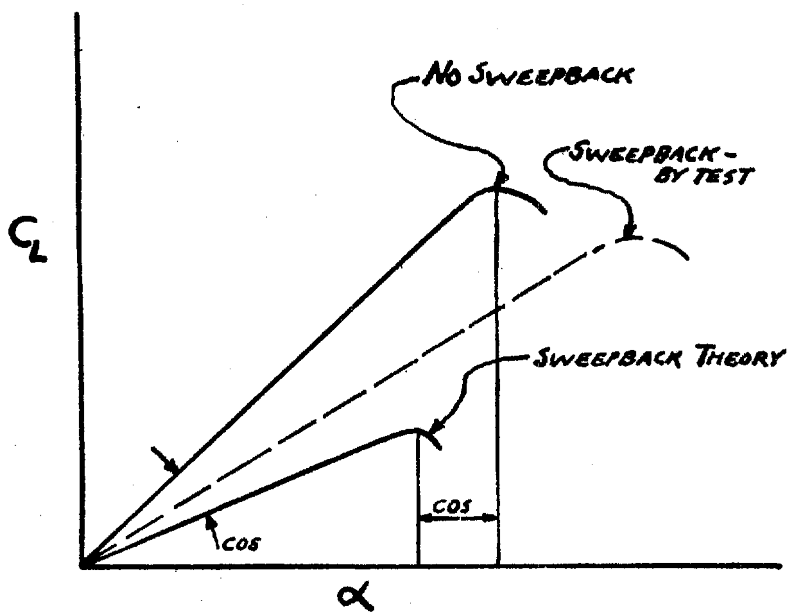
It should be pointed out that with a carefully designed fuselage-wing combination, the interference effect is reduced and with a good leading edge fillet at the wing-fuselage junction, the drag loss is further lowered and the critical speed can be equivalent to that at a point outboard on the wing.

In no case, however, except perhaps on very low aspect ratio wings, is the effect of sweepback nullified by this center interference.

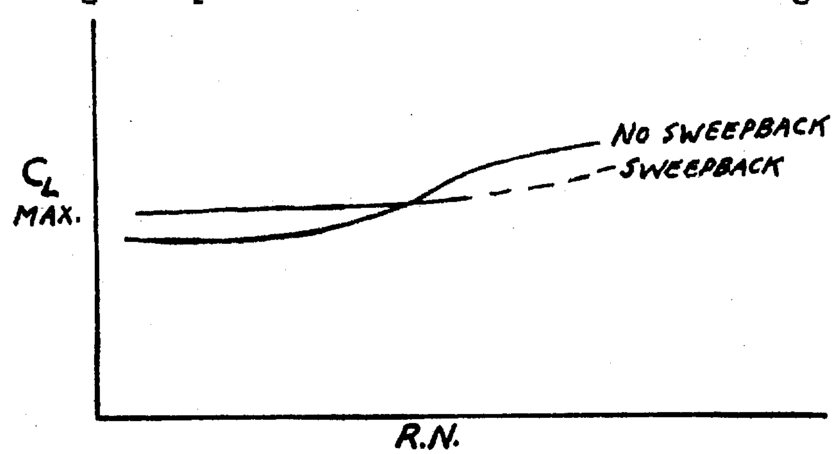
Ailerons and flaps are less effective on sweptback wings because they are subject to the "cosine law" mentioned above in conjunction with compressibility. A rough comparison indicates $C_{L_{MAX}}$ with flaps of 1.9 for a normal wing and 1.7 with a typical sweptback wing.

Theoretically the shape of the C_L vs α curve is reduced in proportion to the cosine of the angle of sweepback and the angle of attack for $C_{L_{Max}}$ is similarly reduced, so that reduction in $C_{L_{MAX}}$ is proportional to $(\cos\alpha)^2$. In practice, however, the loss is not as serious as this, as shown in the sketch. The reason was rather hesitantly given as being connected with the boundary layer transition.

B. SWEEPBACK



The Reynolds number effect upon $C_{L_{max}}$ with sweepback is somewhat as shown in the following sketch, with the dotted portion of the curve indicating unexplored and therefore doubtful region.



A major concern in the application of sweptback wings is the tendency for C_L distribution to peak at the tip (- i.e. produce tip stall). This is attributed to spanwise crossflow in the boundary layer, and it has a favorable effect on longitudinal stability below the region of stall because it results in a more rearward location of the neutral

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point in relation to the mean chord than in the case of a normal wing due to the more rearward location of the lift C.P. (The outboard concentration of lift is also aft because of sweepback).

It is interesting to note that washout of 6^{o} to 10^{o} had been found ineffective in preventing this tip stall tendency - general German policy was to use leading edge slots, although other methods will be discussed later.

A detrimental effect of tip stall (or of root stall on swept forward wings) is the airplane stalling moment created near complete stall due to the earlier separation of the aftmost portion of the wing.

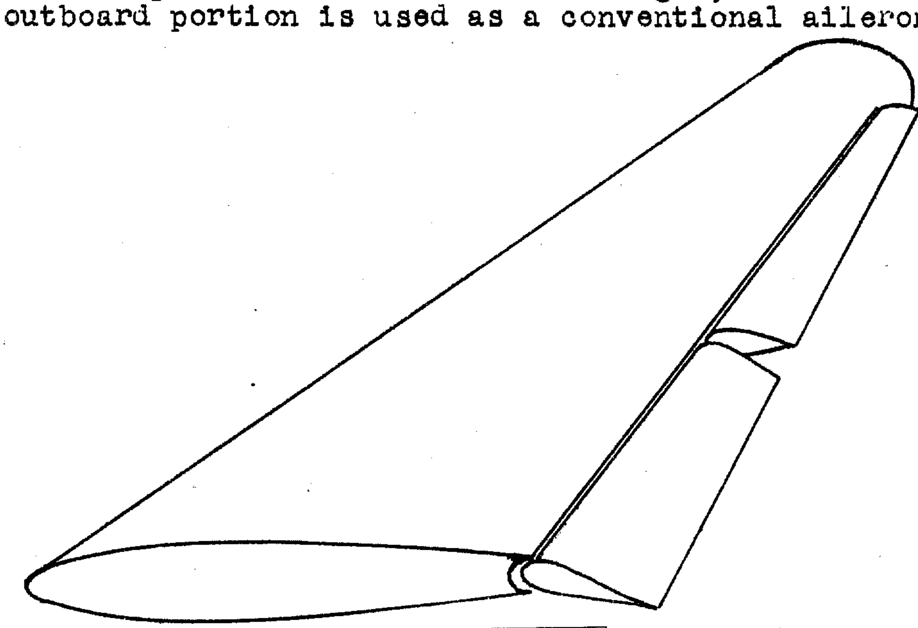
There are several structural difficulties entailed in sweptback wings:

Larger bending moments because the spar is longer for a given span.
Larger bending moments because of the high local tip C_L.
Very large torsion loads - especially at the fuselage-wing joint.

The theory of sweepback and its application was the subject of a conference held at Göttingen in 1941, and the papers presented and discussed were published as Lilenthal Bericht, 1941. In general, the whole gain anticipated by theory was not attained owing to difficulties of predicting the airflow characteristics at the wing tip and at wing centerline. Dr. Göthert of the Deutsche Versuchsanstalt der Luftfahrtforschung at Berlin investigated end effects on airfoil sections, having 0^{o}, 15^{o}, 30^{o} and 45^{o} of sweepback, up to a Mach Number of 1.0 and his results indicated that it was necessary to cut off the wing tips parallel to the center line of the fuselage and not perpendicular to the wind mean chord line.

Dr. Walchner at Göttingen had examined the aerodynamic characteristics of sweptback airfoils at supersonic speeds. The results of his investigations were distributed as an Untersuchungen und Mitteilungen Report in 1944, indicating that at Mach Numbers above unity, sweepback is beneficial down to very low aspect ratios and that the value of $dC_M/d\alpha$ is independent of the sweepback angle, providing $M \cos \phi > 1.0$. These characteristics are not valid at subsonic speeds.

The chief difficulty arising in conjunction with the utilization of sweptback wings, is the achievement of a maximum lift coefficient of sufficient magnitude. Since flaps become much less effective with large angles of sweep-back, the solution advocated by Dr.Kruger of Göttingen consisted of the combined use of normal flaps and a nose flap having a width equal to 10-20 percent of the wing chord and projecting downward from the leading edge at an angle of 30°. The normal flap alone could not be made sufficiently effective even with the introduction of suction. Dr.Kruger's tests were conducted on an NACA 23015 section with 45° of sweepback and a 2:1 taper ratio. A normal split flap gave a ΔC_{Lmax} of 0.4 while the addition of a 20% nose flap gave a ΔC_{Lmax} of 0.7 which produced a total lift coefficient of 2.3. There was, however, no noticeable improvement in wing dropping characteristics, inasmuch as the boundary layer still flows outwards towards the wing tip. An attempt was made to remedy this by incorporating a "differential split aileron" of the type shown in the sketch below. The low pressure region behind the downward deflected portion tends to induce the boundary to flow parallel to the line of flight, while the outboard portion is used as a conventional aileron.



Aerodynamic characteristics including pressure distributions resulting from measurements taken at 400 measuring stations are included in Dr.Kruger's work published as Untersuchungen und Mitteilungen Report 3155.

Investigation of the possible advantages that could be realized by applying the theory of sweepback to propeller design had been conducted at both AVA in Göttingen and DVL in Berlin with excellent agreement. Tunnel tests of constant thickness - chord ratio blades, 8% and 16%, with a diameter of 11.40 inches rotating at 30,000 RPM, gave a tip Mach Number of 1.30 and indicated that the peak efficiency could be maintained at a Mach Number 20% to 30% higher for 30° of sweepback than for the conventional blade. Dr. Quick, who conducted these investigations, had also developed a blade section which placed the center of gravity and the aerodynamic center of the blade on the pitch-changing axis. A complete description of these investigations has been published in AVA Report 43/F/08.

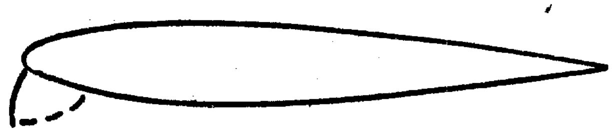
So far, only the general thoery and basic phenomena of sweepback have been covered. The following will attempt to outline some of the trends in German design of sweptback wings.

The problem of tip stall was receiving the most varied attention, in conjunction with attaining simultaneously good C_{LMAX}.

The most common approach was to use leading edge slots, and it was generally agreed that washout was of little benefit. A further advantage of slots was a possible increase in C_{Lmax} of 30% to 40% (actually obtained on Me-262 with full span slots). It is claimed (by Gropler of Junkers) that a good design of slot will not affect the critical Mach No. of a wing, although it does produce a higher drag below M_{cr} and consequently above M_{cr}. Dr. Lippisch states that the slots on the Me-163 affected its top speed less than 5%, and this effect could have been further decreased if it had been possible to maintain a close tolerance in manufacture on the slot gap on the top surface. This point is argued by Voigt of Messerschmitt and Pabst of Focke-Wulf. One reason given for not using our laminar flow airfoil section is that a leading edge slot spoils its effectiveness.

On tailless airplanes, the slots almost have to be fixed, because the operation of the elevons causes the stagnation point on the wing leading edge to wander, thus preventing the use of automatic slots.

An alternative to the slot, which has been promising in the wind tunnel, is shown in the following sketch:



Trailing edge flaps (or mid-chord flaps as on the Me-163) were usually used in conjunction with slots to reduce the angle of attack at which C_{Lmax} occurs. With increasing amounts of sweepback, the C.P. of the lift produced by a lift flap moves forward relative to the C.G. and tends to reduce the moment change, thereby allowing flaps to be located at the trailing edge where they are most effective for lift.

Messerschmitt tried an Me-163 without its normal wing structural twist of 5° but with up-turned elevons. The behavior in flight and in the wind tunnel at both low and high speeds were almost identical to the original Me-163.

The Horton Brothers, who have been working on tailless airplanes and gliders since 1930, have never used leading edge slots, although they have used as much as 7° twist on some of their gliders. On their

B. SWEEPBACK

recent ships including the Horton IX, a jet powered fighter, they have caused the stall to occur at the middle third of the semi-span. This was done by designing effective washout at both root and tip (only $1\frac{1}{2}^{\circ}$ at the tip) with respect to the middle third of the semi-span. The chord at the root was increased also, effectively, reducing t/c. This procedure gave a very well behaved stall with no change in pitching moment or danger of wing dropping. However, a C_{Lmax} of 1.3 was the highest obtained on the Ho-IX. This was with flaps.

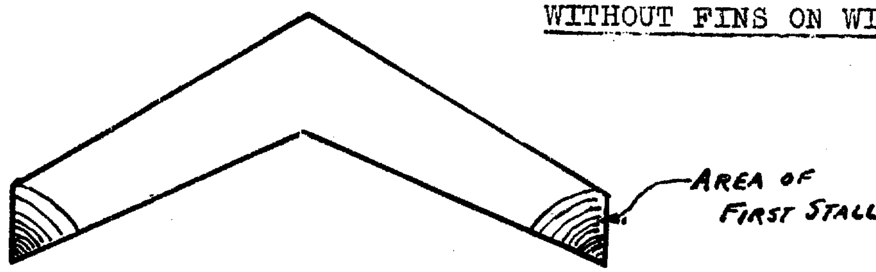
Although Messerschmitt was still using leading edge slots on their latest sweepback wing for the 1101 (24% C slot at tip and 13% C at root) they also used a thicker section at the tip than at the root. This was done partially to reduce interference of wing and fuselage, but also partially to help the tip stall situation. Pabst of Focke-Wulf recommended a middle path - make the wing a constant t/c and use very little planform taper. As a side note, he also recommended that the inboard 25% of the semi-span outboard of the fuselage be considered to have no sweepback when calculating the performance of an airplane with sweptback wings.

According to tests made at LFA, sweepback of a short wing (semi-span equal to chord) had only $\frac{1}{4}$ to 1/3 the usual increase in critical Mach no. - no doubt due to the center interference already described. But Dr. Lippisch, with his very low aspect ratio triangular wing has attained an $M_{cr} = .9$ with a root section of t/c = 13.6%. (This was in combination with a fuselage). And he further claims that this low aspect ratio eliminates the problem of tip stall, in spite of the very sharp tips. He could not give an explanation, except to say that with his wing planform, the flow is completely three dimensional - and since no theory exists that predicts this flow, only tests can be used to find its behavior. It is also known that the L/D of this wing is definitely better at high Mach numbers (M = 1.2 and up) than the high aspect ratio wings. Low speed C_{LMAX} is in the neighborhood of 1.0. Lippisch was not worried about the high angle of attack that goes with C_{Lmax}

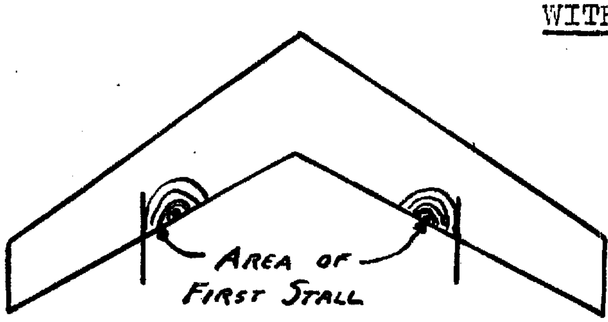
at low aspect ratios because the ground effect increases effective aspect ratio sufficiently to make the landing angle practical.

Junkers had still another approach in the use of sweptback wings. On their EF 128, proposed tailless fighter, they had two vertical tails each centered on the aft portion of the wing at a point half the semi-span outboard of the fuselage. This procedure was said to stop the spanwise boundary layer flow along the wing and transfer the stall point from the tip to a point just inboard of the fin. The explanatory diagrams by Gropler, Chief Designer, are shown below.

WITHOUT FINS ON WING



WITH FINS ON WING



As a final note on sweepback, the use of a sweptback tail, both horizontal and vertical surfaces, can be mentioned. Both Focke-Wulf and Messerschmitt were working on airplanes with sweptback wings and sweptback tails. Their recommendations for design of the tail agreed - the tail surfaces should be thinner than the wing (Messerschmitt used 6% thickness ratio) and the sweepback should be the same as, or greater than, that of the wing.

2. AERODYNAMICS

C. SWEEPFORWARD

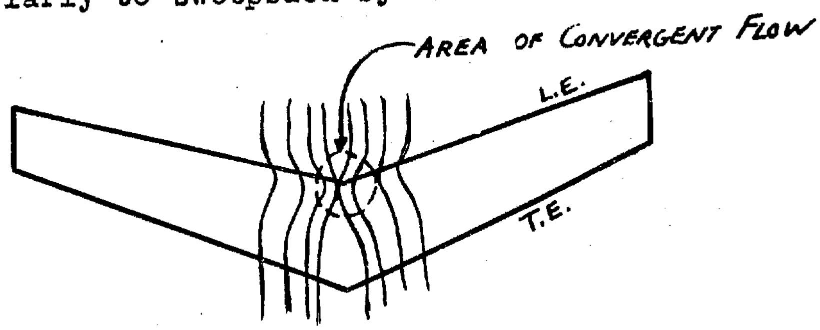
As an answer to some of the problems of sweepback, Junkers had developed sweepforward and was incorporating it in the Ju-287, a six-engined jet bomber.

The claims made for sweepforward are:

(1) The sweep angle has the same effect of increasing the critical Mach number as an equal angle of sweepback.

(2) There is no tendency for tip stall - initial stall occurring at the innermost portion of the wing since the spanwise flow is inboard rather than outboard.

(3) There is less interference at the wing center for high speed. This may be explained similarly to sweepback by the sketch below:



Although there is an appreciable area of interference, as for sweepback, the flow is at a lower speed and should not affect M_{cr} as much as for sweepback.

The elimination of tip stall allows higher taper ratios in both planform and thickness and moves the required leading edge slots from the tip to the root of the wing.

C_{LMAX} may tend to be lower than for a comparable sweepback because the inboard region that stalls first is a larger percentage of the total wing area than the "first-stall" area is on a sweptback wing. Comparative wind tunnel tests on four wings, 30° sweepforward, 0° sweep, 30° sweepback, and 45° sweepback, indicated that sweepforward had the lowest C_{LMAX} and 45° sweepback had the highest C_{LMAX}. These tests were at low Reynolds number and, according to the Reynolds number correction shown in the preceding section 2 B, the full scale flight comparison may be completely different. Junkers was expecting a C_{LMAX} of 1.8 on the Ju-287.

Sweepforward has similar stability characteristics both before and after "first-stall", because the lift centers are similarly located on the mean aerodynamic chord.

There was some thought that sweepforward would have a structural disadvantage because of the fact that at high C_L or high g, the inner part of the wing could be stalled thereby moving the spanwise center of load outboard with a consequent increase in bending moment.

So far as is known, no tailless or tail first airplanes had been designed in Germany using swept-forward wings. It would seem that a normal tail-in-rear airplane might offer complications when sweepforward is used, since the horizontal tail is closer to the root section of the wing than for either normal or sweptback arrangements (assuming equal tail lengths to the c.g.).

2. AERODYNAMICS

D. Methods of Control at High Speeds

In general, German aircraft engineers have stuck with conventional aerodynamically balanced control surfaces on the airplanes built to date. However, they were becoming aware of high speed control difficulties and were studying several alternate types of control for use on proposed and experimental airplanes.

The normal practice was a larger threw of surface (Messerschmitt had 20° deflection of the surfaces on the Me-262) and as might be expected, variation in gap tolerances greatly affected the control behavior. Until very recent designs stick forces were maintained at desired levels by increasing the amount of aerodynamic balance.

Messerschmitt was considering three alternatives to increasing aerodynamic balance for keeping down the stick forces.

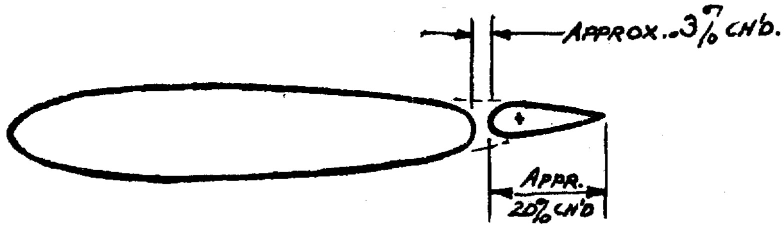
1. Using a sealed balance similar to NACA design but vented to a point further aft on the running surface to get higher unit pressure for more effective boost.

2. Hydraulic boost or electric servo system.

3. Using Flettner servo tabs in which a small control surface was direct connected to the cockpit controls to provide feel. Planned to provide a disconnect lever on the control stick to disconnect and lock the direct connected surfaces at very high speeds.

Mr. Messerschmitt did not like the hydraulic boost system because he considered it too complicated and heavy. However, the Flettner system had given indications of flutter trouble, according to Mr. Voigt.

Both Dr. Lippisch of Vienna and Mr. Multhopp of Focke-Wulf recommended using normal aerodynamic balance, but rounding off the trailing edge portion of the wing (or other surface) just forward of the control surface. The recommendations of Lippisch are shown below:



This configuration allows high deflection of the control surface without danger of control reversal due to compressibility shock forming on the leading edge. In other words, a comparatively smooth flow is maintained even at high surface deflections. Stick forces are maintained by the amount of aerodynamic balance.

Lippisch was using this arrangement on his LP-13 which was intended to be a super-sonic airplane.

The Horten brothers have used many and varied types of control on their tailless gliders and airplanes. These are outlined below.

1. <u>Longitudinal and Lateral Control.</u>

The first Horten aircraft (the H-1) was fitted with one trailing edge control surface on each wing which acted in the usual way as an elevon giving longitudinal and lateral control. Landing flaps were not fitted.

On the H-11, two trailing edge control surfaces were fitted on each wing and were coupled so that the outboard surfaces give primarily lateral control and the inboard surfaces primarily longitudinal control. Owing to the kinematics of the linkage system, however, slight displacements of the inboard surfaces occur when the stick was put to the left or right and slight displacements to outer surfaces when the stick was put forward or back.

The H-111 had two pairs of control surfaces working as on the H-11 and in addition was provided with landing flaps on the innermost section of the wing.

On the H-IV, the controls were still further split up. There are three movable control surfaces on each wing. The outermost of these surfaces give primarily lateral control by a symmetrical motion when the stick is put to the left or right, but they also deflect slightly when the stick is pushed forward or back. The middle surfaces are capable of a large deflection upward but only a small deflection downward, while the innermost surfaces are capable of a large deflection downward and only a small deflection upward. By suitable coupling between the control system and the control surfaces it is so arranged that the deflections of the three controls for a given stick position increase or decrease progressively along the span. Thus, on a down-going wing, the outermost control will have the greatest deflection, the middle control the lesser deflection and the innermost control the least deflection; while on an up-going wing, the innermost control will have the greatest deflection and the outermost control the least. By this means the washout of the wing is maintained even when the controls are displaced.

With the H.V a reversion to the two-control system was made. On this aircraft the outer controls are used to produce a nose up pitching moment, while the inner controls are used to produce nose down moments. The relationship between the deflections of the two pairs of surfaces is arranged to keep the wing twist constant whatever the amount of control applied. By using the elevating control on one side and the depressing control on the other an aileron effect is obtained. It is claimed that by this use of the controls, no adverse yaw is caused.

The H.V is provided with three flaps for take-off and landing. These are situated on the centre section. The part of the flaps between the engines drops to 60°, the parts just outboard to the engines to 40°, while on the original version of the aircraft, the inner control surfaces were arranged to drop 30° while still retaining their function as elevators giving nose down moments.

The control systems on the H-VI and H—VII are respectively similar to those on the H-IV and H-V.

The Hortens appear to have been satisfied with the use of two pairs of control surfaces in this way because the same scheme has been adopted on the HVIII and HIX. The maximum control deflections are said to be as follows:-

Control column position	Outermost control, port wing	Middle control port wing	Middle control starboard wing	Outermost control starboard wing
Left	20° up	2° up	20° down	2° down
Right	2° down	20° down	2° up	20° up
Forward	5° up	30° down	30° down	5° up
Back	30-45° up	5° up	5° up	30-45° up

With regard to control shapes, the final practice seems to be the use of a Friese nose on the outermost control (which functions mainly as an aileron) usually with a large amount of forward balance. The inner controls have usually blunt rounded noses or a very blunt nosed Friese section and a combined trim and balance tab is normally fitted. On the H-IV the Friese aileron is mounted on a skew hinge whose effect is to cause a large projection of the nose below the wing surfaces when the aileron is displaced upward. This helps to give a favorable yawing moment.

2. Directional Control

On all the Horten aircraft, directional control is provided by drag rudders operated in the usual way. None of the aircraft are fitted with vertical fins. On the H-1, II, III and V, the control is situated on the leading edge near the wing tip. The portions of the wing surfaces between the stagnation point and the main spar are arranged to pivot outwards about a hinge line approximately at the stagnation point; this gives a drag force on the appropriate wing.

On the H-IV and H-VI a plate type spoiler is provided on the top and bottom surfaces of each wing between the ailerons and the front spar. The plates project when the rudder bar is deflected. A similar system is believed to be used on the H-VIII.

On the H-IX, this development has been taken a stage further. The installation consists of a large and a smaller span spoiler on the upper surface and a similar pair on the lower surface just behind the main spar near the wing tip. Spring links are incorporated in the control system so that as control is applied the two smaller spoilers on one side appear first and are followed by the two large spoilers when the former are fully deflected. This is, of course, to give smooth and progressive control at high speed combined with adequate control at low speeds.

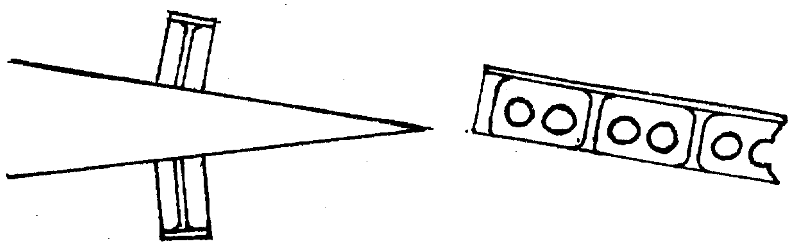
The H-VII was stated to be fitted with an entirely different type of directional control. A wooden bar is mounted on a system of rollers behind and parallel to the main spar at each wing tip. Displacement of the rudder bar causes the bar on one side to project in a spanwise direction from the wing tip causing a drag on this side. This control was said to be very pleasant and satisfactory in flight. It has the advantage over the other types of drag rudders that it does not blanket the effect of the ailerons.

The use of a movable wing tip to give lateral control has been actively investigated by the Horten Brothers. In short, the whole wing tip is hinged on a skew axis so that it can rotate forward with a decreasing incidence or backward with increasing incidence. A geared tab may be provided to balance the loads on this control.

The Horten III was first modified to take this control but the flight results were not satisfactory. The pilot reported that he was unable to hold the stick which was threshing about fairly violently in all directions.

The plastic version of the H-V was built with an improved form of this control but as this aircraft crashed on its first flight, no information is available on its behavior. The Hortens had not given up hope of making this type of control satisfactory.

The section of the spoilers as used on the H-IX is as shown in the sketch. When both rudder pedals were pressed simultaneously, the spoilers on both wings operated and acted as drag brakes.



The amount of venting varied the pedal force, Hortens, on the H-IX obtaining 1 kg pedal force at 150 km/hr. for operation of the spoilers. This force varied only slightly with speed.

On the H-IX, a novel arrangement of the control column allows an increase in mechanical advantage for high speed flight. Unlike the simple telescoping stick sometimes tried before, this device uses a constant overall length control column, but shifts the pivot point. This increased leverage is at the expense of a decrease in maximum available control surface deflection, but this is of no importance at high speeds.

Dr. Seiferth of Goettingen indicated that recent tests using extendable wing tips of thin cross section appeared to have considerable merit and resulted in a rolling moment somewhat greater than would be calculated on the basis of area alone. This he attributed to the change in the spanwise distribution at the tip, resulting from the change from a rounded to a rectangular tip shape. A plate hinged at the leading edge appeared to be better than one hinged near the trailing edge. This type of aileron is superior to the conventional type from the flutter standpoint.

Dr. Göthert has made a considerable number of experiments at DVL in Berlin on the use of spoilers for control and braking devices at high Mach numbers. The normal type of ailerons at various chord ratios were also investigated. In the critical speed range from M = .75 to .9, Göthert stated that small spoilers up to 2.5% chord located at 90% of chord aft of the leading edge were very effective and appeared to be preferable to the normal aileron. Spoilers at any location forward of this gave poor characteristics. Even a complete reversal of moment coefficient is possible.

Another type of spoiler is a flat sheet which can project out through the fin or wing surface at a point between 10% and 50% aft from the leading edge and reduce the lift there. This type has been tested in a wind tunnel at LFA by Dr. Zobel. The idea is old, but he developed an improvement, by providing an air duct, or slit, from the stagnation point to the point just aft of the spoiler. This feature greatly reduces the time lag, between raising the spoiler, and the effect of this on the lift of the airfoil. (The spoiler considered here is hinged.) The spoiler with the duct is on the upper side of the airfoil located about 10% of the chord length from the leading edge. A similar spoiler, but having no duct is located on the lower side of the airfoil about 40 to 50% aft of the leading edge. The duct width is about 1% of the chord length. The spoilers are used primarily for ailerons, in place of movable ailerons. Tests of various shapes of spoilers - notched, screen arrangements, saw-toothed, etc. - were made, but the simple. solid plate type was chosen as the best by Dr. Zobel.

The Germans were developing a code for evaluation of flight stability and control. This report was written by Doetsch of DVL and had been circulated to the aircraft manufacturers for comment. Much of it had not been adopted by the manufacturers yet and it was not an accepted regulation, but only in the formative stage.

Dynamic longitudinal stability could not be divergent. Static longitudinal stability was regulated by stick force rules. When a plot of stick force versus speed squared is extrapolated to zero speed, the zero speed intercept in kilos must lie between $2 + 2.2 \log_{10} W$ and $10 + 9 \log_{10} W$, where W is the gross weight in kilograms. (Stick force is also in kilograms.) This must be met with any C.G, and when trimmed either at .8 climb speed or 1.2 maximum level flight speed. There was no stick fixed requirement.

3. AIRPLANE DESIGN NOTES AND PROCEDURES

Quite a number of miscellaneous facts and ideas concerning airplane design and design problems have been picked up in Germany that are either difficult or too much isolated to classify under separate discussion headings. These are gathered here without much attempt at classification.

A. Use of High Speed Wind Tunnels

The German aircraft industry had available quite a few good high speed tunnels and much use was made of them. These tunnels covered higher Mach numbers than those available in America; .95 M being quite common and M ¥ 6.0 being available. Most of these tunnels are short duration types - energy being stored over a comparatively long period of time and expended in a very small period. With ingenious measuring devices, the Germans were able to get all necessary readings in the very small period of time that the tunnel ran.

Of most importance is the fact that all recent German experimental and proposed airplanes were based on a good deal of high speed wind tunnel data on the specific airplane design - and many of them had been altered and changed on the basis of this data.

An interesting German practice is the use of half models in most of the smaller tunnels. These models - half a fuselage and only one wing panel - effectively double the size of the tunnel.

B. Water Tunnels

The water tunnel is a useful and simple aid in the design of high speed airplanes. A normal wind tunnel type section, with water flowing instead of air, is used to determine the critical points on a model from the compressibility viewpoint. (Of course, the model can be towed beneath the surface of a water tank also.) Points of cavitation on the model indicate locations of compressibility burbles at high speed on the actual airplane. Also, the border of the cavitated area shows the shape of the fairing required to eliminate the compressibility burble at that point.

This latter fact is explained as follows. The water in the bubble, apart from being turbulent, is stationary or is not flowing with the main stream and there is no definite flow from any one part of the bubble to another. Therefore, the pressures within it are constant at all points, and the pressure at all points on its outer boundary is the same.

Thus, water tunnel tests not only indicate the critical drag points on a high speed airplane, but also give a method of fairing these points.

C. Jet vs. Propeller

Messerschmitt's ideas on the use of jet power are as follows:

1. For daylight fighters, no plane using a reciprocating engine and/or propeller would be satisfactory. Such a plane cannot compete with jet power in speed and light weight for short range and could not take off and intercept a fast bomber as effectively as a short range jet fighter.

2. It was not believed that large heavily armored bombers of the type now in use would prove practical in the face of an adequate force of high speed jet fighters. Nor would long range fighters be able to match the speed and climb of short range jet fighters. Consequently, design studies favored jet powered bombers capable of flying in and out of the target area at high speed.

3. For a night fighter, requiring more range and endurance for locating enemy craft, it is felt that the plane powered with a reciprocating engine still has some use. Preferably, however, the plane should be powered by a gas-turbine-propeller power plant to assure sufficient high speed to cope with jet bombers.

D. Athodyd Power-Effects on Design.

Dr. Lippisch, who was designing high speed athodyd powered airplanes, feels that athodyds must be designed as part of the airplane--not as a unit power plant to be installed in the airplane. He believes this is the only way that design versatility and variety can be obtained--any size airplane designed for any performance.

In conjunction with this design policy, Lippisch had used two equations as a basis for obtaining a comparison of efficiencies of designs from both the drag and power plant viewpoint.

The first factor concerned the installation of the athodyd or Lorin Power plant and compares the success obtained in maintaining maximum internal volume for a given surface area. The second factor compares the actual surface area to the absolute minimum that can exist.

1. Coefficient of Surface $= \frac{\sqrt[3]{V}}{\sqrt{S}}$

This coefficient must be as great as possible for best athodyd installation.

2. Coefficient of Drag Efficiency $= \frac{2A}{S} < 1$

This also should be as high as possible to keep surface drag a minimum.

In the above equations,

V = volume
S = surface area
A = wing area

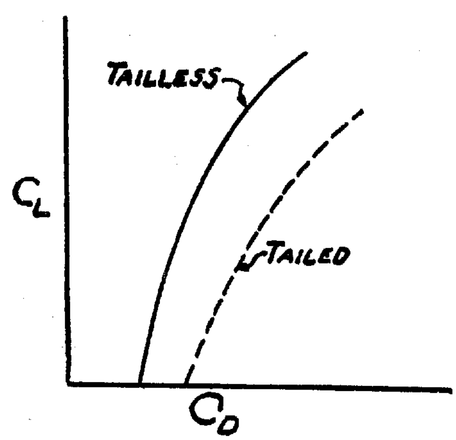
E. Tailless vs. Tailed Airplanes

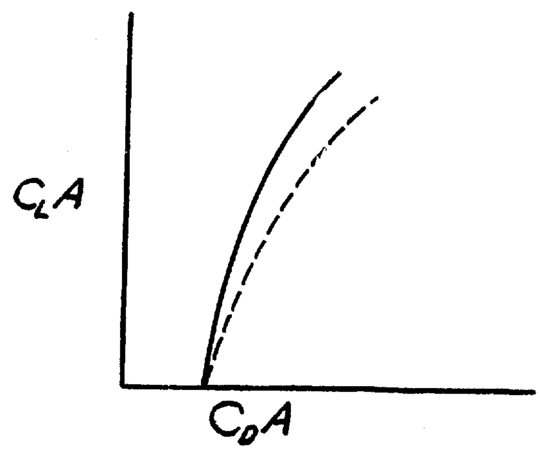
Although the use of sweepback was, in general, prevalent throughout the German airplane designing brotherhood, there was some resistance to its full use in making the airplane tailless--a flying wing. Both Messerschmitt and Focke-Wulf were working on airplanes with sweptback wings and sweptback tails. Both had a similar reason; they felt that, for flight at high Mach number, the tailless airplane was unsatisfactory because it lacked sufficient control and damping in the pitching plane to counteract pitching oscillations. It should be noted that they are specifically referring to the region just beyond M_{cr} where violent changes in trim occur.

Dr. Lippisch takes an opposite view. He believes that no tailed airplane has yet stopped the violent instability that occurs above M_{cr} and further states that the flow at the tail for these Mach numbers is so indeterminate and so impossible to calculate that no tail ever could be designed to give the desired controllability. Also, he believes that the most logical means of controlling the sudden changes in pitching moment is to control the wing airfoil shape as is done with the tailless airplane.

But, probably of most importance, is Lippisch's statement that the unstable region above M_{cr} should not be flown through in level flight, but rather in a zero lift dive (or climb) so that there need not be any moment change (if a symmetrical airfoil isused).

The Messerschmitt company made a study that almost convinced them that a conventional wing and tail airplane had very little disadvantage compared to a swept-back wing tailless airplane. The study was based on maintaining a fairly low landing speed for both arrangements, and their conclusions were drawn from the following polar curves.



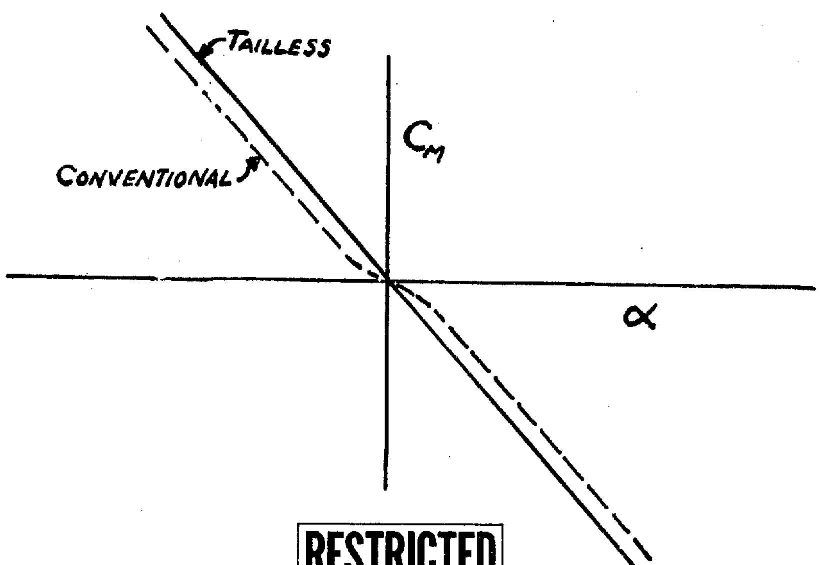


The tailless airplane has a lower drag coefficient, but its wing area is greater because of the lower $C_{l_{max}}$ used when determining landing speed ($C_{l_{max}}$ = 1.3 for tailless and 2.0 for tailed). Consequently, the drag, C_DA, is the same for the tailless as for the tailed at low C_L, but less for the higher C_l. Their conclusion was, therefore, that the tailless airplane was no better for high speed, but did have the advantage for range, climb, and ceiling.

This is aninteresting study and seems especially applicable to airplanes with high speeds in the region of $M = .75$ to $.8$, the maximum M_{cr} that can be expected without sweepback. But above that point, say at a $M = .9$, the drag comparison breaks down because the normal wing would be above M_{cr} and the sweptback tailless wing would still be below M_{cr}. And if the wings of the tailed airplane were swept back to correct this discrepancy, the tailed airplane would lose its $C_{l_{max}}$ advantage. Further, on the basic comparison, the $C_{l_{max}} = 1.3$ seems a little low for the sweptback wing and $C_{l_{max}} = 2.0$ seems a little high for the normal wing, but it is interesting to note that even on this basis, the flying wing has an advantage--especially if it is for a bomber configuration.

It should be noted that Focke-Wulf's sweptback wing and tail airplane, the FW 183, was to be flown first with normal control surfaces; it was intended at a later date to change to flying wing procedure, i.e. elevons, and to use the horizontal tail only for trim.

Dr. Lippisch claimed a directional stability advantage for the tailless airplane, if it is designed correctly. This is due to the fact that its short fuselage and comparatively large vertical tail makes it easy to maintain a constant center of pressure point throughout the entire range of yaw. Almost all normal airplanes have a snaking tendency at 0° yaw, while Lippisch's Me. 163 did not, and therefore was considered a very good gun platform. A comparison is shown below.



Dr. Lippisch was very strong in his insistence that flying wings require vertical tail surfaces, in spite of the fact that the Horton brothers never used a vertical surface. His point is supported by the fact that Horton's latest, the Ho IX, although stable, took four times as long to damp out directionally as a normal airplane did. Hortons claimed that this was no disadvantage because the oscillations were so long that the plane was more comfortable in gusty weather and also it made a better gun platform.

F. Directed Missiles vs. Interceptors

Dr. Wurster, who designed the Enzian 4 and 5 and developed them at Messerschmitt, was very insistent in his belief that interceptor fighters would soon be unable to stop high speed bombers. He based this belief on the fact that there is a speed wall at somewhere around $M = .9$ beyond which it is impractical for an airplane, as such, to go. Although it may be true that there is a point beyond $M = 1.2$ where the drag coefficient drops down to low speed values, the region between that point and M_{cr}, say $M = .9$, is such that a minimum of 6 times the power required to fly at $M = .9$ is necessary. Even if the low drag coefficient region above $M = 1.2$ could be reached, for instance by diving or by assist rockets, the steady power required to maintain flight would be approximately twice that required to fly at $M = .9$. Based on these facts, Wurster believes, that no practical airplane will be built to fly much above M_{cr}. But there is no reason why bombers cannot be made to fly at that speed almost as easily as fighters.

Consequently, the time will come when the fighters, or interceptors, will have very little speed advantage over bombers, and their job of taking-off, climbing, catching and downing the bombers will become most difficult. And anti-aircraft guns are even now inadequate because of their comparative inaccuracy and small size of shell.

So, according to Wurster, the time is very near when the Flak rocket (or directed missile) will be the only thing capable of stopping the bombers.

This can be done (although it has not yet been done in Germany) because:

1. The directed missile can be made to fly at supersonic speed because of its small required range and its small size and weight. (No pilot, equipment, or maximum weight requirement for landing purposes) Therefore, it can

catch the bombers easily.

2. With the infra-red directing head, which is sensitive to exhaust gases, the missile will be unerringly directed to the target (the bomber).

3. Because of its simplicity, it can be made much cheaper than an airplane, even cheaper than an anti-aircraft shell.

4. It can be launched from almost any place, no airport is required. And the launching point can be moved more easily than an anti-aircraft gun emplacement.

Further, Wurster emphasized that the Flak rocket directed missile should carry a good sized warhead (1000 lbs. or so) so that, with a proximity fuse, a direct hit will not be necessary.

G. Nacelle Location on Wings

Junkers made a series of tests varying the chordwise location of a typical jet nacelle on a wing to find the best location from a drag standpoint. Although the tests were run in their rather small high speed wind tunnel, the results are generally accepted throughout the German aircraft industry. (Their findings were mentioned by both Multhopp of Focke-Wulf and Voigt of Messerschmitt).

The conclusions were that the leading edge of the nacelle should either be far forward of the leading edge of the wing (at least 50% of the wing chord ahead of the wing leading edge) or should be aft of the 50% chord point on the wing. Not much preference was shown whether the nacelle centerline was above or below the wing chord line, although Junkers was designing a flying wing bomber with the nacelle leading edge at 50% chord and the whole nacelle above the wing. In general, the aft location of nacelle (50% chord) seemed to be preferred, especially for sweptback or sweptforward wings. This location has the least effect on spanwise flow, but it is poor from the flutter standpoint.

Dr. Kuchemann of the Kaiser Wilhelm Institute at Göttingen has made a detailed study of the fitting of nacelles to wings and had placed special emphasis on gas turbine nacelles. He was of the opinion that the position of the nacelle, whether highwing, midwing, or lowwing, fore or aft was not particularly vital, but that the fairing was far more important. He indicated that extreme fore or aft locations were best from the drag viewpoint but were the most difficult to fair correctly into wing.

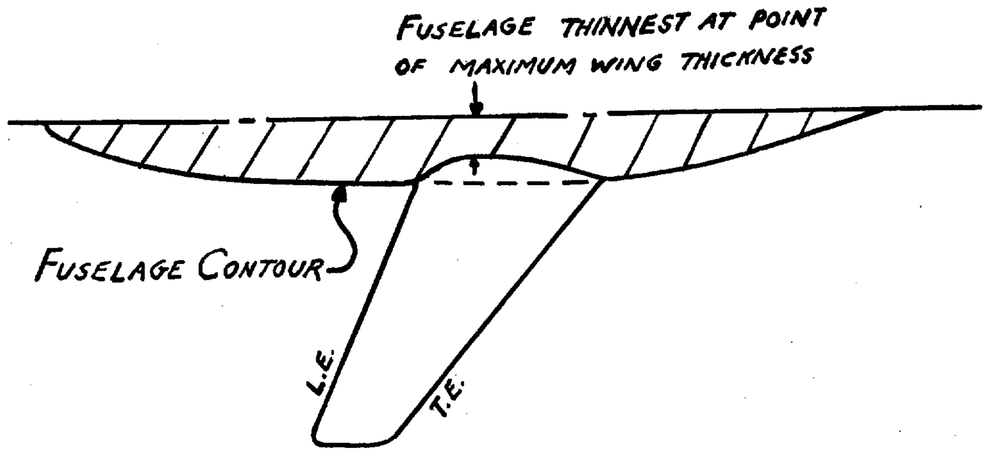
For supersonic entries, Dr. Oswatitch and Dr. Busemann, advocate the utilization of a central pointed needle which projects through the duct entry, and creates a series of shocks ahead of the nacelle. The actual device proposed consisted of a conical point with a re-entrant angle which creates two oblique shock waves which are tangent to the leading edge of the duct entrance. Wind tunnel tests indicated that at Mach Numbers of approximately 3.0 the recovery was nearly 95 percent and that the duct flow was good up to a Mach Number of 3.2.

In considering duct entries, Dr. Kuchemann calculated pressure distributions from a ring source and from vortex rings. High speed wind tunnel tests conducted by Prof. Ludweig at Reyershausen, on Kuchemann's duct entrances indicated that these designs with a high peak suction near the lip are better at high Mach Numbers than those designs that avoid the peak, but on which the suction further aft is greater. A report by Dr.Kuchemann and Dr. Webber in Deutche Akademie for 1943 and Dr. Kuchemann's Untersuchungen und Mitteilungen Report Nr.3125 explain the detailed results of these investigations. Results of the analysis of symmetrical and unsymmetrical duct entries located on a symmetrical wing have been compiled in a Untersuchungen und Mitteilungen Report by Dr. Schirer issued in 1944.

H. Fuselage Shape at Wing Joint

Another result of Junkers high speed tunnel tests of interest is the fuselage-wing junction shown below. This was claimed to have raised the critical

Mach number from .75 to .78 compared to a normal junction.



I. Jet Intake Duct Design

Wind tunnel tests for Messerschmitt indicated the desirability of gradually decreasing the duct cross section as it neared the compressor intake (axial flow, high speed air intake). This minimizes the growth of the boundary layer within the duct.

It was found by Prof. Busemann of L.F.A. that in any axially symmetric compression without a central body one always gets a normal shock wave with a resulting decrease in efficiency. It is therefore necessary to have such a central body which is designed so as to obtain a compression initially away from the axis instead of toward it. If a suitably long central body is used, no normal shock is obtained but rather a series of oblique shock waves with considerably less total energy loss.

The same applies to an entrance to a duct in supersonic flow. Otherwise, there will always be a normal shock wave just a head of the duct. Consequently, a central spike is recommended to be used ahead of the duct entry of, say an athodyd, for supersonic operation.

J. Nose Shape for Supersonic Speeds

According to Guderley of L.F.A., the nose should be flat and square for minimum wave resistance. The square surface, being normal to the flow direction, is followed by an arc of constant pressure with a corner at the junction. It seems that the high pressure in the flat nose is more than counterbalanced by the large suction due to expansion around the corner. Guderley was planning to calculate the shape of the body nose section, but had not gotten to that point yet. In the meantime, L.F.A. favored a nose shape of 10 caliber radius tangent ogive for supersonic missiles.

K. Effects of Jet Wake

Dr. Braun of L.F.A. found that the effect of the rocket blast must be included in the calculation of the stability of flak rockets just after launching. The rate of change of the mass of the rocket must be taken into account in the moment of inertia for longitudinal stability since the mass is decreasing due to ejection of gases in the jet. Hence, the effective moment of inertia for a pitch (or yaw) angle of is:

$$\frac{d(Id\theta/dt)}{dt}, \text{ not merely } \frac{Id^2\theta}{dt^2}$$

where I is the static moment of inertia t is the time.

Also, results of tests made at Aachen on a V-2 model showed a 70% increase in the form drag of the body, due to the effect of the jet on the flow of air, over the tail. This was based on comparative pressure measurements over the body, excluding the cross-section occupied by the jet. The range of conditions was very limited, however, so the result may not be representative of other cases, particularly if the ratio of jet cross-section to maximum cross-section of the body were much different.

According to Mr. Kerris of L.F.A. there is a region in the jet wake through which radio signals will not pass due to ionization. The lateral extent of this region and how far it extends behind the rocket vary greatly with the design and operating conditions of the rocket. The ionization is an exponential function of temperature, so it varies considerably with the temperature at which the jet gas is ejected. Higher temperatures make it worse.

L. De-Icing of High Speed Airplanes

Voigt of Messerschmitt claimed that de-icing installations were not necessary for high speed airplanes because the heat of compression due to the high speed is sufficient to prevent icing. On the Me-262, about 27°C rise in temperature had been measured from this cause and no wing de-icing was used nor were gun heaters used.

M. Tailplane Arrangement

Messerschmitt had test flown an Me-109 with a Vee tail and found it to be satisfactory. It was considered a desirable arrangement for high speed aircraft because the number of junctions and the likelihood of compressibility trouble is reduced.

Dr. Lippisch was seriously considering making the vertical tail on his supersonic airplane symmetrical about the centerline of thrust, i.e. half the tail above and half below the fuselage. He favored this arrangement because he thought there would be less chance of a moment change occurring in a dive above M_{CR}. Several airplanes had this type of tail, such as the "Natter" and the Dornier 335.

There was some thought among the Flak rocket designers that two sets of swept back wings at 90° to each other should be used to improve stability and controllability over one set with a conventional vertical tail (or tip fins). However, L.F.A. tests showed the double winged arrangement had 30% more drag, 20% less speed, 10% more weight, and half the payload of the single wing type, offsetting the advantages mentioned.

N. High Lift Devices - Boundary Layer Control

Dr. Regenscheit of the Kaiser Wilhelm Institute at Göttingen had been experimenting with the application of suction at the flap as a means of increasing lift. Tests were made on an NACA 23012 rectangular section of aspect ratio of 2. With a trailing edge angle of 30°, a change in zero lift angle of 3.3° was obtained with a C_w = .015. The effect was found to decrease with decreasing trailing edge angle and the optimum S/C ratio is approximately 1%. The advantage obtained with a given suction increases with the thickness of the wing section and Regenscheit's investigations also included examination of a wing section having suction applied to a flap along the center section, and air exhausting over a slotted flap on the outer portions. This combination produced a maximum lift coefficient of 12 with a C_w of 0.06 and is discussed in detail in AVA report 41/4/14.

Regenscheit examined the possibilities of increasing lift by the application of suction above the trailing edge. Actually, the slot is formed by the omission of 1.25 - 1.40% of the chord of the upper surface at the trailing edge. This configuration moves the stagnation point aft with the application of suction and increases in C_{Lmax} of 1.0 were obtained for a C_w of .012 on a NACA 23012 section. To obtain self induced suction he utilized the core of the training vortex by installing louvres in the wing tip. Regenscheit had also duplicated the slit, one above and one below the trailing edge as a control device, but it was found to be too sensitive for aircraft use and the tip vortex suction was not powerful enough for practical application.

A considerable amount of work has been done at various German research institutes in connection with boundary layer control and these programs included systemmatic investigations of a wide variety of airfoil sections, flaps, slot positions, slot shapes and size, volume of air, etc. In general, the airflow required was impractically large (C_w = 0.02 for C_{LMAX} = 4.0) and it was the general opinion that the only possible application was to very large aircraft where the complexity and weight of the equipment required might be a small enough portion of the total to be justifiable.

Actual flight tests were conducted by Professor Stuper at AVA on two aircraft equipped with suction wings. The Absauge - Flugzeug 1 (AF-1) was built in 1932 and had a $C_{L_{max}}$ of 5.5. The initial tests were made with a floating aileron which was relatively ineffective and was replaced by a special aileron developed by Herr Groppler of Junkers in which the rear portion of the double flap was used for lateral control. The AF-2 had a normal Storch (Fi-156) fuselage, empennage and power plant with special wings constructed by Grunau. A maximum lift coefficient of 4.4 was obtained with this aircraft, which was lower than the AF-1 due to the inadequate capacity of the blower. However, for the same C_w, the corresponding C_L was higher than for the AF-1. A complete report (AVA Report FB-1821) includes a comparison of the characteristics of these aircraft with the original Storch and the results indicate that while higher lift coefficients were obtained, the lateral control was definitely inferior. Dr. Kruger has written a comprehensive report on all available high lift devices which was published as Untersuchungen und Mitteilungen Nr.3025.

4. STRUCTURAL DESIGN CRITERIA

A. General

Part of Germany's success with high speed airplane design is due to the flexibility and logical philosophy of their structural requirements. They did not design for regions of high speed flight where the characteristics of the airplane were unknown and unpredictable. Rather, they tried to extend the region of known aerodynamic behavior and placarded the airplane to flight in that region.

The most recent edition of the German Handbook specifying structural strength requirements (the Bauforschriften) was put out in 1936 and is almost entirely obsolete at this time.

For the past few years the strength requirements for a new airplane design were established for each individual design by the company producing it. This criteria was written into the "Description of the Airplane", which corresponds to a detail specification in the United States, and was made in co-operation with the resident Air Ministry strength engineer at the plant.

B. Load Factors

The manoeuvre load factors were specified as "safe" factors and were based as much as possible on flight test data from combat airplanes. This safe factor was the maximum that was normally experienced in combat and was almost anything that could be substantiated by statistical summary of the flight test data mentioned. However, common factors for fighters were +7 g and -3 g, and for bombers (medium) were +4 g and -2 g. The Ju-263 used a safe load factor of 6.3, but this was an outgrowth of the fact that it used the same wings as the Me-163 but was approximately 20% heavier. As a further example, the Ju-263 was limited to 5 g at Mach Number .8, as a result of aerodynamic troubles beyond this point. Aerodynamics was supposed to design tail surfaces and other control surfaces that would maintain this "safe" load factor as a maximum.

Landing load factors were specified by the Bauforschriften, of which many copies are available in the United States, and have not changed in 15 years or so. These landing loads fundamentally are based on a free drop of 3.7 to 4.5 meters/sec., depending on the type of airplane. The side load is .4 to .5 the gross weight of the airplane.

C. Gust Load

The basic gust velocity, for "safe" load is 10 meters/sec (see "safety" factors). The Bauforschriften gives the equation used for determining gust load factor, which is:

$$n = 1 + q \frac{s}{n} \cdot \frac{V_{Gust}}{V_{Airpl}} \times m \cdot n$$

m = slope of lift curve corrected for compressibility.

n = coefficient depending on aspect ratio and wing loading $= \frac{W/g}{b/2 \quad dC_L/d\alpha \quad \int c^2 dy}$

This equation differs largely from that used in the United States only in the value of n compared toour K.

D. Maximum Design Speed - Diving Speed.

According to the Bauforschriften, the maximum diving speed is 1.35 x VMax. in level flight, but this has not been adhered to in most recent airplanes.

The permissible diving speed is agreed upon by the company engineers and the Air Ministry representative and is usually based on wind tunnel tests, for high speed planes. For instance, the diving speed of the Ju-263 was taken as 600 mph, which is about its high speed in level flight, because according to flight and wind tunnel tests speeds in excess of this caused the airplane to be unstable, i.e. the center of pressure began to move rapidly.

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The diving speed is given as a true speed and the airplane placard is given as true speed so that compressibility effects are definitely limited. Indicated air speed limits are therefore a function of altitude. The limit diving speed was also sometimes given in relation to altitude. For instance, the Me-262 allowable speed was 620 mph below 20,000 ft. and 560 mph above 20,000 ft.

For some airplanes, the diving speed was greater than 1.35 x Vlevel. This applied, usually, to dive bombers - such as the Ju-88 which had a maximum level flight speed of 400 km/hr. and a design diving speed of 700 km/hr. However, this airplane incorporated an automatic pull-out device which did not permit the plane to exceed 3.2 g in pulling out of a dive. (For description of this device, see AI-2(G) No.2034).

Terminal velocity, or maximum attainable speed in zero lift dive, had never been used as design diving speed. In fact it had not been computed for most designs, as engineers thought it was too hypothetical a number to use in design since so much altitude is required to attain terminal velocity, and so many aerodynamic problems can occur at very high speeds. The German approach is to aerodynamically design the airplane to attain as high a critical Mach Number as possible and then not to structurally design past that critical Mach Number.

For instance, at Messerschmitt, the V-G diagram was continued at 7g to maximum permissible diving speed, but the center of pressure was assumed to remain at a constant 15% M.A.C. (for Me-262) for all load factors at this speed. Consequently, they did not actually design to a true 7 g condition at this speed and the placard of 4 g at this speed indicated the upper limit insofar as Cp. travel and reasonable stability are concerned.

E. Safety Factors

The "safe" load factor was multiplied by a safety factor of 1.35 to obtain the load factor at which no permanent set was allowable. Ultimate load, or failure load, was 1.8 times the "safe" load.

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All loads due to gust calculations are multiplied by 2 for ultimate load at Junkers, but by 1.8 at the other companies.

For landing gear loads, there are only two classifications, "safe" load and ultimate load. The ultimate load is 1.5 times the safe load, 1.5 being used instead of 1.8 because the landing gear is usually steel and the ratio between allowable ultimate stress and yield stress is lower.

All loads mentioned in the remainder of this section are "safe" loads and would be multiplied by 1.8 for design. (1.5 if landing gear)

F. COMPRESSIBILITY CORRECTIONS FOR STRUCTURAL CALCULATIONS

For use when applicable wind tunnel data was not available on a wing, an Air Ministry report was available which gave methods of determining the center of pressure travel, lift variation and drag variation, but it was normally only used up to M_{CR}. Some mention was made of the fact that the shock wave was considered to be a function of aspect ratio, as well as airfoil section, which is a refinement not taken into account in standard American methods. It is logical, however, because of the spanwise variation in effective angle of attack of the wing.

Also, the effect of compressibility on the fuselage pressure distributions was not seriously considered as it was felt that the excess velocity could transfer itself to circulatory velocity around the fuselage. (This was a theory practised at Junkers but not considered reasonable at any of the other companies. However, they all did agree to the fact that compressibility effects on the fuselage were not serious.)

G. Wing Design Conditions

The wing conditions were those normally designed for in the United States, i.e. balancing conditions, landing conditions, unsymmetrical conditions. However, the only unsymmetrical conditions specified were those due to aileron movement, no arbitrary unsymmetrical condition being checked. A condition was checked for full aileron movement at diving speed with pull up, but this was done mainly for information and a safety factor as low as 1.1 was permitted.

Since a flutter model and flutter calculations were made early in the design stage of a new airplane, at Junkers at least, the engineers claimed that the results of the flutter analysis influenced the design of the wing insofar as rigidity was concerned. Also, sufficient rigidity was built into the wing to keep the calculated aileron reversal speed 1.15 times the design diving speed.

Span distributions and chord distributions were made for all conditions, including the aileron conditions and were corrected for compressibility.

These distributions were made according to a DVL report FB 623. It probably was not much different from what we used, since it was published in 1937 and is available in English and probably American libraries.

H. Horizontal Tail Loads

No empirical conditions were specified for the horizontal tail. Based on aerodynamic data (furnished by aerodynamics department) dynamic calculations of tail load were made for all pertinent airplane altitudes and conditions, including rolling of airplane.

Unsymmetrical condition, in addition to that from rolling, was specified as 85% and 100% distribution of maximum tail load, per side.

I. Vertical Tail Loads

The vertical tail was designed for various combinations of rudder deflection and speed at critical airplane center of gravity locations. The minimum conditions checked were as follows:

(a) Full deflection of the rudder in .1 to .2 seconds (depending on airplane) at diving speed.

I.

(b) Full deflection of the rudder at high speed of the airplane.
(c) Side slip - (6^o to 8^o yaw) at diving speed.
(d) Balancing condition from aerodynamics stability requirements and rate of roll calculations.

J. Aileron Loads

Aileron loads were found for 12^o aileron deflection at airplane high speed and for 8^o deflection at diving speed. The ailerons and control system were designed so that these deflections produced a minimum rate of roll at high speed of one complete roll in 3 to 4 seconds for fighters (7 to 8 seconds for bombers). Wing twist was considered in these calculations and also its effect on the wing span distribution.

The limiting stick force for the aileron deflections and rate of roll was a 30 kg couple on a wheel control, or 30 kg stick force (65 lbs. appr.).

NOTE: The ailerons most generally used at Junkers were 20% of the chord of the wing with the hinge line at 25% aileron chord. They were found-nosed and were closely fitted to a mating section of the wing.

L. Control System Loads

"Safe" loads at the control stick are:

(a) 80 kg fore and aft.
(b) 30 kg couple on wheel (for aileron control).
(c) 150 kg rudder pedal force.

Deflection requirements were expressed in allowable percentage of total travel. They are:

(a) For 50 kg fore or aft load, maximum allowable elevator deflection was 20% of total travel.
(b) For 30 kg couple load on wheel, allowable aileron deflection was 30% of total travel.

L.

(c) No deflection requirement on rudder system.

M. Endurance Loads - Life of Airplane

Supposedly at the instigation of Junkers, instrumentation was installed in one to four of all German airplanes of each new type when they went into combat. These instruments recorded data on load factors and strains experienced during the life of the individual airplane. These results, from all airplanes instrumented, were then combined into a statistical summary which showed the number of times each load factor (or load on an important part) occurred in an average hundred hours flight time.

Repeated load tests were then made on all important components of an airplane, which duplicated the hundred hours' loading. The hundred hour cycle was then repeated until failure occurred and the life of the airplane was thereby determined.

The requirements as established by the Air Ministry for the minimum life of an airplane were:

Transport	- 10,000 hours	
Dive Bomber	- 600 hours	(recently changed from 1000 hours)
Fighter	- 200 hours	

N. Static Test Procedure

In addition to, but prior to the repeated load tests, ordinary static tests were made to check the stress analysis on new type airplanes.

All German companies did all their own static tests and most of them used hydraulic jacks exclusively as the loading methods.

Each component was tested to all of the design conditions, first to "safe" load, then to 1.35 times safe load, and then to 1.8 times "safe" load. Finally, the most critical condition was tested to destruction. The Air Ministry could be satisfied with a failure load of 1.75 times "safe" load, but most companies for their own information made material tests at the point of failure to correct the failing load to the stress used in stress analysis.

Inspection was made at "safe" load and 1.35 times safe load for signs of permanent set, and if necessary, at any point in the loading up to 1.8 times safe load.

The landing gear was both drop tested and static tested.

O. Vibration and Flutter Testing

At Junkers, a flutter model was made early in development of a new airplane and tested in a wind tunnel. Most of the other companies were not quite this progressive concerning flutter, but Messerschmitt was becoming very active. The models were made of wood, but more recently of a plastic with proportionate E and strength ratios to dural. The models were of course elastically similar to the actual airplane, as was the weight and weight distribution. From these scale similarities, a factor could be computed so that a model speed in the tunnel could be compared to a much higher speed on the actual airplane. Of course, this procedure does not account for compressibility effects.

P. Source of Material Allowables

From a government publication "Flugwerkstoffe" or "Flight Material", the companies made up as part of their standard book series or "Norme", a selection of the materials usable in their particular companies.

Each material listed included the government code number, a summary of "performance" (allowable stresses - which are the minimum acceptable by inspection), a description of the composition, a statement regarding its weldability, and the welding rod to be used.

The government code number acts as a description of the material, the first number denoting general class of material (steel, light metal, etc.), the second number denotes the major alloying material, third and fourth the type of alloy, and the fifth number (comes after a decimal point) gives heat treat state of the material.

In general, the materials used in actual manufacture were inferior to those of the Allies. The Germans had experimental materials that were fully as good as our high strength alloys, but lack of many of the alloying materials kept these alloys from being used in actual aircraft. In fact, many of the new designs were using wooden construction because of the lack of <u>all</u> metals.

Section III

5. Aircraft Propulsion

A. Introduction

This section does not attempt to completely describe and analyse all of the various forms of aircraft propulsion systems existing in Germany but is, instead, an attempt to outline briefly some of the most significant developments, especially in connection with the work done on propulsive ducts and rockets. The material contained herein has been compiled from field trips to aircraft factories and research laboratories, review of available documents, and interrogation of technical personnel engaged in this type of work. It is impossible, in the short period of time available, to examine in complete detail, even the field of propulsive ducts, but it is hoped that this preliminary report will provide a comprehensive survey of the material which will be available through the facilities of the Air Document Research Center.

Detailed discussions and descriptions of the various gas turbine units are contained in the reports listed in Appendix A, which are available through the FIAT (Rear). The scope of this report does not include any discussion of conventional internal combustion engines as this subject has already been covered by other CIOS and U.S. Naval Technical Mission in Europe reports.

B. Terminology

In any discussion of German aircraft power plants there is always reference to the official designations of the various units. The following list of common terms and designations is, therefore, included to clarify any confusion in connection with the accepted terminology.

R-Antrieb	Pure rocket motor
L-Antrieb	Rocket motors with air intakes including Lorin and V-1 (stationary and intermittent)

RL-Gerat - Rocket pulse motor using combination of rocket fuels and pulsating air intakes

IL - Intermittent jet

TL-Turbinenluftstrahl - Gas turbine - propeller and jet combination.

PTL- Propeller-Turbinenluftstrahl- Gas turbine - propeller and jet combination.

TLR - Jet turbine combined with rocket

Kolbenmotern - Reciprocating engines

Hypergol - Spontaneously igniting mono-propeller

Ergol - Single component of hypergol

Monergol - Mono-propellant

Katergol - Catalyst fuel

Lithergol - Solid liquid propellent combination

Solbei - Red fuming nitric acid

Solbeik - Highly concentrated nitric acid.

Myrol - 75% methyl nitrate - 25% methanol

T-Stuff - Hydrogen peroxide - 80% + 20% water

C-Stuff - Hydrocarbon (gasoline, alcohol, etc.)

C. Athodyds or Ramjets

1. General

The ramjet or athodyd, as it is sometimes known in the United States and Great Britain, is referred to in Germany as the Lorin Drive (Lorin-Antrieb or Lorin Triebwerk) after the French inventor of this propulsion system.

When used in the range of supersonic speeds it is sometimes known as the Trommsdorff-Drive in honor of a German officer who was active in its promotion during the past few years.

The ramjet has been considered as a propulsion unit for both subsonic and supersonic speeds. The problems involved in the two fields of application are in several respects quite different, although some basic elements are essentially the same. Practically all of the experimental work on complete ramjets has been done at supersonic speeds to date, However, it seems quite feasible that the greatest efficiency and usefulness of this power plant will eventually be in the region of high supersonic velocities.

The major portion of the German work on athodyds had been done by a few extremely high calibre engineers who were associated with the various aircraft factories and research laboratories. These men worked independently, largely due to the policy of LFM which severely restricted the exchange of technical information between individuals and manufacturers. The outstanding propulsive duct projects as developed by the following technicians will be discussed separately.

1. Dr. Lippisch, associated with Messerschmitt and later with the Aeronautical Research Institute of Vienna.

2. Ing. Otto Pabst, Focke-Wulf, Bad Eilsen.

3. Dr. Ing. Eugen Sänger, DFS, Ainring

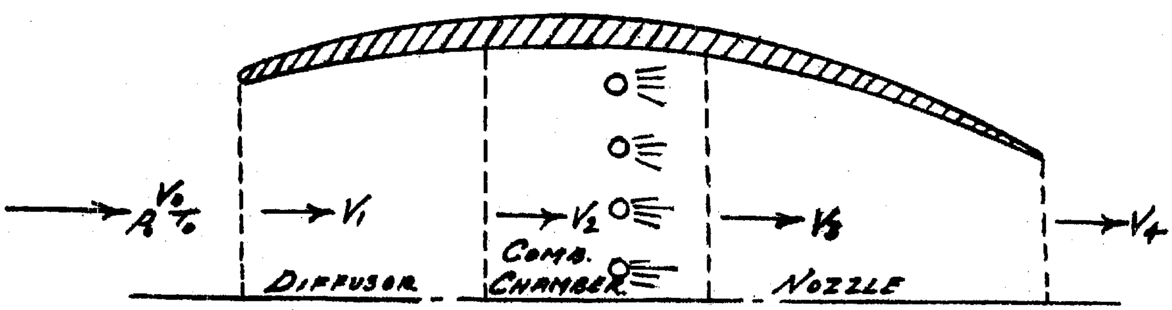
4. Dr. Klaus Oswatitsch, KWI, Gottingen

5. H. Walter, Kiel.

Appendix B comprises a partial list of available reports dealing with the work of these various individuals and these reports, like those listed in Appendix A, are available through the FIAT (Rear) and/or the Air Document Research Center.

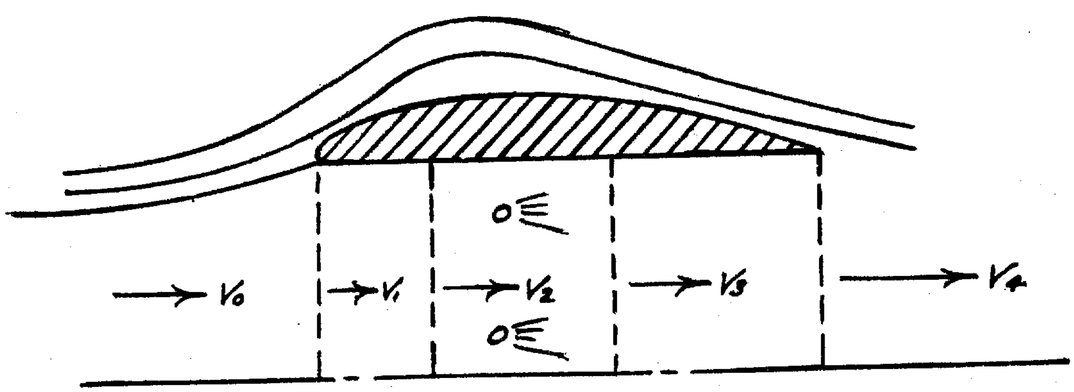
2. Generalized Theory

The ramjet is undoubtedly the simplest possible type of propulsive device. It comprises, as indicated in the sketch, a diffusor where air entering with velocity V_1 has its velocity reduced to V_2 (with an accompanying pressure rise to P_2), a combustion chamber where heat is added and the density lowered with corresponding rise in velocity to V_3, and a nozzle from which the combustion products and air are ejected with velocity of V_4. These elements are surrounded by a fairing or cowling, and all existing ramjets, as far as is known, are figures of revolutions as indicated in the sketch.



The usual procedure for calculating ramjet performance in the past has been to use the usual aero and thermodynamic relations for the gas flow in the duct in order to determine the exit flow characteristics for given entrance and heat-added conditions. The thrust is then calculated as the difference between the rate at which momentum leaves the ramjet and that at which it enters. Such a procedure does not introduce the conditions over the fairing, except as they affect the external drag producing a quantity which must be subtracted from the calculated thrust to give the net thrust produced by the ramjet. Furthermore, this procedure does not focus attention on the detailed consideration of the actual forces acting on the surfaces of the ramjet to produce the thrust.

Ing. Otto Pabst of Focke-Wulf was apparently the first to advocate a new approach to the problem along these lines and his conception may be understood easily through reference to the idealized case shown in the following sketch. This athodyd is a perfectly cylindrical inner duct surface containing a short combustion chamber, and surrounded by a ring cowling or cover. If there is no combustion, V_1 and V_4 are nearly the same as V_0 and, neglecting skin friction, there is no net drag or thrust on the ramjet. The addition of combustion has the effect of choking the duct so that the air entering is slowed down; i.e., V_1 is much less than V_0. Because of the heat added, however, the density of the exit gases is reduced and the velocity increased so that V_4 is greater than V_0. The ramjet will now produce a thrust which must result from the pressure differences over the surfaces. The inner duct surface, being a cylinder parallel to the flow, cannot contribute to the thrust and it must therefore be the resultant of the pressures over the outer fairing or cover. The outside flow is obviously analogous to that over the leading edge and upper surface of an airfoil at moderately large angle of attack. Hence a high negative pressure over the nose will exist, and it is this which is primarily responsible for the thrust.



This theoretical approach focuses attention on the outer fairing or cover which in previous analyses was not seriously considered. The problem of designing this fairing is closely related to that of radial engine cowling rings on which much work has already been done both in Germany and in the United States.

Kuechemann, at the AVA in Gottingen, has intensively studied this question both theoretically and experimentally for the past several years and has published several valuable reports which are indicated in Appendix B. He likewise conducted the wind tunnel testing on models of the Pabst or Focke-Wulf type of ramjet and subsequently developed methods for calculating the pressure distribution and resultant thrust.

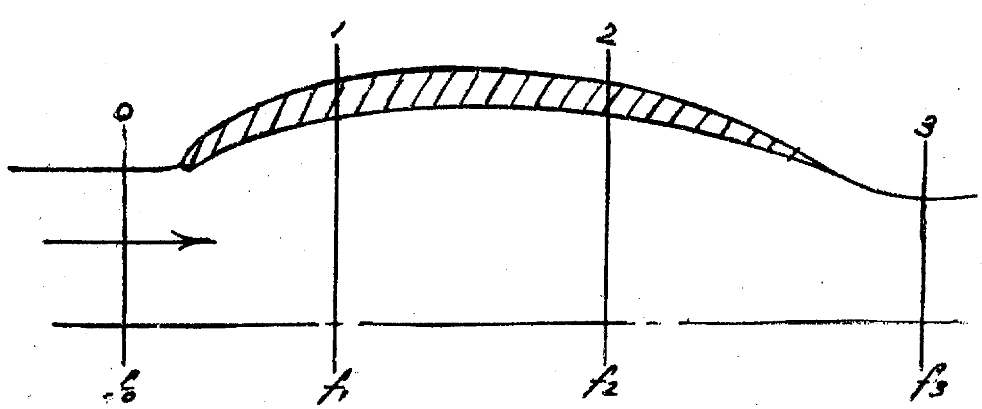
An important feature of the idealized Pabst type of ramjet is that since there is no internal diffuser, all of the pressure increase occurs before the entrance in the free stream. A diffusor of this type, with most of the pressure rise outside, is commonly known in Germany as a "Fang-Diffuseur", as distinguished from an "Einlauf-Diffuseur" where all the pressure rise occurs internally in the diffusor itself. The two types may be characterized by the value of the ratio V_1/V_o. For the Einlauf type the value is approximately 1. For the Fang type, it is 0.5 or less. An experimental fact about the Fang diffusor which is not clearly understood theoretically is that, if properly designed, it will reach high subsonic free flight Mach numbers (0.8 to 0.9) without any rapid rise in drag coefficient. This occurs in spite of the high negative pressure and associated high local velocity over the leading edge.

The supersonic ramjet differs in principle from the subsonic only insofar as the diffusor design is affected by the different pressure relationships and, of course, the outer surface should obviously have a different shape in the supersonic and subsonic cases. The critical problem of the supersonic ramjet is, accordingly, the construction of a supersonic diffusor having good pressure recovery without requiring an external fairing involving high drag. Oswatitsch at Gottingen has been actively working on this problem and appears to have found a satisfactory solution which is discussed in detail in Section 6. In this section it is sufficient to note that, while at subsonic speeds, only low compression ratios can be reached, the supersonic ramjet with a good diffuser can lead to very high compression ratios with a resulting high thermal efficiency. For example, at M = 2.9 Oswatitsch obtained a pressure of 19 atmospheres in the combustion chamber and estimates the overall thermodynamic efficiency of this ramjet at 45 percent.

3. Dr. Lippisch - Aeronautical Research Institute of Vienna

It appears that Dr. Lippisch's work was rather rushed, and that on the basis of simplified mathematical relations, he proceeded directly with free flight tests on scale models. It was anticipated that some portion of the German development work on high speed fighters would be based on the outcome of these tests.

According to Dr. Lippisch, the following shows the derivation of the mathematical relationships that he employed in his athodyd design work. He did not go into the thermal calculations involved in the process but did specify that the calculations were for incompressible flow in the sub-sonic region. For flow in the sonic region, he indicated that by proper diffusor design (he referred to the work of Dr. Oswatitch at the Kaiser Wilhelm Institute at Göttingen) diffusor efficiencies as high as 0.75 could be otained, and that beyond the diffusor, the flow being sub-sonic, the conditions remain the same as those for sub-sonic flow.



SYMBOLS

f = cross-sectional area

T = temperature (absolute

p = pressure

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ρ = mass per cubic unit of gas

w = velocity of gas

S = thrust

C_s = thrust coefficient

μ = specific fuel consumption = $\frac{\text{weight}}{\text{Thrust x Time}}$

C_μ = specific fuel consumption coefficient

C_p = specific heat for air

η = efficiency (thermal over-all)

H = $\frac{1}{\text{Thermal equiv. of work}}$

N.B. Dr. Lippisch worked with C.G.S. system

Starting with equation of flow

$$-dp = w\,dw$$

$$p_1 = p_0 + (\rho/2) \cdot w_0^2 \, (\rho/2) \cdot w_1^2$$

$$\therefore p_1 - p_0 = \rho_0/2 \; w_0^2 \left[1 - \left(\frac{w_1}{w_0}\right)^2\right]$$

and $w_1 = w_3$

$$\frac{T_2}{T_1} = \frac{\rho_1}{\rho_2}$$

$$T_1 = T_0$$

$$T_2 = T_3$$

$$p_1 = p_2$$

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$$p_2 - p_3 = \rho_2/_2\, w_3^2 \left[1 - \left(\frac{w_2}{w_3}\right)^2\right]$$

$$\rho_0 w_0^2 \left[1 - \left(\frac{w_1}{w_0}\right)^2\right] = \rho_2 w_3^2 \left[1 - \left(\frac{w_1}{w_3}\right)^2\right]$$

$$\frac{w_1}{w_0} = \frac{f_0}{f_1} \quad \text{and} \quad \frac{w_2}{w_3} = \frac{f_3}{f_2}$$

$$\frac{T_2}{T_0}\left(\frac{w_0}{w_3}\right)^2\left[1 - \left(\frac{f_0}{f_1}\right)^2\right] = 1 - \left(\frac{f_3}{f_2}\right)^2$$

$$\rho_1 f_1 w_1 = \rho_2 f_2 w_2$$

$$\frac{f_1}{f_2} = \frac{\rho_2}{\rho_1} = \frac{T_1}{T_2}$$

$$S = \rho_0 f_0 w_0 \left[w_3 - w_0\right]$$ Neglecting weight of products of combustion added.

$$= \rho_0/_2\, w_0^2 f_0 \left[2\left(\frac{w_3}{w_0} - 1\right)\right]$$

also $S = C_s \cdot f_2 \cdot \rho_0/_2 \cdot w_0^2$

and $C_s = 2\,\frac{f_0}{f_2}\left[\frac{w_3}{w_0} - 1\right]$

or $S = \rho_0/_2\, w_0^2 \cdot f_2 \cdot 2\,\frac{f_0}{f_2}\left[\sqrt{\frac{T}{T_0} - \left(\frac{f_0}{f_2}\frac{T}{T_0}\right)^2\left(\frac{T}{T_0} - 1\right)} - 1\right]$

and if $\left(\frac{w_3}{w_0} - 1\right) = \left[\sqrt{\frac{T}{T_0} - \left(\frac{f_0}{f_2}\frac{T}{T_0}\right)^2\left(\frac{T}{T_0} - 1\right)} - 1\right]$

$$\therefore C_s = 2\,\frac{f_0}{f_2}\left[\sqrt{\frac{T}{T_0} - \left(\frac{f_0}{f_2}\frac{T}{T_0}\right)^2\left(\frac{T}{T_0} - 1\right)} - 1\right]$$

for $C_{s\,max.}$ $\left(\frac{f_0}{f_2}\right)^2 = \frac{4\frac{T}{T_0}-1-\sqrt{8\frac{T}{T_0}-1}}{8\left(\frac{T}{T_0}\right)^2\left(\frac{T}{T_0}-1\right)}$

$$\mu = \frac{g\cdot C_p\cdot T_0}{H\cdot \omega_0}\left(\frac{\frac{T}{T_0}-1}{\frac{\omega_2}{\omega_0}-1}\right)$$

$$C_\mu = \frac{1}{2}\,\frac{T/T_0-1}{\sqrt{\frac{T}{T_0}-\left(\frac{f_0}{f_2}\,\frac{T}{T_0}\right)^2\left(\frac{T}{T_0}-1\right)}-1}$$

and $\left(\frac{f_0}{f_2}\right)^2_{OPTIMUM}\left(\frac{T}{T_0}\right)^2\left(\frac{T}{T_0}-1\right) = \frac{1}{8}\left(4\frac{T}{T_0}-1-\sqrt{8\frac{T}{T_0}-1}\right)$

$$= \frac{1}{2}\,\frac{T}{T_0}-\frac{1}{8}\left(1+\sqrt{8\frac{T}{T_0}-1}\right)$$

$$\sqrt{\frac{T}{T_0}-\left(\frac{f_0}{f_2}\cdot\frac{T}{T_0}\right)^2\left(\frac{T}{T_0}-1\right)} = \frac{1}{2}\,\frac{T}{T_0}+\frac{1}{8}\left(1+\sqrt{8\frac{T}{T_0}-1}\right)$$

$$= \frac{1}{4}\sqrt{2\frac{T}{T_0}+\frac{1}{2}\left(1+\sqrt{8\frac{T}{T_0}-1}\right)}$$

$$C_{s\,MAX} = 2\left(\frac{f_0}{f_2}\right)_{OPT}\left[\frac{1}{2}\sqrt{2\frac{T}{T_0}+\frac{1}{2}\left(1+\sqrt{8\frac{T}{T_0}-1}\right)}-1\right]$$

C_μ for C_s max. $= \frac{T/T_0-1}{\sqrt{2\frac{T}{T_0}+\frac{1}{2}\left(1+\sqrt{8\frac{T}{T_0}-1}\right)}-2}$

According to Dr. Lippisch, the usual method is to take f_2 as large as the design will allow; therefore f_1 is fixed by the temperature relation (i.e., whether high thrust or high efficiency is desired). For max. C_s, values of $\frac{T}{T_0}$ between 4 and 5 are used, along with f relations for max. C_s. For max. η given a specified C_s, the max. $\frac{1}{C_\mu}$ is obtained from the envelope of curves of $\frac{T}{T_0}$ giving C_s versus $\frac{1}{C_\mu}$. The $\frac{T}{T_0}$ giving max. $\frac{1}{C_\mu}$ at the specified C_s will then allow the calculation of the areas. Though $\frac{T}{T_0}$ can vary between 1 and 8, Dr. Lippisch normally used a ratio of 3.

If $$\mu = \frac{C_p T_0}{H} \cdot \frac{2g}{\omega} \cdot C_\mu$$

$$C_\mu = \frac{1}{2} \frac{T/T_0 - 1}{\omega_3/\omega_1 - 1}$$

In an expression for the fuel consumption per meter of flight we have

$$B/\text{meter} = C_s \cdot C_\mu \cdot \frac{C_p \cdot T_0}{H} \cdot \gamma \cdot f_1$$

where γ = specific wt. of air

and since $C_s \cdot C_\mu = \left(\frac{\omega_1}{\omega_0}\right)\left(\frac{T}{T_0} - 1\right)$ it is indicated that the lowest fuel consumption per unit distance results from the largest possible f_1 or diffusion.

Finally we have the overall thermal η

$$\eta = \frac{1}{C_\mu}\left(\frac{H \cdot \omega_0^2/2g}{C_p \cdot T/T_0}\right)$$

In the sonic region T_0 is no longer the same as T_1, therefore T (T_2 or T_3) must be higher if the same $\frac{T}{T_0}$ ratio as existed for subsonic flow is to be maintained.

At the present time, Dr. Lippisch, in conjunction with his mathematics assistant, Dr. Ringleb, is working on a graphical method to be used in selecting cross-sectional areas for the athodyd. In order to attempt to do this he plots the processes of the gas on coordinates of f_o/f versus w/w_1. He states that this method will indicate the dimensions that will give the minimum pressure loss in the various processes. However, since he was still working on this idea, he was not too sure of its validity.

4. <u>Int. Otto Pabst - Focke-Wulf, Bad Eilsen</u>

The interest of the Focke-Wulf organization in athodyds began in 1941 after examination of the results obtained by Dr. Sänger in his flight tests on a Do. 217. Realizing that Sänger's ducts, with a length of approximately five diameters, were much too long to be practical, Dr. Pabst was assigned the task of investigating combustion problems whose solutions might reduce the tube length required. Pabst's first experiments were conducted at Kirkhorsten (4 miles from Bad Eilsen) and utilized hydrogen for the first phases of the investigation. As a result of these tests he developed the burner shown in Figure A, which has the apex of the cone pointing downstream. The fuel is simply supplied to the dead air space behind the disc and the actual mixing of fuel and air takes place in the turbulent area around the circumference of the disc. By combining a large number of these burners, Pabst evolved the power plant illustrated in Figure B. This configuration resulted in a reduction of the burning length from the five diameter requirement of Sanger to approximately 3/4 of a diameter so that the entire installation with the necessary entry and exit cones does not require more than 2½ diameters. Dr. Pabst did not utilize baffles with his installation and consequently the energy losses in the system are extremely low.

As has already been mentioned, Dr. Pabst's other outstanding contribution to the theory of propulsive ducts was his analysis of the pressure distribution over the outer fairing and his advocation of the theory that most of the pressure increase required for combustion could be accomplished in the free stream, instead of in an internal diffusor.

Tests of Dr. Pabst's athodyd in the wind tunnel at Braunschweig, up to a Mach Number of 0.9 indicated that:

(1) Burning was steady and stable up to a Mach Number of 0.9, the highest value reached.

(2) Air fuel ratios were varied between 2 and 14 times the stoicheiometric value.

(3) Exhaust jet temperatures varied between 200 degrees C. and 900 degrees C. Combustion efficiency, as determined from exhaust gas analysis, was practically 100 percent for exhaust jet temperatures above 450 degrees C.

(4) Maximum thrust coefficient of the order of 0.4 (the exact values cannot be determined from the report since the overall ramjet diameter is not given exactly).

(5) Specific fuel consumption at maximum thrust is 0.41 gm/kg sec. occurs at M. = 0.8. Typical test results for this athodyd are shown on Figure C.

Pabst states that if vaporized gasoline were used as a fuel instead of hydrogen, the above results would remain essentially the same except that the specific fuel consumption would be increased to:

1.2 gm/kg sec for maximum thrust

1.05 gm/kg sec for maximum efficiency

Figures D and E are typical calculated curves of athodyd performance and are presented in the form of the thrust coefficient (C_T = Thrust/$\frac{1}{2}\rho$ v^2 x Max. Area) vs. F4/F2 (the ratio of exit area to combustion chamber area). The variable α is the ratio of fuel weight to air weight and η indicates the overall propulsive efficiency based on the heat content of the fuel. These illustrations indicated that the maximum thrust would be obtained at high speeds with a ratio of F4/F2 = 0.8 - 0.9, but that the maximum efficiency would occur at F4/F2 of approximately 0.5. For the model tests at Braunschweig a ratio of F4/F2 = 0.625 was selected as a compromise and the results are shown on Figure O.

It is understood that Pabst had made many combustion tests with the ramjet burning gasoline as fuel. In this case the gasoline is vaporized and superheated outside of the ramjet before being fed to the burners. Pabst accomplished this by building a heat exchanger into the downstream cross-section of a very small ramjet of his usual design. For an airplane installation requiring two large ramjets each having 50 burners, he required one small heat exchanger-ramjet with two burners to vaporize and pre-heat the gasoline. In other words, about 2% of the total fuel consumed is used for vaporizing and pre-heating.

5. Dr. Ing. Eugen Sänger - Deutsche Forschungsanstalf fur Segelflug - Ainring

Dr. Sanger was one of the first aerodynamists in in Germany to become interested in the problems of propulsive ducts and while he had not developed his theory to the same degree as Lippisch and Pabst, he had acquired much more practical experience from his experiments with actual models in flight on a Do-217 and from ground tests on moving vehicles. This work is discussed in detail in U and M Report No. 3509 by Sänger and I. Bredt entitled, "Ueber Einen Lorinantrieb fuer Strahljoeger". However, no wind tunnel tests, under controlled conditions, were ever made, so that it is impossible to establish any correlation between Sänger's theory and his test results.

Dr. Sänger was an advocate of the Einlauf type of diffusor, which has all the pressure rise inside the unit. His experiments were conducted using gasoline as fuel and he generally injected the fuel upstream, against the airflow, in order to obtain a reasonable efficiency of combustion. The combination of these facts resulted in a high drag unit inasmuch as the combustion chamber normally had a length/diameter ratio of 5, while the entire athodyd length was at least 10 times the diameter. Sanger's experiments were generally conducted with a high fuel/air ratio and consequently the specific fuel consumption was higher than the figures obtained by Pabst - Sänger's models requiring approximately 8 grams of fuel/kg. thrust/sec. Dr. Sänger concentrated on the Einlauf diffusor because he intended to operate his propulsive ducts at Mach Numbers close to unity and he was afraid that the Fang type of diffusor would experience difficulty with local regions of supersonic velocity around the outer lip of the entry.

The results obtained from the flight tests at Ainring are rather difficult to analyse, because of the necessity of accurately measuring the thrust and drag and separating the effect of the drag of the structure supporting the propulsive duct from the airplane drag. The highest thrust coefficient (converted to the conventional basis of net available thrust) was 0.38 and a diffusor efficiency of 85 to 95 percent. Flight speeds were never higher than 200 meters/second and the velocity ratio in the diffusor was approximately 6 to 1. No attempt was made to cool the ramjet by fuel film or any means other than external airflow and radiation. Internal gas temperatures, measured with a pyrometer during night tests, was of the order of 2000° C.

Sänger provided approximately 80 fuel injection nozzles for the 1.5 meter diameter propulsive duct and the pressure drop across these nozzles was calculated to be 40% of the velocity head of the internal flow at that point. The diffusor had an included angle of 10° divergence and the leading edge was slightly rounded to provide for smooth entry flow conditions. The exit area was established so as to produce an entry flow velocity of approximately 150-180 meters/second with complete combustion, and the major portion of the flight work was accomplished at a flight speed equal to this duct entry velocity.

For flight tests conducted with entry velocities other than free stream velocity, an error is introduced in the interpretation of Sänger results, published in U and M Report No. 3509, which should be explained. All test results are expressed as thrust coefficient (C_V) which represents the "internal thrust" and not the net thrust. This coefficient does not include the drag on the external surface or the friction losses of the internal flow. It is possible to calculate the conventional thrust values from Sänger's data given for the drag without combustion, but the results are questionable, due to the difficulty of accurately determining the interference effect of the athodyd supports. Fuel consumption values are always expressed in terms of fuel/air ratio with the assumption that the entrance velocity and free stream velocity are equal. Where this is not the case, a correction must be applied.

6. Dr. Klaus Oswatitsch - Kaiser Wilhelm Institute - Göttingen

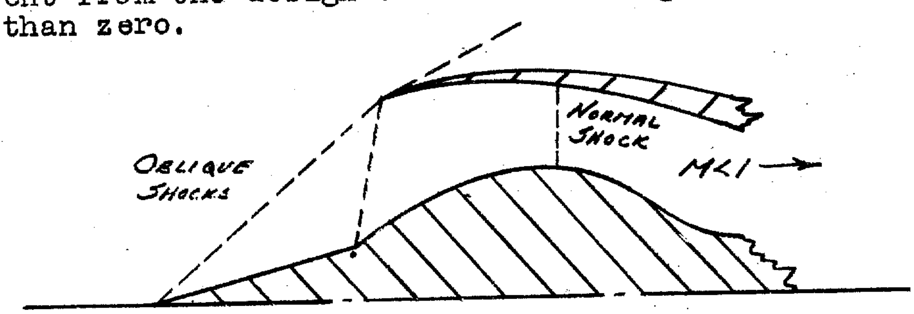
Dr. Oswatitsch, a member of Prandtl's Institute at Göttingen, had been working since 1943 on a ramjet to drive shells at high supersonic speeds, of the order M = 3. This work is referenced in the reports listed in Appendix B. Oswatitsch concentrated the majority of his efforts on the two problems of obtaining a supersonic diffusor with high pressure recovery and of developing an external shape for the combination missile and ramjet which would have relatively low drag. He found that the two problems are closely related and are, in fact, somewhat contradictory in their requirements, so that his final proposed design represented a compromise in respect to both diffusor efficiency and external drag.

The starting point of his diffusor development was the observation that at high Mach Numbers, 2.0 and above, a conventional contracting tube type of supersonic diffusor always involves a normal shock at or before the tube entrance, providing the contraction exceeds a (rather small) limiting amount. The entropy increase through such a normal shock at high Mach Numbers is large, which means that the total pressure downstream of the shock is only a small fraction of the reservoir pressure which would adiabatically produce the free stream Mach Number. This means that the pressure recovery from such a diffusor is relatively low, i.e. the diffusor efficiency is small. On the other hand, the loss in a normal shock at Mach Numbers only a little greater than 1, is very small, and it is possible to reduce a high Mach Number to a lower one with little loss by requiring the air to cross an oblique shock, or better still, a succession of oblique shocks.

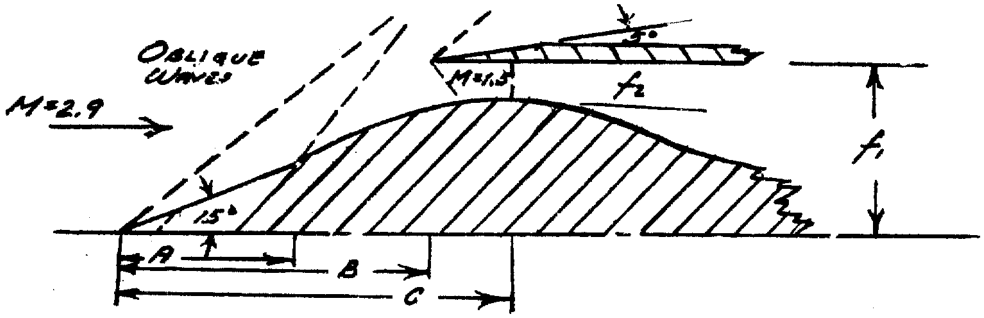
In view of the above considerations, Oswatitsch set out to develop a diffusor in which the air was slowed down to a Mach Number close to 1 by means of a series of oblique shocks and only then passed through a normal shock and into a conventional subsonic diffusor leading to the combustion chamber of the ramjet.

Many calculations were made and very comprehensive tests were carried out, mostly at a Mach Number of 2.9 and a large number of modifications were investigated before the final compromise design was reached. Both the theoretical and experimental work are very completely presented in the reports listed in Appendix B. However, the following notes provide a brief indication of the scope of his work together with an outline of his logic in determining the final configuration.

The first model (indicated in the sketch) was an axial body with two discontinuous increases in slope in the supersonic flow region. These changes in slope produced two oblique shock waves having less total energy loss than a single shock wave. The external portion of the diffusor has its sharp leading edge at the intersection of these two shock waves and has the same external slope as the forward portion of the internal body. When combined with a suitably faired afterbody, this model gave a relatively low total head loss (approximately 25-30 per cent), but did have a high external drag. The model was obviously impractical for operation at Mach Numbers different from the design value and at angles of attack other than zero.



The second model, illustrated below, had a slightly higher total head loss, ranging from 35-40 per cent of the total pressure ahead of the diffusor, but the drag coefficient was only 50-60 percent of Model I, because of the low curvature of the external portion of the diffusor. The ratio of the circumferential disc areas f_1 to f_2 is approximately 2.5 to 1.0. The dimensions A, B, and C are the result of locating the shock waves, approximately as shown, relative to the outer cylindrical structure by using angles for the inner body which will produce successive oblique shock waves, each having a total head pressure loss of only 5%. This design increased the losses external to the diffusor because the shock waves now extend into the free stream and the diffusor is also effective over a small range of Mach Numbers and angles of attack.



The theory indicates that the utilization of infinitely small angles (a smooth curve) for the inner body should result in no total head pressure loss, and in pursuit of this solution, Dr. Oswatitsch produced a model, which when tested proved to be actually inferior to his second attempt.

Although no tests with combustion have been reported, Oswatitsch had worked on the combustion problem and feels that there is no essential difficulty if the fuel is vaporized and preheated (as in Pabst's subsonic ramjet) before being fed to the combustion chamber.

Oswatitsch had also been making performance studies on large, winged supersonic missiles or aircraft with ramjet power. At M = 3, he calculated overall thermodynamic efficiencies of between 40 and 50 percent with combustion chamber temperatures of 1700 degrees centigrade. These extraordinarily high efficiencies, resulting from the high compression ratio, make him very confident that such aircraft can be constructed to fly several thousand miles at altitudes of 20 to 30 kilometers and speeds of 1,500 to 2,000 miles per hour.

7. Walter Project - Kiel

The firm of H. Walter which had been developing rocket motors for various German aircraft had developed an Einlauf type athodyd which was tested by LFA at Braunschweig. The combustion chamber was approximately two diameters in length and the overall length diameter ratio was approximately 7.0. The design was such as to produce considerable pressure rise inside the diffusor and utilizing conventional terminology, the wind tunnel investigation produced the following results:

Drag coefficient 0.3 at speeds of 100-280 m/sec.
Thrust coefficient 0.3 for speeds of 200-260 m/sec.
Spec. fuel consumption 29m/kg.sec. at 200-260 m/sec.
Air/fuel ratio - 25-30 for maximum thrust.

D. ROCKETS

1. General Discussion

As in the previous portions of this report, no attempt will be made to describe in detail the design, manufacture and performance of such well known rockets as V-1, V-2, Wasserfall and various others, but instead, the discussion will consist of an outline of the theoretical work of the outstanding German rocket technicians. A bibliography of existing reports on German production rockets will be found in Appendix C.

Most of the important design work on rockets was accomplished by members of the LFA at Braunschweig, while the actual testing was accomplished in the wind tunnels at Braunschweig and Peenemunde.

The chief item of research was the development of proper fuels and fuel combinations, while the problems of proper external configuration, control, and mechanical installation of the power plant, were largely left to the ingenuity of the engineers of the commercial firms using rocket propulsion.

2. Prof. Otto Lutz - LFA, Braunschweig

Dr. Lutz was in charge of rocket research for the Hermann Goering Aeronautical Research Institute at Braunschweig, Vulkanrode. All of the rocket research work done at LFA came directly under Dr. Lutz's supervision and he was also actively engaged in several personal fuel research programs.

According to Dr. Lutz, the following points should be considered when discussing the best type of rocket fuels without limitations. The main point is the choice of the oxygen carrier - oxygen and nitric acid providing the least ignition delay of all known carriers. Hydrogen peroxide produces lower operating temperatures and has the added advantage of easy handling characteristics but does have a substantially greater ignition delay. Quantitively, in ratio of ignition delay - the choice of the fuel is immaterial.

Considering fuels with maximum temperature limitations, i.e., engines operating for extended periods of time, hydrogen peroxide provides the best possible propellant. The best fuel for hydrogen peroxide is hydrazine hydrate inasmuch as no ignition is required, and the reaction will take place with a 50% solution of hydrogen peroxide. The hydrazine. hydrate can likewise be diluted with water and methanol and the final fuel developed for the Walter installation in the Me-163 was 13% water 30% hydrazine hydrate and 57% methanol containing potassium cupro-cyanide as the catalyst.

For exceptionally low temperatures of the order of 1000-2000°C, ammonia is generally used in excess because, due to the disassociation, free hydrogen is obtained, which lowers the molecular weight and increases the efficiency since the efficiency is directly proportional to the flame temperature divided by the square root of the molecular weight. Brenzkatechin diluted with methanol or vinyl ethers can also be used with the same degree of success.

For nitric acid, the best fuels contain amines such as aniline, ethyl aniline, xylidine or Brenzkatechin which are mixed with crude benzol, xylol or vinyl ethers.

Dr. Lutz had a rule of thumb, which he claimed worked exceedingly well for preliminary rocket design and which consisted of providing a propellant flow of one liter per second for each liter of volume of combustion chamber space. Such large flows require propellants which are readily mutually ignitable in order to prevent explosions occurring from large accumulations of these fuels.

Indicating the relative order of specific fuel consumption for various temperatures, Dr. Lutz has found that for low chamber temperatures with hydrogen peroxide, a SFC of 6 grams/kg. thrust/second is average. With liquid oxygen and resultant high chamber temperatures, a consumption of 4 grams/kg. thrust/second is reasonable.

With almost all fuels a catalyst is required. The best is vanadium chlorite, which was extremely rare in Germany, and therefore with nitric acid, diluted iron salts were used while with hydrogen peroxide, various copper salts such as potassium - copper cyanide or forms of the copper colloids were found to be operationally acceptable.

Considerable work had been done by Lutz in conjunction with Dr. Haussman of I.G.Farben at Ludwigshafen on the development of hypergols. The use of hypergols provides excellent control because there are no stability problems and the temperatures remain low. However, the fuel consumption is high, averaging about 9 grams/kg. thrust/second. In general, with the oxygen carrier of the hypergol a higher temperature is produced than when the decomposition of the fuel takes place alone.

3. Prof. Wolfgang Noeggerath - LFA, Braunschweig

Prof. Noeggerath was assistant to Dr.Lutz at Braunschweig and at the same time was actively perusing several of his own original theories in connection with rocket propulsion.

He had spent considerable time and with some success on investigations of the introduction of ballast into rocket propellants. The dominant performance factor is the ratio of the flame temperature to the average molecular weight of the products of combustion, and therefore. the addition of relatively inert substances which decompose into products of low molecular weight, allows operation at low temperature without an accompanying decrease in the exhuast velocity. Noeggerath had verified this theory by adding excess ammonia to ammonium nitrate or nitrous oxide and decreasing the temperature 200-250°C without effecting the exit velocity.

4. Dr. Johannes Winkler - LFA, Braunschweig

Dr. Winkler was another member of Dr.Lutz's staff at the Hermann Goering Institute and had been working with liquid oxygen and liquid methane rockets since 1926. In 1930 while employed by Junkers, he developed the first successful rocket flown in Germany. Since his association with Dr. Lutz at LFA, he had been concentrating on a nitrous oxide-gasoline rocket which was first flown in 1940. The mixture ratio was 3 parts of nitrous oxide to one part of gasoline which produced 120 kg. of thrust for 5 minutes with a gross weight of

5 kg. and a specific fuel consumption of 6.3 grams/kg. thrust/second. The propellant feed pressure was 22 atmospheres and the combustion chamber operated at 17 atmospheres. The motor length was 350 mm. with a maximum diameter of 60 mm. Ignition was obtained by introducing gasoline and nitrous oxide into a separate combustion chamber and igniting them with a spark plug. A portion of the gasoline injected is used for film cooling in the combustion chamber, similar to the system used in the V-2 rocket.

Dr. Lutz and Dr. Noeggerath were also active in the investigation of the utilization of nitrous oxide as an auxiliary fuel for internal combustion engines. Nitrous oxide is injected into the supercharger inlet at altitude supplying additional oxygen, while the nitrogen helps keep the engine temperature down. At an altitude of 10 km., the conventional 1500 horsepower Junkers engine decreases to 800 brake horsepower - the injection of one gram/second produces 4 brake horsepower, so that 175 grams/second will restore the 1500 horsepower rating at 10 km. altitude. The utilization of GM-1 or nitrous oxide, as a power boost fuel for military aircraft, was in effect operationally at the end of the war and was also being used to provide a 15-20% thrust increase for the Schmidding rocket engine.

4. Prof. Busemann - LFA, Braunschweig

Prof. Busemann was primarily concerned with the thermodynamics of rocket propulsion, especially in connection with the development of the proper fuels. He was of the opinion that for maximum jet velocities, a combination of high pressures and temperatures was necessary. Reasoning that high energy content propellants necessarily produce high temperatures, and that at high temperatures much of the energy goes into the dissociation of molecules and is not available for accelerating the jet. The only means of preventing this disassociation is through the introduction of high pressures - hence Busemann's conclusion. It is entirely possible that weight considerations might force a modification of this theory and this was the basis of practical experiments which were being conducted at Peenemunde.

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Busemann advocated a high expansion rate which would permit the utilization of heavy molecules with poor Cp/Cv values, including carbon-dioxide and even vaporized oxides of metals. This theory was in contradiction to the basic work of the Peenemunde group who used moderate pressures (20 atmospheres) and moderate temperatures, controlled by the addition of water. They were, however, able to retain high values of T/M because water reduces the average molecular weight more rapidly than it reduces the temperature.

5. Dr. Rudolph Edse - LFA, Braunschweig

Dr. Edse, in opposition to Dr. Busemann had been concerned with improving the combustion stability of solid propellant at pressures of 20 atmospheres, in order to decrease the weight of the combustion chamber, and in this connection had worked very closely with the Peenemunde group.

APPENDIX A

CIOS Report No.	Title
Item No.25,26 File No.IV-1,8	Gas Turbine and Jet Propulsion Work in Paris. 29 Aug. - 8 Sept. 1944.
Item No.5,26 File No.XI-6, XII-9, XIV-4	Junkers 004 (203) Jet Propulsion Engines. 5 December 1944.
Item No.5 File No.XXI-5	Heinkel-Hirth TL Gas Turbine Engine 13 April 1945.
Item No.5,26 File No.XXIV-6	Gas Turbine Development at B.M.W., Junkers, Daimler Benz.
Item No.5,19 File No.XXV-23	Junkers Flugzeug und Moternwerke A.G.Ausbildung, Dessau.
Item No.5 File No.XXVI-27	Research and Development of Engines at Hermann Goering Institute, Volkenrode.
Item No.5 File No.XXVI-28	Research and Development on Gas Turbines at Hermann Goering Institute, Vulkenrode.
Item No.5 File No.XXVI-29	Research and Development on Gas Turbines at Junkers Moternwerke.
ADI(k) Report 354, 1945	German Progress in Field of Gas Turbines, Athodyds and Turbo-Jets.

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APPENDIX B

BIBLIOGRAPHY OF GERMAN PUBLICATIONS ON ATHODYDS

Sänger and Collaborators

Report	Summary of Contents
U&M 3509	Calculations and description of experiments with ramjets.
U&M 3538	Rocket power for a long range bomber.
U&M 3557	Ramjet power for the Me-262.
FB 1958	Application of ramjets to unmanned aircraft.

Focke-Wulf (Pabst)

Focke-Wulf Ber. 0940	Ramjet calculations without through flow.
Focke-Wulf Ber. 0941	Ramjet calculations with through flow.
Focke-Wulf Ber. 0945	Results of F.W. ramjet tests in LFA tunnel A-9.

Oswatitsch (Supersonic Ramjets).
(From the series: Forschungen und Entwicklungen des Heereswaffentamtes)

Ber. 1005	Theoretical conceptions and preliminary experiment results.
Ber. 1010	Results of further experiments.
Ber. 1010/2	Additional experiments and summary of all results.
FB 1236 and 1736 (each in several parts)	A summary of his work was stated by Kuchemann to have appeared as a Technische Bericht for inclusion in the 1943 Jahrbuch des Deutschen Luftfahrtforschung. This report has not yet been located. He is also

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	preparing a summary report of his work for Col. Paul Dane, AAF, for inclusion in the USSTAF files.
Walter - Kiel U&M 2014	Tests on Walter ramjet in LFA tunnel A-9 (Also Walter-report 1.631)
CIOS Report Item 25 File No.XXVI-6	Focke-Wulf Designing Offices and General Management at Bad Eilsen
USSTAF Report TIR No.I-82	Interrogation of Dr. Lippisch Regarding Design and Development of the Athodyd.
USSTAF Report TIR No.I-83	Interrogation of Dr. Oswatitsch of KWI, Göttingen on Supersonic Diffusors for Athodyds.
Nav.Tech.Mis.Ev. Report 95-45	Survey of German Ramjet Developments.

Note:

1. U & M reports refer to Untersuchungen und Mitteilungen Reports published by ZWB.

2. Ber. reports refer to Technische Bericht reports.

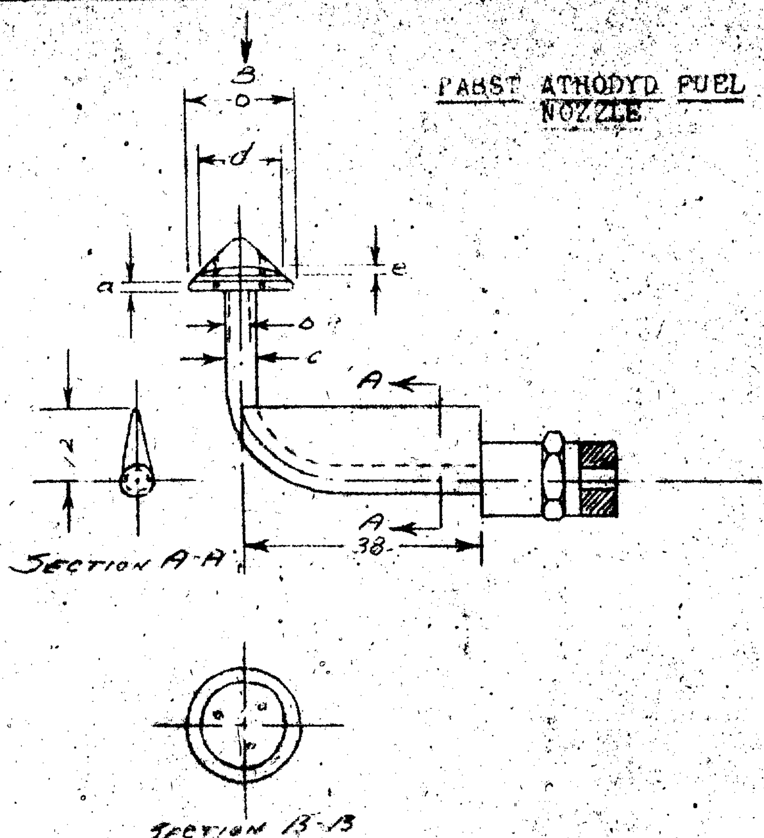
APPENDIX C

Available reports on German Rockets

Report	Title
AI2(g) Report 2323	Me-163 Rocket Propelled Intercepter.
AI2(g) Report 3492	German Long Range Rocket -V-2.
AI2(g) Report 1735	German 8 cm. Rocket Projectile.
AI2(g) Report dated 18 April 1945	German Liquid Rocket Fuels
EWD Report dated 7 November 1944	Notes on German Rocket Production.
MEW Report dated 10 April 1945	V-2 Manufacture.
ADRC Doc. 1218	Correspondence on production, delivery, launching results and failures of V-2.
ADRC Doc. 1193	BMW Rocket Research and Development.
ADRC Doc. 1318	Details of Walter 109-509 A Rocket.

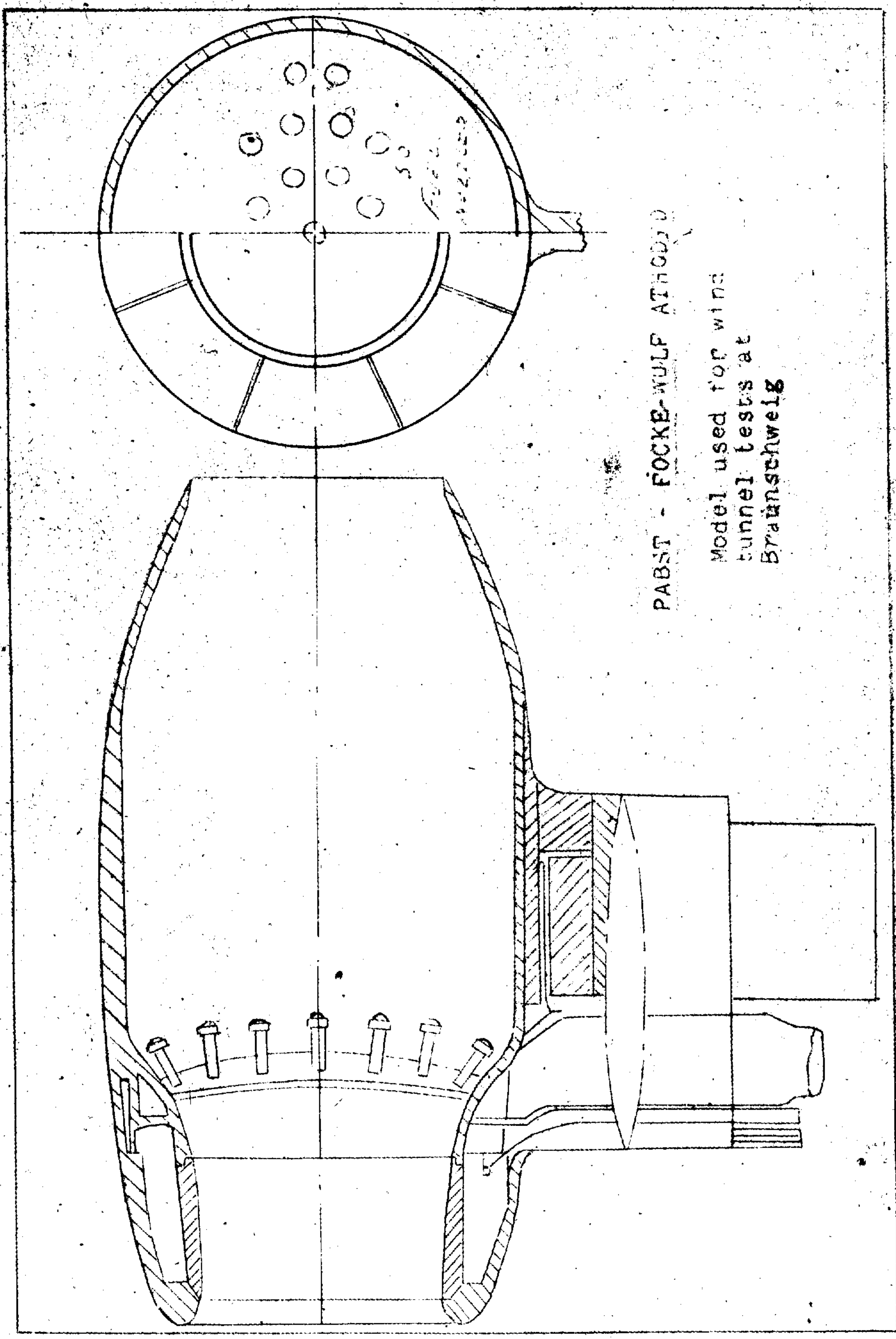
Note
1. AI2(g) are British Aircraft Intelligence Reports.
2. EWD are Economic Warfare Department Reports.
3. ADRC Documents are those available at the Air Document Research Center, 59 Weymouth Street, London, W.1.

PABST ATHODYD FUEL NOZZLE



o	a	b	c	d	e
9.0	0.75	2.0	3.0	7.0	0.25
11.3	0.90	2.5	3.0	8.75	0.30
12.4	1.05	2.0	4.0	9.68	0.35
15.8	1.30		5.0	12.25	0.45
18.0	1.50	4.0	5.0	14.00	0.50

Figure B.



PABST - FOCKE-WULF ATHODYD

Model used for wind tunnel tests at Braunschweig

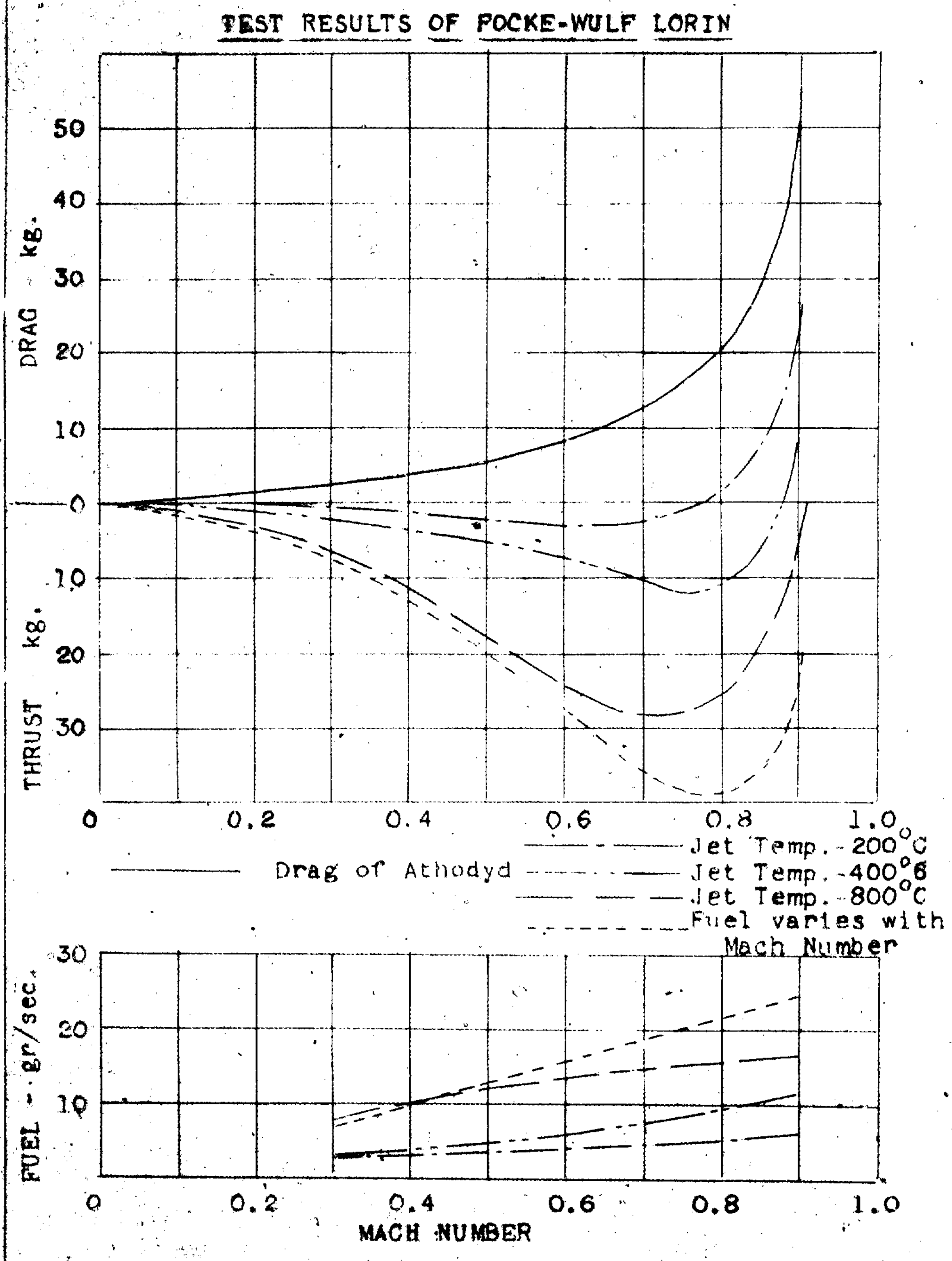
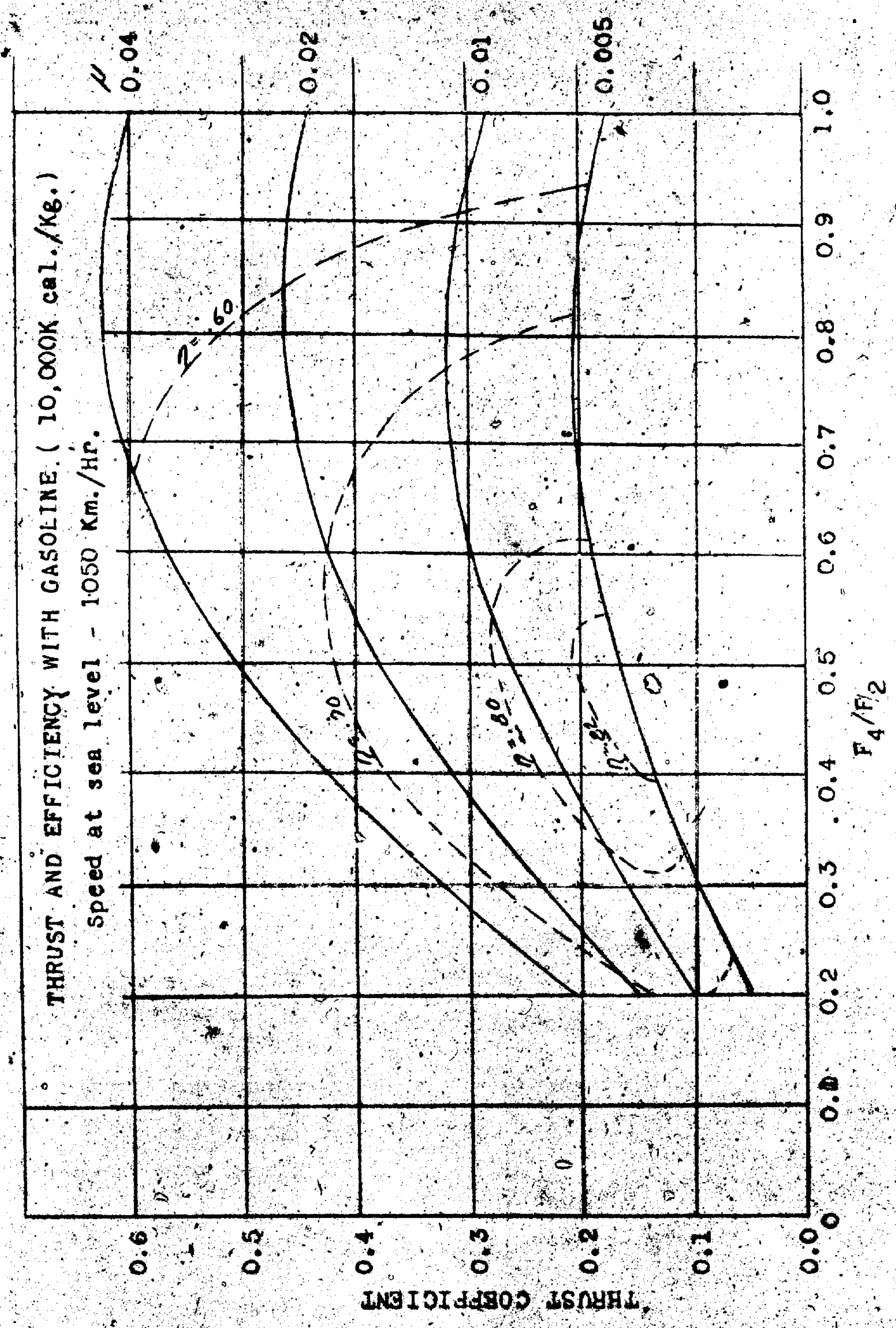
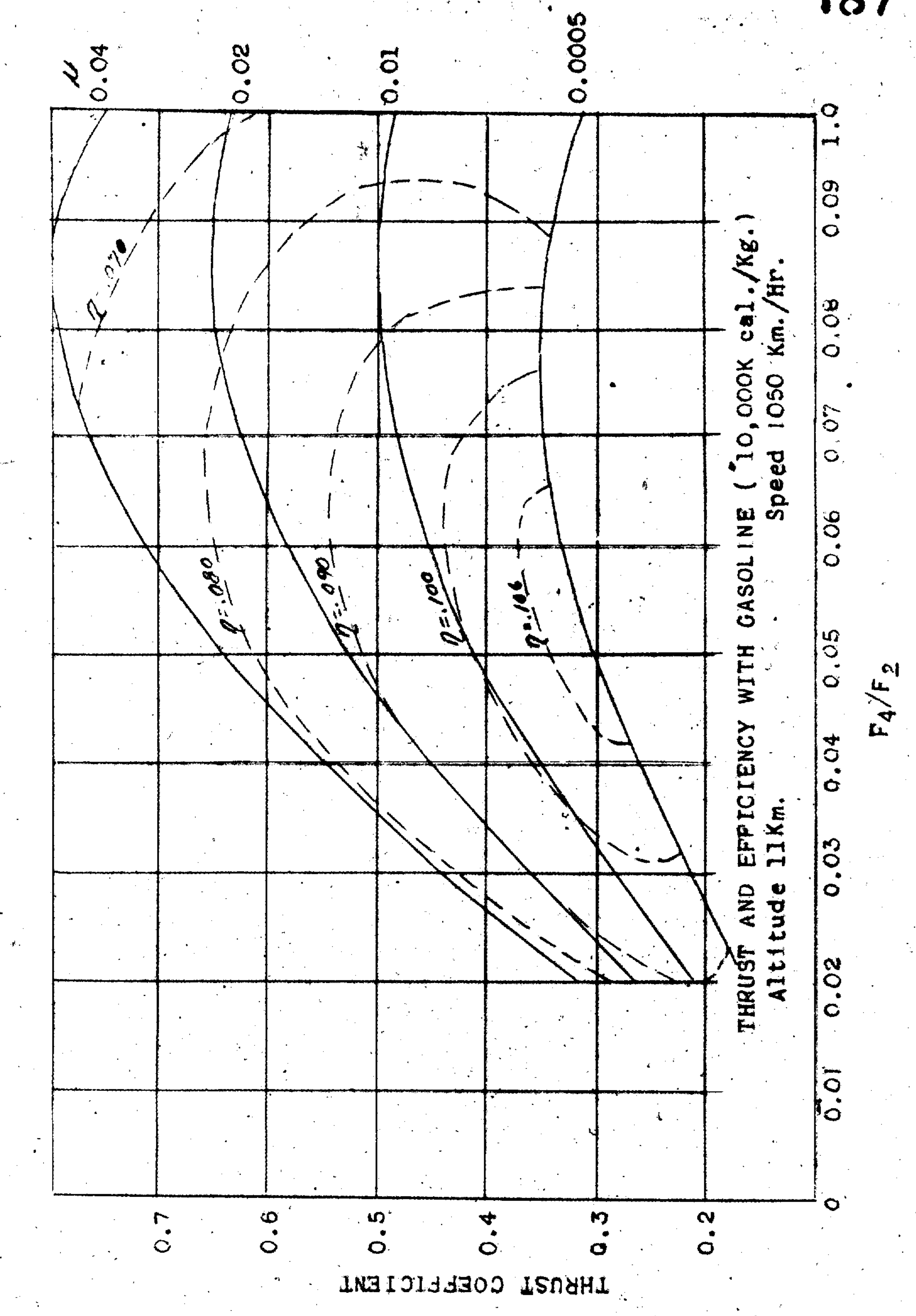

TEST RESULTS OF FOCKE-WULF LORIN
DRAG - kg.
THRUST kg.
50
40
30
20
10
0
10
20
30
0
0.2
0.4
0.6
0.8
1.0
Drag of Athodyd
Jet Temp. - 200°C
Jet Temp. - 400°C
Jet Temp. - 800°C
Fuel varies with Mach Number
FUEL - gr/sec.
30
20
10
0
0.2
0.4
0.6
0.8
1.0
MACH NUMBER

Figure D.



THRUST AND EFFICIENCY WITH GASOLINE (10,000K cal./Kg.)
Speed at sea level - 1050 Km./Hr.
THRUST COEFFICIENT
F4/F2
μ
0.04
0.02
0.01
0.005
0.0
0.1
0.2
0.3
0.4
0.5
0.6
0.7
0.8
0.9
1.0
η=60
η=70
η=80

Figure E.


THRUST AND EFFICIENCY WITH GASOLINE (10,000K cal./Kg.)
Speed 1050 Km./Hr.
Altitude 11Km.
F_4/F_2
THRUST COEFFICIENT
μ
0.04
0.02
0.01
0.0005
η=.070
η=.080
η=.090
η=.100
η=.106
0 0.01 0.02 0.03 0.04 0.05 0.06 0.07 0.08 0.09 1.0
0.2 0.3 0.4 0.5 0.6 0.7