Sept. 20, 1966 F. I. TANCZOS ETAL 3,273,334

RAMJET MISSILE

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FIG. 1

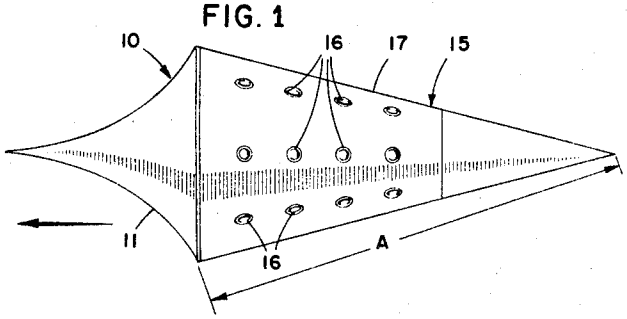


FIG. 2

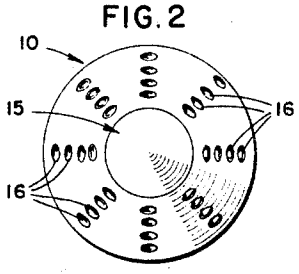


FIG. 3

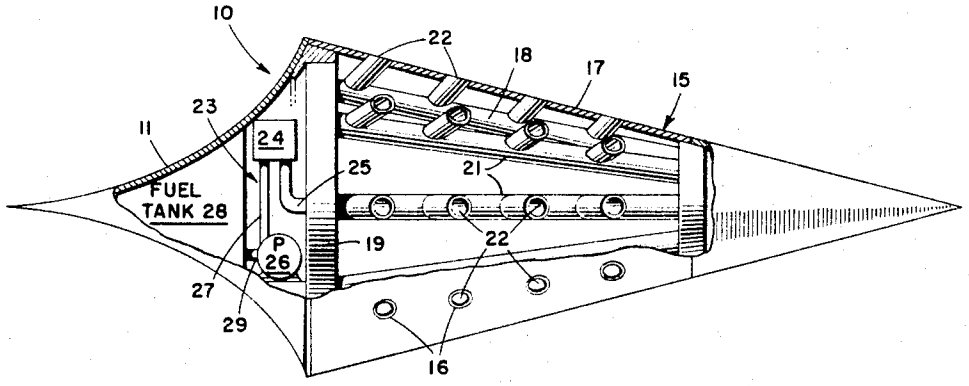
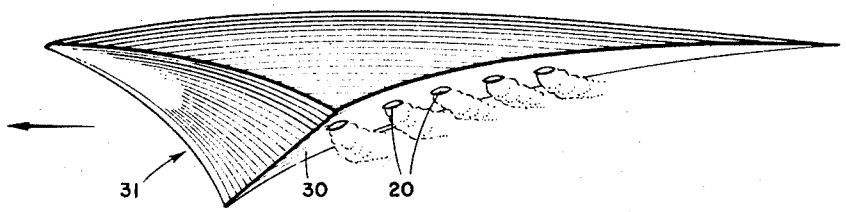


FIG. 4



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FIG. 5

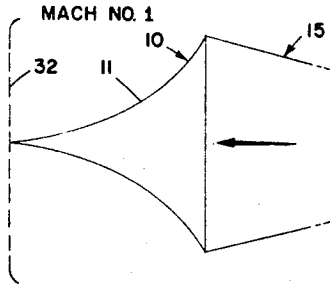


FIG. 7

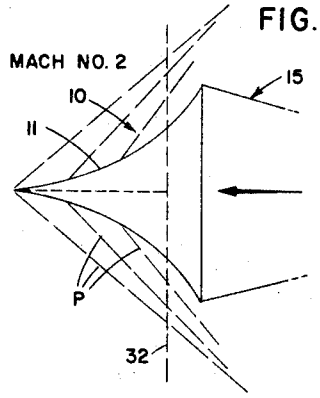


FIG. 6

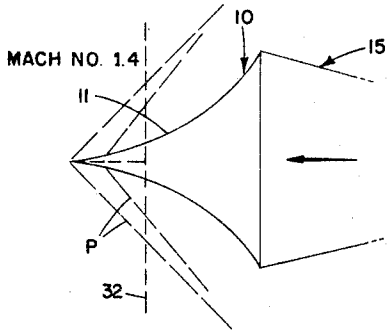


FIG. 8

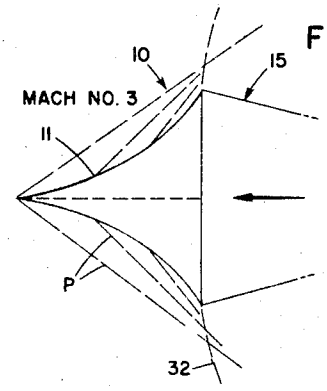
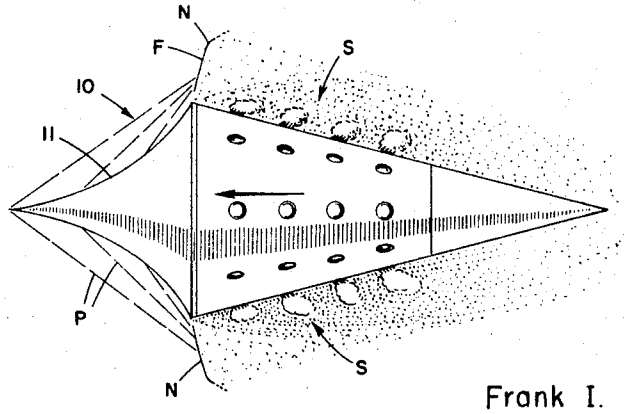


FIG. 9



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3,273,334

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Frank I. Tanczos, Washington, D.C., Thomas Davis and Edward H. Smith, Silver Spring, Md., and James W. Mullen II, Richmond, Va., assignors, by direct and mesne assignments, to the United States of America as represented by the Secretary of the Navy

Filed Sept. 10, 1959, Ser. No. 839,260
2 Claims. (Cl. 60—35.6)

This application is a continuation-in-part of application Serial No. 480,018 for Ramjet Missile, now Patent No. 3,008,669, filed in the names of Frank I. Tanczos, Thomas Davis, Edward H. Smith and James W. Mullen II, on January 5, 1955.

This invention relates to a reactive propulsion engine and more particularly to a supersonic ductless airfoil capable of obtaining combined thrust and lift forces at fuel specific impulse values considerably higher than those possible with conventional ramjets or rockets.

The present state of development in ramjet engines requires that a complicated fuel supply and distribution system be combined with a complex ignition and flame-spreading system to augment and sustain supersonic flight. The combustion problems encountered in the present state of the ramjet art are numerous, particularly with respect to flame propagation and fuel supply in the higher supersonic ranges. Furthermore, in order to govern effectively the ram inlet air to suit the variable conditions encountered at supersonic flow rates, ramjet engines employ a shell in the form of an elongated tubular duct designed to accommodate a convergent-divergent type diffuser at the ram air inlet which may be mounted within the interior walls of the tubular duct, or a nozzle throat member may be coaxially aligned within the nozzle throat capable of automatic adjustment to control inlet air according to the supersonic speeds at which the ramjet travels. The express purpose for these diffusers is to control the rammed air inlet so that suitable compression ratios may be formed in the various zones within the ram duct at the supersonic speed ranges and the kinetic energy of the inlet rammed air may be utilized as potential energy in the thermic cycle within the internal combustion chamber.

Since the ramjet unit only reaches its practical efficiency as a propulsive device at translational velocities above the speed of sound, an auxiliary launching apparatus is usually necessary to augment flight of the projectile or missile prior to utilizing the ramjet power plant. Upon reaching the speed of sound or a Mach number of 1, the ramjet inlet is subjected to a shock front attendant at these high translational velocities and the shock wave effects a compression of a large volume of air to be supplied to the combustion chamber. The magnitude of the compression ratio at a Mach number of 1.7 will approximate 3.6 to 1 and at a Mach number of 3.0 the compression ratio can be greater than 17.0 to 1. Effective control of the compression ratio to compensate for increase or decrease in ramjet speed is performed by the position and character of shock waves impinging within the ram air inlet.

The present invention is predicated upon the principle that through considerable external burning at or near the surface of a supersonic airfoil structure sufficient thrust can be obtained with practical efficiency to enable an airfoil to travel at supersonic speed. Moreover, the invention provides for release of heat into the airstream near an aerodynamic body in the subsonic or supersonic region to produce a thrust on the body together with a desirable drag reduction. The "inside-outside" ramjet of the present invention employs a supersonic airfoil, without the conventional ducted ram inlet, in which the supersonic stream breaks down ahead of the combustion region

and burning may then occur subsonically and the exhaust products, that is, the gases and products of combustion and the heated air, expand to ambient pressure to produce a substantial forward thrust on the airfoil structure by means of heat release in the airstream near the airfoil body, known as the ramjet thrust produced by external burning. This external burning is produced by utilizing an acetylenic type of rocket monopropellant or any other suitable fuel wherein the initial burning which will take place within the interior of the airfoil structure will produce a rocket thrust by the rapidly expanding exhaust products of combustion discharging into the adjacent high pressure airstream. The exhaust products, composed largely of carbon and hydrogen, will be self-ignitible upon contacting the airstream adjacent the airfoil thereby producing a ramjet thrust and when the thrust is resolved with the rocket thrust a net resultant will be produced which is a combination of the forward thrusts attributed to both the ramjet action and the rocket thrust action. Alternatively, a solid fuel propellant may be substituted for the liquid fuel (acetylene or hydrazine) suggested above, in which case the burning powder grain fragments may be ejected through the discharge orifices of the burner housing while still in the partially consumed state. The final stage of powder burning, with continuing release of its own combustion supporting oxygen supply, would then occur in the external atmosphere immediately to the rear of the airfoil surface, to produce a ramjet thrust in a manner similar to that produced by the liquid fuel combustion heretofore described. As a further alternative, gaseous fuels, such as hydrogen, and liquid fuels, such as aluminum borohydride may replace the liquid or solid fuels above mentioned, or the fuel employed may be a combination of all three types, or any two of them.

Heretofore, it has not been feasible to add heat into a supersonic stream, however, this invention provides a ready means to inject heat into the airstream adjacent the combustion zone by causing a supersonic stream to break down by normal shock and then permitting the actual combustion to occur under subsonic flow conditions thereby accounting for a higher degree of flame stabilization.

From the foregoing external burning is accomplished by feeding a gaseous, liquid, or granular solid rocket fuel or monopropellant from a supply system through nozzles to the outer surface of the airfoil. If desired, initial fuel decomposition or preparation is adapted to take place within the interior of the airfoil in suitable combustion chambers to produce a rocket thrust by the rapidly expanding exhaust products of combustion discharging through nozzles into the adjacent high pressure airstream.

An object of the present invention is to obtain a thrust and lift on an aerodynamic airfoil traveling at supersonic speeds without the conventional ramjet duct.

Another object of this invention is to provide an integral airfoil structure capable of supersonic speeds with means for reducing the aerodynamic drag and producing a thrust on the airfoil body by means of heat release in the airstream adjacent the airfoil structure.

A further object of this invention is to provide an airborne body with an aerodynamic lift which may be attained more economically by external combustion rather than from a ducted motor having a wing flown at an angle of attack.

Another object is to provide a ductless airfoil structure having a rocket thrust that is produced by the internal burning of the propellant by exhausting the products of combustion into the subsonic region of the airstream adjacent the downstream section of the airfoil structure with a ramjet thrust which is imparted by burning the exhaust products of the internal combustion in the atmosphere surrounding the airfoil structure.

Other objects and many of the attendant advantages of this invention will be readily appreciated as the same becomes better understood by reference to the following detailed description when considered in connection with the accompanying drawings wherein:

FIG. 1 is a side elevational view of an embodiment of the present invention;

FIG. 2 is an end elevational view of the device of FIG. 1;

FIG. 3 is a view in diagrammatic form illustrating a fuel injection system and a fuel supply system suitable for use with the device of FIG. 1;

FIG. 4 is a perspective view of a modified embodiment of the device of FIG. 1;

FIGS. 5 through 8 are fragmentary, diagrammatic views illustrating the manner in which the reflected shock waves are directed laterally beyond the airfoil structure; and

FIG. 9 is a diagrammatic view derived from a schlieren photograph and illustrating supersonic airflow.

Referring to the drawings in which like numerals represent like elements in the various views, there is illustrated an airfoil structure **10**, traveling in the direction of the arrow, which has a shock type diffuser cone **11** for providing the requisite aerodynamic design characteristics for suitable supersonic flight speeds. The embodiment illustrated in FIGS 1 and 3 show the upstream portion of the airfoil structure as defined by the concave conical ogival nose portion **11** which will produce a normal shock wave front relative to the axis of the airfoil **10**. The trailing downstream or tail portion **15** of the airfoil, as illustrated in FIGS. 1 and 2 is a slender, tapering, conic structure containing a plurality of selectively spaced burner ports **16**, from which the products of combustion from suitable propellants emerge to the surface **17** of tail portion **15** to produce a thrust on the airfoil thereby passing the exhaust products of the propellant into the surrounding high pressure airstream.

Referring now to FIG. 3, it will be noted that a fuel injection system generally indicated by the numeral **18** is disposed within the tapered conical tail portion **15**. The aforesaid system comprises a manifold **19** having a plurality of individual branch pipes **21** carried thereby, each pipe being provided with a plurality of nozzles **22**, the open end of which is disposed within one of the aforesaid burner ports **16** with the nozzles inclined and disposed downstream thereby to supply fuel to the airstream for combustion downstream alongside the tail portion **15** subjacent the maximum area of the body to heat the airstream an amount sufficiently to cause the supersonic airstream to break down by normal shock and thus form a subsonic combustion zone downstream from the normal shock front.

A fuel supply system generally indicated by the numeral **23** is employed with the airfoil, however, this fuel system when employed as a flight augmenter in an aircraft, rocket or the like may be installed in a suitable chamber apart from the airfoil. The aforesaid system comprises a fuel regulator or meter device **24** connected to the manifold **19** by a pipe **25** and to a fuel pump **26** by a pipe **27**, the pump being connected to a fuel tank **28** by a pipe **29**. Thus, by this arrangement, fuel under pressure is fed to the manifold **14** and thence into the airstream by way of nozzles **22**. It will be noted, FIG. 2, that the nozzles **22** are inclined toward the rear or in a downstream direction and in spaced relation with respect to each other and extend longitudinally along the surface **17** of the tail portion **15**. As the decomposition products from the fuel are injected into the airstream through nozzles **22**, ignited and burning occurs downstream subjacent the maximum cross-sectional area of the body **10** to introduce heat into the airstream such that external combustion may occur under subsonic flow conditions in a sheltered area and thus flame stabilization is maintained. Individual rocket elements may be employed in the devices of FIGS. 1 and 4 if desired, these rocket elements being spaced along the tail portion thereof in a manner similar to the nozzle arrangement and as they are fired the decomposition products will be ignited and burned downstream from the rocket exhausts. It will be apparent from the foregoing that thrust and lift components are obtainable directly from the rocket exhaust as well as from the downstream combustion in accordance with the degree of rearward and downward canting of the nozzles such, for example, as when the operating nozzles are disposed on the underside of the airfoil.

In supersonic flight a resultant high pressure region will be encountered within the zone A, FIG. 1, which will provide a suitable combustion zone along the trailing surface **17** of tail portion **15**. Suitable liquid and/or solid propellants are contemplated to provide sufficient combustion and heat release, such, for example, as an acetylenic type fuel, which is of such a nature as to be completely combustible with the passing airstream. Hydrogen is a fuel which is self-igniting at about Mach 3 and aluminum borohydride is another fuel which is self-igniting at still higher flight speeds and possibly at low supersonic combustion velocities. The resultant effect of utilizing propellants of this nature is that upon supplying fuel to the various ports **16** an internal reaction occurs which will provide a substantial rocket thrust, and the products of combustion of the acetylenic monopropellants, constituted almost entirely of carbon and hydrogen, when subject to the high temperatures surrounding the structure during supersonic flight will be self-ignitible upon contact with the outside atmosphere, thereby producing the desired ramjet thrust upon the airfoil. Alternately fuels such, for example, as hydrogen, aluminum borohydride and other fuels which are spontaneously ignitible with high speed air can be used to produce the desired ramjet thrust upon the airfoil. Flight control of the airfoil may, if desired, be effected by proper positioning and throttling of the fuel ports around the trailing surface **17** as well as controlling the fuel injection to the various fuel ports. The thrust obtained by the external burning taking place within the airstream adjacent the airfoil structure to produce the ramjet thrust is independent of and in addition to the rocket thrust obtainable principally from the propellant burning within the fuel port.

The modification shown in FIG. 4 illustrates another embodiment of the present invention in which a system of burners **20** or individual rocket elements may be used on one side **30** of an oblique airfoil structure **31** to provide the requisite lift, and a modified form of shock front is produced for the rammed air which will result in the desired normal shock wave ahead of the high compression subsonic region of the burner ports.

FIGS. 5 through 8 illustrate the resulting positions of shock waves as the forward speed of the airfoil structure increases to provide efficient compression of the air by the ram action of the shock waves at the different speeds in the supersonic range. As the airfoil approaches the speed of sound a bow wave **32** is formed ahead of the apex of the cone **11**, FIG. 5. As the translational velocity of the airfoil increases above the speed of sound, that is, at Mach number 1.4 as shown in FIG. 6, the point of convergence of the shock wave moves rearwardly, as illustrated in FIGS. 7 and 8; however, beyond the point of convergence on the downstream side, a subsonic zone results which affords suitable combustion conditions for external combustion of the products of combustion exhausted from the burner ports upon completion of the internal combustion within the airfoil structure **10**.

It will be noted in FIG. 8 that the shock wave pattern P commences at the upstream point or apex of the cone **11**, and progresses up to the flame front F in which area a normal shock wave N occurs due to the addition of a sufficient amount of heat introduced into the airstream as the fuel is injected into the airstream through the nozzle arrangement and ignited. It will be further noted

that the cone 11 is constructed and arranged such that a sheltered subsonic combustion zone S is provided behind the normal shock front since the cone 11 causes oblique deflection of the shock pattern away from the combustion zone, whereupon external combustion will be carried out under subsonic flow conditions and flame stabilization will be maintained.

Obviously many modifications and variations of the present invention are possible in the light of the above teachings. It is therefore to be understood that within the scope of the appended claims the invention may be practiced otherwise than as specifically described.

What is claimed is:

1. A reactive propulsion engine comprising a body suitable for supersonic flight having an ogival nose section and a convergent sheltered rear section with the maximum cross-sectional area of the body disposed at the junction of said nose and rear sections and normal to the longitudinal axis of the body, said body being defined on the underside thereof by a plurality of intersecting concave surfaces of revolution and on the upper side thereof by a convex surface of revolution intersecting said concave surfaces, and means disposed within said sheltered rear section for supplying combustible fuel into the airstream downstream from said maximum cross-sectional area and within said sheltered rear section to introduce heat into the airstream in an amount sufficient to breakdown the supersonic airstream by normal shock and allow continuous combustion of fuel within the sheltered rear section under subsonic conditions.

2. A reactive propulsion engine comprising a unitary ductless body for supersonic flight having an ogival nose section and a convergent sheltered tail section with the maximum cross-sectional area of the body disposed at the junction of the nose and tail sections, a plurality of intersecting concave surfaces on the underside of the body, a convex surface on the upper side of the body and intersecting said concave on the underside thereof, and means including a plurality of ports disposed in the sheltered tail section for supplying combustible fuel to the tail section to introduce heat into the airstream near the surface of the tail section in an amount sufficient to break down the supersonic airstream by normal shock to allow continuous combustion of the fuel within the sheltered tail section under subsonic conditions.

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